SMR151

JET Star

FAA APPROVED
HANDBOOK
OF

OPERATING AND MAINTENANCE INSTRUCTIONS

THIS MANUAL HAS BEEN APPROVED BY THE FAA AS BEING IN COMPLIANCE WITH ACCEPTABLE PROCEDURES FOR STRUCTURAL REPAIR AND MAINTENANCE OF THE JETSTAR, HOWEVER, THE MANUAL IS PRIMARILY INTENDED AS A GUIDE FROM WHICH AN OPERATOR MAY DEVELOP PROCEDURES TO SUIT HIS OWN OPERATION, PROVIDED METHODS USED COMPLY WITH FAR 43. IT SHOULD NOT, THEREFORE, BE CONSTRUCTED THAT THE MANUAL SETS FORTH GUARANTEED OR VALIDATED PROCEDURES OR THE ONLY PROCEDURES WHICH MAY BE FOLLOWED. MANUFACTURER RESERVES THE RIGHT TO RECOMMEND REVISIONS TO THE MANUAL AS SEEM APPROPRIATE FROM TIME TO TIME.

Lockheed Martin Aircraft & Logistics Centers
GREENVILLE, SOUTH CAROLINA 29605

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1 MAY 1964
CHANGED 31 MAY 2005

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Page No.	*Issue*
Title	15 Dec 96
*A	31 May 05
B – E	15 Dec 96
F	19 Oct 90
G	30 Dec 95
H	30 Dec 95
J	15 Dec 98
*K – L	31 May 05
M	18 Nov 93
N Blank	18 Nov 93
i thru ii	Original
iii	1 Apr 68
Iv Blank	1 Apr 68
Table of Contents 1	20 Dec 91
Table of Contents 2	30 Oct 87
*Table of Contents 3	31 May 05
Table of Contents 4	31 Aug 83
Table of Contents 5 Added	31 Aug 83
Table of Contents 6 Blank Added	31 Aug 83
Introduction 1	Original
Introduction 2	Original
Introduction 2A Added	18 Nov 93
Introduction 2B Added	18 Nov 93
*Introduction 3	31 May 05
Introduction 4 thru Introduction 8	15 Dec 98
Introduction 9 thru Introduction 10B Deleted	1 Jan 67
Introduction 11 thru Introduction 14	1 Oct 85
5-1	16 Oct 89
5-2 Blank	16 Oct 89
*5-3	31 May 05
*5-4 Blank	31 May 05
5-5	31 Aug 83
5-6 Blank	31 Aug 83
5-7 Added	1 Feb 65
5-8 Blank Added	1 Feb 65
5-9 thru 5-10	31 Aug 83
5-11	1 Oct 88
*5-12	31 May 05
5-13	18 Nov 933
5-14	20 Dec 91
5-14A	25 Oct 88
5-14B Blank	25 Oct 88
5-15	20 Dec 91
*5-16 thru 5-17	31 May 05
5-18	30 Dec 96
5-19	1 Oct 85
*5-20	31 May 05
5-21 thru 5-22	30 Dec 96
5-22A	30 Oct 87
*5-22B	31 May 055
5-23	30 Dec 96
5-24	1 Oct 86
5-25	31 Aug 83
5-26	20 Dec 91
5-27	15 Dec 98
5-28	30 Oct 87
5-29	18 Nov 93
5-30	31 Aug 83
5-31	1 July 69
*5-32 thru 5-33	31 May 05
5-34	1 Oct 85
5-35	30 Dec 94
5-36 Added	31 Aug 83
*5-37	31 May 05
*5-38 thru 5-39/5-40 Added	31 May 05
6-1 thru 6-8	Original
6-9	31 Aug 83
6-10 Blank	31 Aug 83
7-1	1 July 69
7-2 Blank	1 July 69
7-3	19 Oct 90
7-4	15 Dec 98
7-5	Original
7-6 thru 7-7	31 Aug 83
7-8 thru 7-10	Original
7-11	30 Dec 95
7-12 Blank	30 Dec 95
7-13	1 July 69
7-14	1 Oct 85
7-15	16 Oct 89
7-16 thru 7-18	1 July 69
8-1	Original
8-2 Blank	Original
8-3 thru 8-4	Original
8-5	1 Feb 65
8-6	1 Jan 66
8-7	Original
8-8 Blank	Original
9-1	Original
9-2 Blank	Original
*9-3 thru 9-4	31 May 05
9-5	Original
9-6 Blank	Original
9-7	19 Oct 90
9-8	Original
10-1	31 Aug 83
10-2 Blank	31 Aug 83
10-3	30 Oct 87
10-4 Blank	30 Oct 87
10-5	Original
10-6 Blank	Original
10-7 thru 10-8	Original
10-9	16 Oct 89
10-10 Blank	16 Oct 89
12-1	31 Aug 83
12-2 Blank	31 Aug 83
12-3	1 Oct 85
12-4	Original
12-5	31 Aug 83
12-6	1 Oct 85
12-7 thru 12-8	30 Dec 96
12-9	Original
12-10	1 Apr 68
12-11	1 Oct 85
12-12 Blank	1 Oct 85
12-13	16 Oct 89
12-14	1 July 69
12-15	Original
12-16 Blank	Original
12-17	31 Aug 83
12-18	1 Oct 86
12-19	1 Oct 85
12-20	16 Oct 89
12-20-1 Added	16 Oct 89
12-20-2 Blank Added	16 Oct 89
12-20A	1 Oct 86
12-20B thru 12-20D Added	1 Oct 85
12-21	1 Oct 85
12-22 thru 12-23	1 July 69
12-24	1 Apr 68
12-25 thru 12-26	1 Oct 85
12-27	1 Dec 65
12-28	1 Apr 68
12-29	1 Oct 85
12-30	1 Dec 65
12-30A Added	1 Dec 65
12-30B Blank Added	1 Dec 65
12-31	28 Oct 88
12-32	18 Nov 93
12-32A	16 Oct 89

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LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the changes is indicated by a vertical line in the outer margins of the page. Changes to illustrations are indicated by miniature pointing hands.

Page No.	*Issue*
12-32B Blank	16 Oct 89
12-33 thru 12-34	1 July 69
12-34A	31 Aug 83
12-34B Blank	31 Aug 83
12-35	31 Aug 83
12-36 thru 12-37	1 Oct 85
12-38 Blank	1 Oct 85
12-39	16 Oct 89
12-40	Original
12-40A	16 Oct 89
12-40B Blank	16 Oct 89
12-41	1 Oct 86
*12-42	15 Dec 98
12-43	1 Oct 85
12-44	19 Oct 90
12-44A thru 12-44B	1 Oct 85
12-44C	18 Nov 93
12-44D	19 Oct 90
12-44E thru 12-44F Added	31 Aug 83
12-45 thru 12-46	Original
12-47	31 Aug 83
12-48	18 Nov 93
12-49 thru 12-50	16 Oct 89
12-50A thru 12-50C Added	16 Oct 89
12-50D Blank Added	16 Oct 89
12-51	19 Oct 90
12-52	31 Aug 83
12-53 thru 12-54	16 Oct 89
12-55	Original
12-56	31 Aug 83
12-57	19 Oct 90
12-58 thru 12-65	31 Aug 83
12-66	16 Oct 89
12-67 thru 12-72	31 Aug 83
12-73 thru 12-74	1 Dec 65
12-74A thru 12-74D	31 Aug 83
12-75	Original
12-76 thru 12-77	31 Aug 83
12-78	1 July 69
12-79	31 Aug 83
12-80 Blank	31 Aug 83
12-81 thru 12-152 Deleted	31 Aug 83
12-153	19 Oct 90
12-154 Blank	19 Oct 90
21-1	Original
21-2 Blank	Original
21-3	1 June 67
21-4	16 Oct 89
21-5	31 Aug 83
21-6	1 June 67
21-7	1 July 69
21-8	1 Apr 68
21-8A Deleted	1 June 67
21-8B Blank Deleted	1 June 67
21-9 thru 21-10	1 June 67
21-10A Deleted	1 June 67
21-10B Blank Deleted	1 June 67
21-11	31 Aug 83
21-12	1 July 69
21-13	31 Aug 83
21-14	1 July 69
21-14A Added	1 July 69
21-14B Blank Added	1 July 69
21-15	1 July 69
21-16	1 June 67
21-17	30 Dec 95
21-18	1 Dec 69
21-18A thru 21-18B	31 Aug 83
21-18C	5 Oct 84
21-18D	16 Oct 89
21-19	30 Dec 96
21-20	1 Apr 68
21-20A	1 Apr 68
21-20B Blank	1 Apr 68
21-21	1 Oct 85
21-22 Blank	1 Oct 85
21-23	31 Aug 83
21-24	Original
21-25	1 Apr 68
21-26	1 June 66
21-27	1 June 67
21-28 Blank	1 June 67
21-29	1 Dec 65
21-30 Blank	1 Dec 65
21-30A	1 Dec 65
21-30B Blank	1 Dec 65
21-30C	1 Dec 65
21-30D Blank	1 Dec 65
21-30E Added	1 Dec 65
21-30F Blank Added	1 Dec 65
21-30G	31 Aug 83
21-30H Blank	31 Aug 83
21-30J	31 Aug 83
21-30K Blank	31 Aug 83
21-31 thru 21-32	31 Aug 83
21-33	Original
21-34	1 Apr 68
21-35	Original
21-36	1 June 67
21-37 thru 21-38	Original
21-39	1 June 66
21-40	5 Oct 84
21-40A Added	31 Aug 83
21-40B Blank Added	31 Aug 83
21-41	1 June 67
21-42 thru 21-44	31 Aug 83
21-45	1 June 66
21-46	1 June 67
21-47	1 Oct 85
21-48	31 Aug 83
21-49	28 Oct 88
21-50 Blank	28 Oct 88
21-51	1 Apr 68
21-52	31 Aug 83
21-52A	31 Aug 83
21-52B Blank	31 Aug 83
21-53	16 Oct 89
21-54	31 Aug 83
21-54A Added	31 Aug 83
21-54B Blank Added	31 Aug 83
21-55	1 Apr 68
21-56	5 Oct 84
21-57	1 June 66
21-58	1 June 68

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LIST OF EFFECTIVE PAGES

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Page No.	*Issue*
21-58A Added	1 June 66
21-58B Blank Added	1 June 66
* 21-59 thru 21-60	15 Dec 98
22-1	1 July 69
22-2 Blank	1 July 69
22-2A	31 Aug 83
22-2B	1 July 69
22-2C Added	1 July 69
22-2D Blank Added	1 July 69
22-3 thru 22-4	1 July 69
22-5 thru 22-6	Original
22-7 thru 22-8	31 Aug 83
22-8A Added	31 Aug 83
22-8B Blank Added	31 Aug 83
22-9 thru 22-16	Original
22-17	16 Oct 89
22-18	1 Dec 65
22-19	1 Dec 69
22-20	1 July 69
22-21	1 Dec 69
22-22	1 July 69
22-23 thru 22-26	1 Dec 69
22-26A	1 Dec 69
22-26B Blank Added	1 Dec 69
22-27	1 Dec 69
22-28 thru 22-30	1 July 69
22-31 thru 12-32	1 Dec 69
22-32A Added	1 Dec 69
22-32B Added	1 Dec 69
22-33 Added	1 July 69
22-34 Blank Added	1 July 69
23-1	1 Dec 65
23-2 Blank	1 Dec 65
23-3	1 Dec 65
23-4 Blank	1 Dec 65
23-5 thru 23-8	1 Dec 65
23-9 thru 23-13	Original
23-14	1 Dec 65
23-15	Original
23-16 Blank	Original
23-17	1 Dec 65
23-18 Blank	1 Dec 65
23-19	1 Feb 65
23-20	31 Aug 83
23-20A Added	31 Aug 83
23-20B Blank Added	31 Aug 83
23-21	31 Aug 83
23-22 thru 23-24	Original
23-25	1 Feb 65
23-26	Original
23-27	1 July 69
23-28 Added	1 Dec 65
23-29	1 June 67
23-30 Added	1 Dec 65
23-31 Added	1 Dec 65
23-32 Blank Added	1 Dec 65
23-33	1 June 67
23-34 Added	1 Dec 65
23-35 Added	1 Dec 65
23-36 Blank Added	1 Dec 65
24-1	30 Oct 87
24-2 Blank	30 Oct 87
24-3	28 Oct 88
24-4 thru 24-5	31 Aug 83
24-6	18 Nov 93
24-6-1	18 Nov 93
24-6-2 Blank	18 Nov 93
24-6A	30 Oct 87
24-6B	31 Aug 83
24-7	1 Dec 65
24-8	31 Aug 83
24-9	1 Oct 86
24-10	Original
24-10A Added	1 June 67
24-10B Blank Added	1 June 67
24-11	1 Oct 85
24-12	31 Aug 83
24-13	Original
24-14	1 Dec 65
24-15 thru 24-16	31 Aug 83
24-17 Deleted	Original
24-18 Blank Deleted	Original
24-19	Original
24-20	1 June 67
24-21	1 Dec 65
24-22 Blank	1 Dec 65
24-23	1 Dec 65
24-24 Blank	1 Dec 65
24-25	31 Aug 83
24-26 Blank	31 Aug 83
24-27	31 Aug 83
24-28 Blank	31 Aug 83
24-28A Added	31 Aug 83
24-28B Blank Added	31 Aug 83
24-29 thru 24-30	31 Aug 83
24-30A Added	31 Aug 83
24-30B Blank Added	31 Aug 83
24-31	31 Aug 83
24-32	Original
24-32A	1 June 67
24-32B Added	1 Dec 65
24-33	1 Dec 65
24-34	31 Aug 83
24-34A	31 Aug 83
24-34B Blank	31 Aug 83
24-35	31 Aug 83
24-36 thru 24-38	Original
24-38A Added	1 Dec 65
24-38B Blank Added	1 Dec 65
24-39	1 Dec 65
24-40	Original
24-41	Original
24-42 Blank	Original
24-43 thru 24-44	Original
24-45	31 Aug 83
24-46 Blank	31 Aug 83
24-47	31 Aug 83
24-48 Blank	31 Aug 83
24-49	5 Oct 84
24-50	Original
24-51	Original
24-52	1 Oct 85
24-52A	1 July 69
24-52B Blank	1 July 69
24-53	1 Dec 65
24-54	1 June 67
24-55	Original
24-56 Blank	Original
24-57 thru 24-58	Original
24-59	31 Aug 83

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Page No.	Issue
24-60 Blank	31 Aug 83
24-60A Added	31 Aug 83
24-60B Blank Added	31 Aug 83
24-60C Added	31 Aug 83
24-60D Blank Added	31 Aug 83
24-61	31 Aug 83
24-62 Blank	31 Aug 83
24-63	31 Aug 83
24-64 Blank	31 Aug 83
24-65	Original
24-66 Blank	Original
24-67	5 Oct 84
24-68 Blank	5 Oct 84
24-69 thru 24-72 Added	30 Oct 87
25-1	Original
25-2 Blank	Original
25-3	Original
25-4 Blank	Original
25-5	Original
25-6 Blank	Original
25-7	31 Aug 83
25-8 Blank	31 Aug 83
26-1	31 Aug 83
26-2	30 Oct 87
26-3	Original
26-4 thru 26-5	Original
26-6	1 Oct 86
26-7 thru 26-8	Original
26-9	1 July 69
26-10 Blank	1 July 69
27-1	31 Aug 83
27-2 Blank	31 Aug 83
27-3	5 Oct 84
27-4	31 Aug 83
27-5	18 Nov 93
27-6	1 Dec 65
27-6A	31 Aug 83
27-6B Blank	31 Aug 83
27-7 thru 27-8	31 Aug 83
27-9	5 Oct 84
27-10	18 Nov 93
27-11	31 Aug 83
27-12	5 Oct 84
27-13	Original
27-14	Original
27-15	16 Oct 89

Page No.	Issue
27-16	31 Aug 83
27-17	28 Oct 88
27-18	18 Nov 93
27-18A	1 June 66
27-18B Blank	1 June 66
27-19	Original
27-20	31 Aug 83
27-20A Added	31 Aug 83
27-20B Blank Added	31 Aug 83
27-21	16 Oct 89
27-22	Original
27-23	31 Aug 83
27-24 thru 27-25	Original
27-26 Blank	Original
27-27	31 Aug 83
27-28	Original
27-29	1 Dec 69
27-30	28 Oct 88
27-31	31 Aug 83
27-32 Blank	31 Aug 83
27-33	Original
27-34	Original
27-35	16 Oct 89
27-36	31 Aug 83
27-36A	30 Oct 87
27-36B Blank	30 Oct 87
27-37	30 Oct 87
27-38	31 Aug 83
* 27-39 thru 27-40	15 Dec 98
* 27-40A Added	15 Dec 98
* 27-40B Blank Added	15 Dec 98
27-41	31 Aug 83
27-42	30 Dec 95
27-42A	31 Aug 83
27-42B Blank	31 Aug 83
27-43	31 Aug 83
27-44	Original
27-45	31 Aug 83
27-46	30 Dec 96
27-46A	30 Oct 87
27-46B Blank	30 Oct 87
27-47	1 July 69
27-48 Blank	1 July 69
27-49	Original
27-50	1 July 69
27-51 thru 27-52A	31 Aug 83
27-52B Blank	31 Aug 83
27-53	Original

Page No.	Issue
27-54	31 Aug 83
27-55	1 Jan 67
27-56	1 June 66
27-56A Added	1 June 66
27-56B Blank Added	1 June 66
27-57	31 Aug 83
27-58	5 Oct 84
27-58A thru 27-58B	30 Dec 96
27-58C	31 Aug 83
27-58D Blank Added	31 Aug 83
27-59 thru 27-60	28 Oct 88
27-60A Added	28 Oct 88
27-60B Blank Added	28 Oct 88
27-61	Original
27-62	28 Oct 88
27-63 thru 27-64	16 Oct 89
27-64A	18 Nov 93
27-64B Blank	18 Nov 93
27-65	Original
27-66	5 Oct 84
27-67 thru 27-77	Original
27-78	31 Aug 83
27-79	Original
27-80	30 Dec 95
27-81	19 Oct 90
27-82 thru 27-83	1 Oct 85
27-84 thru 27-85	Original
27-86	30 Dec 94
27-87	18 Nov 93
27-88	5 Oct 84
27-88A	5 Oct 84
27-88B	1 Oct 85
27-89	31 Aug 83
27-90	1 July 69
27-91	Original
27-92 Blank	Original
27-93	30 Dec 95
27-94 Blank	30 Dec 95
27-95	30 Dec 95
27-96 Blank	30 Dec 95
27-97	1 June 67
27-98	30 Dec 95
27-99	1 June 67
27-100	30 Dec 95
27-101	1 Oct 86

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Page No.	*Issue*
27-102 Blank	1 Oct 85
27-102A Added	31 Aug 83
27-102B Blank Added	31 Aug 83
27-102C Added	31 Aug 83
27-102D Blank Added	31 Aug 83
27-103 thru 27-104	28 Oct 88
27-104A	18 Nov 93
27-104B Blank	18 Nov 93
27-105	Original
27-106	15 Dec 98
27-106A Added	15 Dec 98
27-106B Blank Added	15 Dec 98
27-107	31 Aug 83
27-108 Blank	31 Aug 83
27-109	Original
27-110 thru 27-114	1 June 67
27-114A Added	1 June 67
27-114B Blank Added	1 June 67
27-115	1 June 67
27-116	1 June 67
27-117	Original
27-118 Blank	Original
27-119	1 Apr 68
27-120 Blank	1 Apr 68
27-121	1 June 67
27-122 Blank	1 June 67
27-123	1 July 69
27-124 thru 27-130	1 June 67
27-131	1 Oct 86
27-132 Blank	1 Oct 86
27-133 thru 27-134	31 Aug 83
28-1	Original
28-2 Blank	Original
28-3	31 Aug 83
28-4	5 Oct 84
28-4A Added	31 Aug 83
28-4B Blank Added	31 Aug 83
28-5 thru 28-6	Original
28-7 thru 28-8	31 Aug 83
28-9	Original
28-10	30 Oct 87
28-11	30 Oct 87

Page No.	*Issue*
28-12 thru 28-13	Original
28-14	31 Aug 83
28-15	Original
28-16 Blank	Original
28-16A Added	31 Aug 83
28-16B Blank Added	31 Aug 83
28-17	Original
28-18	1 Dec 65
28-19	30 Oct 87
28-20 Blank	30 Oct 87
28-21	16 Oct 89
28-22	1 Oct 85
28-22A Added	31 Aug 83
28-22B Blank Added	31 Aug 83
28-23	Original
28-24	31 Aug 83
28-24A Added	31 Aug 83
28-24B Blank Added	31 Aug 83
28-25	31 Aug 83
28-26 thru 28-27	5 Oct 84
28-28	Original
28-29	5 Oct 84
*28-30	31 May 05
28-31	5 Oct 84
28-32	30 Oct 87
28-33	16 Oct 89
28-34 thru 28-37	Original
28-38	1 Oct 86
28-39 thru 28-40	31 Aug 83
28-41	1 Oct 86
28-42	Original
28-43	31 Aug 83
28-44	30 Oct 87
28-44A	15 Dec 98
28-44B Blank	31 Aug 83
28-45	1 Oct 86
28-46	31 Aug 83
28-46A	1 Oct 85
28-46B Blank	1 Oct 85
28-47	1 Oct 85
28-48	31 Aug 83
28-49	1 Oct 85
28-50	31 Aug 83
28-50A	1 Oct 86
28-50B	15 Dec 98

Page No.	*Issue*
28-51 thru 28-54	31 Aug 83
28-55	1 June 66
28-56	31 Aug 83
28-57	19 Oct 90
28-58 Blank	19 Oct 90
28-59 thru 28-60	31 Aug 83
28-60A Added	31 Aug 83
28-60B Blank Added	31 Aug 83
28-61	31 Aug 83
28-62	Original
28-63	16 Oct 89
28-64 Blank	16 Oct 89
29-1	Original
29-2 Blank	Original
29-3	30 Oct 87
29-4	5 Oct 84
29-4A thru 29-4B	31 Aug 83
29-4-1	16 Oct 89
29-4-2 Blank	16 Oct 89
29-5	31 Aug 83
29-6	15 Dec 98
29-7 thru 29-8	Original
29-8A	16 Oct 89
29-8B	30 Oct 87
29-8C Added	30 Oct 87
29-8D Blank Added	30 Oct 87
29-9 thru 29-12	Original
29-13 thru 29-14	31 Aug 83
29-15	28 Oct 88
29-16 Blank	28 Oct 88
29-16A	28 Oct 88
29-16B Blank	28 Oct 88
29-17	28 Oct 88
29-18 Blank	28 Oct 88
29-18A	28 Oct 88
29-18B Blank	28 Oct 88
29-19	28 Oct 88
29-20	31 Aug 83
29-20A Added	31 Aug 83
29-20B Blank Added	31 Aug 83
29-21	19 Oct 90
29-22	28 Oct 88
29-22A Added	28 Oct 88

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LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the changes is indicated by a vertical line on the outer margins of the page. Changes to illustrations are indicated by miniature pointing hands.

Page No.	Issue
29-22B Blank Added	28 Oct 88
29-23	28 Oct 88
29-24	31 Aug 83
29-24A Added	1 June 66
29-24B Blank Added	1 June 66
29-25 thru 29-27	Original
29-28	31 Aug 83
29-29	Original
29-30	1 July 69
29-30A thru 29-30B	31 Aug 83
29-30C Added	31 Aug 83
29-30D Blank Added	31 Aug 83
29-31	1 Jan 67
29-32	1 June 66
29-32A	1 June 67
29-32B Blank	1 June 67
29-32C	31 Aug 83
29-32D Blank	31 Aug 83
29-32E	1 June 67
29-32F Blank	1 June 67
29-33	1 July 69
29-34	1 June 66
29-35	28 Oct 88
29-36	1 Oct 85
29-36A	28 Oct 88
29-36B Blank	28 Oct 88
29-37	28 Oct 88
29-38	Original
29-39	1 June 67
29-40 Blank	1 June 67
29-41	1 June 67
29-42	Original
29-43	1 June 67
29-44 Blank	1 June 67
30-1	Original
30-2 Blank	Original
30-3	Original
30-4 Blank	Original
30-5	31 Aug 83
30-6	Original
30-7	28 Oct 88
30-8	Original
30-9	31 Aug 83
30-10	5 Oct 84
30-11	31 Aug 83
30-12	5 Oct 84
30-12A	16 Oct 89
30-12B Blank	16 Oct 89
30-13	16 Oct 89
30-14	Original
30-15	31 Aug 83
30-16	Original
30-17	1 June 67
30-18	Original
30-19 thru 30-20	31 Aug 83
30-21	Original
30-22	31 Aug 83
30-22A Added	31 Aug 83
30-22B Blank Added	31 Aug 83
30-23	Original
30-24 Blank	Original
30-25	Original
30-26 Blank	Original
30-26A Added	31 Aug 83
30-26B Blank Added	31 Aug 83
30-27 thru 30-30	1 June 67
30-30A Added	1 June 67
30-30B Added	1 June 67
30-31 thru 30-33	1 June 67
30-34 Blank	1 June 67
30-35	1 June 67
30-36 Blank	1 June 67
30-37	1 June 67
30-38 Blank	1 June 67
30-39	31 Aug 83
30-40 Blank	31 Aug 83
30-41	31 Aug 83
30-42 Blank	31 Aug 83
30-42A Added	31 Aug 83
30-42B Blank Added	31 Aug 83
30-43	1 Dec 69
30-44	1 Dec 69
30-45 thru 30-47	Original
30-48 Blank	Original
31-1	1 June 67
31-2	1 Dec 65
31-3	Original
31-4 thru 31-5	1 Dec 65
31-6 Blank	1 Dec 65
31-6A	1 June 67
31-6B Blank	1 June 67
31-6C	1 June 66
31-6D Blank	1 June 66
31-6E	1 Jan 67
31-6F	1 Jan 67
31-6G Added	1 Jan 67
31-6H Blank Added	1 Jan 67
31-7	Original
31-8 Blank	Original
31-9	1 Dec 65
31-10 Blank	1 Dec 65
31-10A Added	1 Dec 65
31-10B Blank Added	1 Dec 65
31-11	1 June 66
31-12 Blank	1 June 66
31-13 Added	1 June 66
31-14 Blank Added	1 June 66
32-1	1 July 69
32-2 Blank	1 July 69
32-3	28 Oct 88
*32-4	19 Oct 90
32-5 thru 32-7	31 Aug 83
32-8	28 Oct 88
32-8A thru 32-8B	28 Oct 88
32-8C Added	28 Oct 88
32-8D Blank Added	28 Oct 88
32-9	Original
32-10	31 Aug 83
32-10A Added	1 Feb 65
32-10B Blank Added	1 Feb 65
32-11	31 Aug 83
32-12	5 Oct 84
32-12A	16 Oct 89
32-12B Blank	16 Oct 89
32-13	16 Oct 89
*32-14	19 Oct 90
*32-14A	19 Oct 90
*32-14B Blank	19 Oct 90
32-15	31 Aug 83
32-16 thru 32-17	16 Oct 89
32-18	31 Aug 83
32-19	16 Oct 89
32-20	1 June 67
32-21	Original
32-22	31 Aug 83
32-23	Original

*The asterisk indicates pages changed, added, or deleted by the current change.

LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the changes is indicated by a vertical line on the outer margins of the page. Changes to illustrations are indicated by miniature pointing hands.

Page No.	*Issue*
32-24	31 Aug 83
32-25 thru 32-26 Blank	5 Oct 84
32-26A	31 Aug 83
32-26B Blank	31 Aug 83
32-27	1 Oct 85
32-28	18 Nov 93
32-29 thru 32-30	31 Aug 83
32-30A	1 Oct 86
32-30B Blank	1 Oct 86
32-31	31 Aug 83
32-32	18 Nov 93
32-33 thru 32-34	1 Oct 86
32-35 thru 32-37	31 Aug 83
32-38 thru 32-40	5 Oct 84
32-40A thru 32-40B Blank	5 Oct 84
32-41	31 Aug 83
32-42	30 Oct 87
32-43	5 Oct 84
32-44	31 Aug 83
32-44A	16 Oct 89
32-44B	5 Oct 84
32-44C thru 32-44D	31 Aug 83
32-45	1 Dec 65
32-46	1 Oct 85
32-47	16 Oct 89
32-48 Blank	16 Oct 89
32-49	16 Oct 89
32-50 Blank	16 Oct 89
32-50A	16 Oct 89
32-50B Blank	16 Oct 89
32-50C	16 Oct 89
32-50D Blank	16 Oct 89
32-50E	16 Oct 89
32-50F Blank	16 Oct 89
32-50G	16 Oct 89
32-50H Blank	16 Oct 89
32-50J	16 Oct 89
32-50K Blank	16 Oct 89
32-51	19 Oct 90
32-52	16 Oct 89
32-52A Added	28 Oct 88
32-52B Blank Added	28 Oct 88
32-53	16 Oct 89
32-54	31 Aug 83
32-54A Added	31 Aug 83
32-54B Blank Added	31 Aug 83
32-55	Original
32-56	1 June 66
32-57	28 Oct 88
32-58 Blank	28 Oct 88
32-59	1 July 69
32-60	Original
32-61	1 Jan 67
32-62 Blank	1 Jan 67
32-63 thru 32-64	31 Aug 83
32-64A Added	1 Dec 65
32-64B Blank Added	1 Dec 65
32-65 thru 32-67	1 Dec 65
32-68 Blank	1 Dec 65
32-69	31 Aug 83
32-70 Blank	31 Aug 83
32-71	31 Aug 83
32-72 Blank	31 Aug 83
32-72A	31 Aug 83
32-72B Blank	31 Aug 83
32-73	28 Oct 88
32-74 thru 32-75	31 Aug 83
32-76	19 Oct 90
32-76A	28 Oct 88
32-76B Blank	28 Oct 88
32-77	31 Aug 83
32-78	1 June 67
32-79	1 June 67
32-80	1 Oct 85
32-80A Added	1 Oct 85
32-80B Blank Added	1 Oct 85
* 32-81	30 Dec 96
32-82	31 Aug 83
32-82A	28 Oct 88
32-82B Blank	28 Oct 88
32-82C Added	1 June 67
32-82D Blank Added	1 June 67
32-83	31 Aug 83
32-84 Blank	31 Aug 83
32-84A Added	1 July 69
32-84B Blank Added	1 July 69
32-85	1 July 69
32-86	16 Oct 89
32-87 thru 32-88	28 Oct 88
32-88A	28 Oct 88
32-88B Blank	28 Oct 88
32-89	28 Oct 88
32-90	5 Oct 84
32-90A thru 39-90B Blank	5 Oct 84
32-91	31 Aug 83
32-92	1 Oct 85
32-93	1 Oct 86
32-94 thru 32-96	Original
32-96A	16 Oct 89
32-96B	1 Oct 85
32-97	28 Oct 88
32-98 thru 32-99	16 Oct 89
32-100	31 Aug 83
33-1	Original
33-2 Blank	Original
33-3	1 Oct 85
33-4	31 Aug 83
33-5	1 July 69
33-6	31 Aug 83
33-6A	19 Oct 90
33-6B Blank	19 Oct 90
33-7	16 Oct 89
33-8	Original
33-9	5 Oct 84
33-10	1 June 66
33-10A Added	1 June 66
33-10B Blank Added	1 June 66
33-11	Original
33-12	31 Aug 83
33-12A Deleted	5 Oct 84
33-12B Blank Deleted	5 Oct 84
33-13	31 Aug 83
33-14	16 Oct 89
33-14A	28 Oct 88
33-14B thru 33-14C Added	31 Aug 83
33-14D	1 Oct 85
33-15	31 Aug 83
33-16	5 Oct 84
33-16A	31 Aug 83
33-16B Blank	31 Aug 83
33-17 thru 33-22	31 Aug 83
34-1	20 Dec 91
34-2 Blank	20 Dec 91
34-3 thru 34-4	31 Aug 83
34-4A Added	31 Aug 83

*The asterisk indicates pages changed, added, or deleted by the current change.

LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the changes is indicated by a vertical line on the outer margins of the page. Changes to illustrations are indicated by miniature pointing hands.

Page No.	Issue
34-4B Blank Added	31 Aug 83
34-5	1 Feb 65
34-6 Blank	1 Feb 65
34-7	Original
34-8	1 Feb 65
34-9	31 Aug 83
34-10 thru 34-12	1 Feb 65
34-13 thru 34-16	Original
34-17	1 Feb 65
34-18 Blank	1 Feb 65
34-19	1 Feb 65
34-20	1 Dec 65
34-21 thru 34-24	Original
* 34-25	30 Dec 95
* 34-26 Blank	30 Dec 95
34-27	1 Feb 65
34-28 Blank	1 Feb 65
34-29	Original
34-30 Blank	Original
34-31	Original
34-32 Blank	Original
34-33 thru 34-34	1 Feb 65
34-35 thru 34-37	Original
34-38 thru 34-40	1 Feb 65
34-41 thru 34-44	Original
34-45 thru 34-46	1 Feb 65
34-47 thru 34-49	Original
34-50 Blank	Original
34-51	Original
34-52	1 Oct 85
34-53	Original
34-54	31 Aug 83
34-55 thru 34-57	Original
34-58	1 Oct 85
34-59	1 July 69
34-60	Original
34-61 thru 34-62	1 Feb 65
34-63	1 Dec 65
34-64 thru 34-70	Original
34-71 thru 34-72	1 Feb 65
34-73	1 Oct 85
34-74 Blank	1 Oct 85
34-75 thru 34-76	Original
34-77	1 Feb 65
34-78 thru 34-79	Original
34-80 Blank	Original
34-81 thru 34-82	Original
34-83	31 Aug 83
34-84	1 June 67
34-84A Added	1 June 67
34-84B Blank Added	1 June 67
34-85	1 Dec 69
34-86 thru 34-87	1 June 67
34-88	31 Aug 83
34-88A thru 34-88B Added	1 June 67
34-88C	19 Oct 90
34-88D Blank	19 Oct 90
34-89 thru 34-90	Original
34-91 thru 34-92	31 Aug 83
34-93 thru 34-94	1 Oct 86
34-95	1 July 69
34-96	31 Aug 83
34-97	20 Dec 91
34-98 Blank	20 Dec 91
34-98A Added	20 Dec 91
34-98B Added	20 Dec 91
34-99	1 Dec 65
34-100 Added	1 Feb 65
34-101	1 July 69
34-102 thru 34-106 Added	1 Dec 65
35-1	31 Aug 83
35-2 thru 35-3	1 Dec 65
35-4	1 Oct 85
35-4A Added	1 Dec 65
35-4B Blank Added	1 Dec 65
35-5	Original
35-6	1 Oct 85
35-6A Added	31 Aug 83
35-6B Blank Added	31 Aug 83
35-7	31 Aug 83
35-8 thru 35-9	Original
35-10 thru 35-11	1 Dec 65
35-12	Original
35-13	1 June 67
35-14 Blank	1 June 67
35-15	1 June 67
35-16 Blank	1 June 67
38-1	Original
38-2	16 Oct 89
39-1	1 Apr 68
39-2 thru 39-3	Original
39-4 Blank	Original
39-5 thru 39-6	Original
39-7	1 Dec 65
39-8 thru 39-15	Original
39-16 Blank	Original
51-1	31 Aug 83
51-2 Blank	31 Aug 83
51-3	19 Oct 90
51-4 thru 51-5	30 Dec 94
51-6	1 Oct 85
51-7	Original
51-8	31 Aug 83
51-9	Original
51-10	1 Dec 65
51-11	Original
51-12	Original
51-13 thru 51-16	31 Aug 83
51-16A thru 51-16B Blank	5 Oct 84
51-17 thru 51-18	31 Aug 83
51-19	5 Oct 84
51-20 Blank	5 Oct 84
51-20A Deleted	31 Aug 83
51-20B Blank Deleted	31 Aug 83
51-21 Deleted	5 Oct 84
51-22 Blank Deleted	5 Oct 84
51-23 thru 51-25	31 Aug 83
51-26 Blank	31 Aug 83
51-27	31 Aug 83
51-28 Blank	31 Aug 83
51-29	Original

*The asterisk indicates pages changed, added, or deleted by the current change.

LIST OF EFFECTIVE PAGES

Note: The portion of the text affected by the changes is indicated by a vertical line on the outer margins of the page. Changes to illustrations are indicated by miniature pointing hands.

Page No.	*Issue*
51-30 thru 51-33.	31 Aug 83
51-34 Blank.	31 Aug 83
51-35.	31 Aug 83
51-36 thru 51-37.	Original
51-38 Blank	Original
51-39	31 Aug 83
51-40	1 Oct 85
51-40A	19 Oct 90
51-40B Blank	19 Oct 90
51-41	1 Oct 85
51-42 thru 51-43	28 Oct 88
51-44	18 Nov 93
51-44A thru 51-44B	31 Aug 83
51-45	31 Aug 83
51-46 Blank	31 Aug 83
51-47	28 Oct 88
51-48	5 Oct 84
51-49	31 Aug 83
51-50	1 Oct 85
51-50-1 thru 51-50-2	28 Oct 88
51-50A thru 51-50B	19 Oct 90
51-51 thru 51-52	31 Aug 83
51-53.	5 Oct 84
51-54 thru 51-56.	31 Aug 83
51-56A thru 51-56B Added.	31 Aug 83
* 51-57	15 Dec 98
* 51-58 Blank	15 Dec 98
51-59 thru 51-60.	31 Aug 83
51-61 thru 51-92 Deleted.	1 Apr 68
52-1	1 June 67
52-2 Blank	1 June 67
52-3	31 Aug 83
52-4	1 Dec 65
52-4A	1 June 66
52-4B	1 June 66
52-4C Added	1 June 66
52-4D Blank Added	1 June 66
52-5 thru 52-7.	Original
52-8	31 Aug 83
52-9 thru 52-10.	Original
52-11.	31 Aug 83
52-12.	Original
52-13.	31 Aug 83
52-14 thru 52-15.	1 Dec 65
52-16 thru 52-17.	Original
52-18	1 Dec 65
52-19 thru 52-22	Original
52-23	1 Oct 85
52-24	Original
52-25	1 Oct 85
52-26 thru 52-30	Original
52-31	1 June 66
52-32 Blank	1 June 66
52-33 thru 52-34	1 June 67
52-35	1 Oct 85
52-36 thru 52-43	1 June 67
52-44 Blank	1 June 67
52-45 Added	1 June 67
52-46 Blank Added	1 June 67
52-47 Added	1 June 67
52-48 Blank Added	1 June 67
52-49 thru 52-54 Added	1 June 67
53-1	1 Oct 85
53-2 thru 53-8	Original
53-9	31 Aug 83
53-10 thru 53-11	16 Oct 89
53-12 - 53-17	Original
53-18	1 Apr 68
53-19	31 Aug 83
53-20 thru 53-23	Original
53-24 Blank	Original
54-1	1 July 69
54-2	1 Oct 85
54-3 thru 54-4	Original
54-5 thru 54-7	31 Aug 83
54-8	Original
54-9	1 Feb 65
54-10	1 July 69
55-1	Original
55-2	1 Oct 85
55-3	Original
55-4	1 Oct 86
55-5 thru 55-10	Original
55-11	31 Aug 83
55-12 Blank	31 Aug 83
56-1	30 Oct 87
56-2	31 Aug 83
56-3 thru 56-4	18 Nov 93
56-5 thru 56-6	30 Dec 96
56-7	31 Aug 83
56-8	Original
56-9	19 Oct 90
56-10	1 Oct 85
56-11	31 Aug 83
56-12	1 Feb 65
56-13	31 Aug 83
56-14	16 Oct 89
* 56-15	15 Dec 98
56-16	31 Aug 83
56-17	30 Oct 87
56-18	19 Oct 90
57-1 thru 57-2	31 Aug 83
57-2A	1 Apr 68
57-2B Blank	1 Apr 68
57-3	31 Aug 83
57-4 Blank	31 Aug 83
57-5 thru 57-11	Original
57-12.	31 Aug 83
57-13.	Original
57-14.	31 Aug 83
57-15 thru 57-23.	Original
57-24.	5 Oct 84
57-25.	31 Aug 83
57-26 Blank.	31 Aug 83
57-27 thru 57-28 Deleted.	31 Aug 83
58-1	31 Aug 83
58-2 Blank	31 Aug 83
58-3 thru 58-16.	31 Aug 83
58-17 thru 58-20	5 Oct 84
58-21.	31 Aug 83
58-22 Blank.	31 Aug 83
58-23 thru 58-24	31 Aug 83
58-25	16 Oct 89
58-26 thru 58-34	31 Aug 83

*The asterisk indicates pages changed, added, or deleted by the current change.

Use and/or disclosure is governed by the statement on the title page of this document.

INSERT LATEST CHANGED PAGES. DESTROY SUPERSEDED PAGES.

LIST OF EFFECTIVE PAGES

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Page No.	*Issue*
58-35	1 Oct 85
58-36 thru 58-49	31 Aug 83
58-50	16 Oct 89
58-51 thru 58-52	31 Aug 83
58-53 thru 58-54	5 Oct 84
58-55 thru 58-57	31 Aug 83
58-58 thru 58-60	18 Nov 93
58-60	31 Aug 83
58-61	16 Oct 89
58-62 thru 58-71	31 Aug 83
58-72	5 Oct 84
58-73	31 Aug 83
58-74 thru 58-75	18 Nov 93
58-76	31 Aug 83
58-77	18 Nov 93
58-78 thru 58-82	31 Aug 83
58-83	16 Oct 89
58-84	1 Oct 85
58-85 thru 58-94	31 Aug 83
58-94A thru 58-94H Added	5 Oct 84
58-94J thru 58-94N Added	5 Oct 84
58-94P thru 58-94T Added	5 Oct 84
58-94U	16 Oct 89
58-94V thru 58-94Z Added	5 Oct 84
58-94AA thru 58-94AC	5 Oct 84
58-94AD	28 Oct 88
58-94AE Added	5 Oct 84
58-94AF Blank Added	5 Oct 84
58-95	20 Dec 91
58-96	16 Oct 89
58-97 thru 58-126	31 Aug 83
58-127 thru 58-128	5 Oct 84
58-128A	5 Oct 84
58-128 B thru 58-128C	18 Nov 93
58-128D Added	5 Oct 84
58-128E	30 Dec 94
58-128F	18 Nov 93
58-128G thru 58-128H Added	20 Dec 91
58-129	5 Oct 84
58-130 thru 58-131	18 Nov 93
58-132 thru 58-133	31 Aug 83
58-134	30 Dec 96
58-135 thru 58-136	31 Aug 83
58-137	30 Dec 94
58-138 thru 58-141	31 Aug 83
58-142	1 Oct 85
58-143 thru 58-154	31 Aug 83
58-155	28 Oct 88
58-156 thru 58-158	31 Aug 83
58-159 thru 58-160	5 Oct 84
58-160A thru 58-160H Added	5 Oct 84
58-160J thru 58-160N Added	5 Oct 84
58-160P thru 58-160T Added	5 Oct 84
58-160U	16 Oct 89
58-160V thru 58-160Z Added	5 Oct 84
58-160AA Added	5 Oct 84
58-160AB Blank Added	5 Oct 84
58-161	5 Oct 84
58-162 thru 58-164	18 Nov 93
58-165 thru 58-168	31 Aug 83
58-169	16 Oct 89
58-170 thru 58-175	31 Aug 83
58-176	30 Dec 94
58-177 thru 58-178	18 Nov 93
58-179	31 Aug 83
58-180 Blank	31 Aug 83
58-181 thru 58-182	5 Oct 84
58-183 thru 58-184	31 Aug 83
58-185 thru 58-188	18 Nov 93
59-1	31 Aug 83
59-2 Blank	31 Aug 83
59-3	31 Aug 83
59-4 Blank	31 Aug 83
59-5	1 Oct 85
59-6	1 June 67
59-7	1 Oct 85
59-8	30 Oct 87
59-9	30 Oct 87
59-10	1 Oct 86
59-10A thru 59-10B Blank	5 Oct 84
59-11	5 Oct 84
59-12	1 Apr 68
59-12A Added	1 Apr 68
59-12B Blank Added	1 Apr 68
59-13	19 Oct 90
59-14 thru 59-17	31 Aug 83
59-18	19 Oct 90
59-19	18 Nov 93
59-20	1 June 67
*59-21	31 May 05
*59-22 Blank	31 May 05
59-23 thru 59-24	5 Oct 84
59-24A thru 59-24B Added	5 Oct 84
59-25 thru 59-26	5 Oct 84
59-27 thru 59-34 Added	1 June 67
59-35	5 Oct 84
59-36	1 June 67
59-37	5 Oct 84
59-38	1 Oct 85
59-39	31 Aug 83
59-40	1 Oct 85
59-4 thru 59-45	31 Aug 83
59-46 thru 59-47	19 Oct 90
59-48 thru 59-52	31 Aug 83
59-53	16 Oct 89
59-54 thru 59-55	31 Aug 83
59-56	1 June 67
59-57	30 Oct 87
59-58 Blank	30 Oct 87
59-59 thru 59-60	28 Oct 88
59-61	31 Aug 83
59-62 Added	1 June 67
59-62A Added	31 Aug 83
59-62B Blank Added	31 Aug 83
59-63	18 Nov 93
59-64 Blank	18 Nov 93
59-65	18 Nov 93
59-66 Blank	18 Nov 93
59-66A thru 59-66E	30 Dec 94
59-66F Blank	30 Dec 94
59-67 Deleted	31 Aug 83
59-68 Deleted	31 Aug 83
59-69 Deleted	31 Aug 83
59-70 Deleted	31 Aug 83
59-70A thru 59-70F	31 Aug 83
59-70G thru 59-70H Added	31 Aug 83
59-70J thru 59-70M Added	31 Aug 83
59-70N	19 Oct 90
59-70P	5 Oct 84
59-70Q	30 Oct 87
59-70R Added	31 Aug 83
59-71	31 Aug 83
59-72 Blank	31 Aug 83
59-73	1 Oct 85

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INSERT LATEST CHANGED PAGES. DESTROY SUPERSEDED PAGES.

LIST OF EFFECTIVE PAGES

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Page No.	*Issue*
59-74	31 Aug 83
59-75	1 Oct 85
59-76 thru 59-77	31 Aug 83
59-78	5 Oct 84
59-79 thru 59-80	28 Oct 88
59-80A Added	28 Oct 88
59-80B Blank Added	28 Oct 88
59-81	5Oct 84
59-82 thru 59-85	1 June 67
59-86	5 Oct 84
59-87	19 Oct 90
59-88 thru 59-91	31 Aug 83
59-92 thru 59-93	1 Apr 68
59-94	31 Aug 83
59-95 Deleted	31 Aug 83
59-96 Deleted	31 Aug 83
59-97	31 Aug 83
59-98 Blank	31 Aug 83
59-99	1 Dec 69
59-100 Blank	1 Dec 69
59-101 thru 59-102	1 Dec 69
59-103 thru 59-104	1 Apr 68
59-105 thru 59-106	31 Aug 83
59-106A Deleted	31 Aug 83
59-106B Blank Deleted	31 Aug 83
59-107	31 Aug 83
59-108	19 Oct 90
59-109 thru 59-117	31 Aug 83
59-118	16 Oct 89
59-119 thru 59-136	31 Aug 83
59-136A thru 59-136E Deleted	31 Aug 83
59-136F Blank Deleted	31 Aug 83
59-137 thru 59-145	31 Aug 83
59-146	1 Oct 85
59-150	20 Dec 91
59-151	31 Aug 83
59-152	28 Oct 88
59-153	31 Aug 83
59-154	5 Oct 84
59-155	31 Aug 83
59-156	19 Oct 90
59-157	31 Aug 83
59-158	16 Oct 89
59-159 thru 59-181	31 Aug 83
59-182 Blank	31 Aug 83
59-183 thru 59-21	31 Aug 83
59-211 thru 59-243 Deleted	31 Aug 83
59-244 Blank Deleted	31 Aug 83
71-1 thru 71-2	31 Aug 83
71-3	1 July 69
71-4	1 June 67
71-5	31 Aug 83
71-6 Blank	31 Aug 83
71-6A Added	31 Aug 83
71-6B Blank Added	31 Aug 83
71-7	1 Dec 65
71-8	Original
71-9 thru 71-108	31 Aug 83
71-11 thru 71-12 Blank	5 Oct 84
71-13	1 July 69
71-14 Blank	1 July 69
71-15	16 Oct 89
71-16 Added	1 June 67
72-1	1 June 67
72-2	1 June 67
72-2A Deleted	1 June 67
72-2B Blank Deleted	1 June 67
72-3	1 June 67
72-4	16 Oct 89
72-4A Added	1 June 67
72-4B Blank Added	1 June 67
72-5	1 June 67
72-6	1 June 67
72-7	1 Dec 65
72-8	1 Feb 65
72-9 thru 72-10	31 Aug 83
72-11	1 Apr 68
72-12	31 Aug 83
72-12A Deleted	1 June 66
72-12B Blank Deleted	1 June 66
72-13	1 June 67
72-14	1 June 67
72-14A	31 Aug 83
72-14B Blank	31 Aug 83
72-15	1 Oct 85
72-16	1 Dec 65
72-17	Original
72-18	Original
72-18A	31 Aug 83
72-18B Blank	31 Aug 83
72-19 thru 72-20	31 Aug 83
72-21	Original
72-22	1 June 66
72-23	31 Aug 83
72-24 Blank	31 Aug 83
72-25	Original
72-26 Blank	Original
72-27	1 June 67
72-28	1 June 67
72-29 Added	1 June 67
72-30 Blank Added	1 June 67
73-1	31 Aug 83
73-2	31 Aug 83
73-3	Original
73-4	31 Aug 83
73-4A Added	31 Aug 83
73-4B Blank Added	31 Aug 83
73-5	1 Dec 69
73-6 Blank	1 Dec 69
74-1	Original
74-2 Blank	Original
75-1	Original
75-2 Blank	Original
76-1	31 Aug 83
76-2	1 June 67

*The asterisk indicates pages changed or deleted by the current change.

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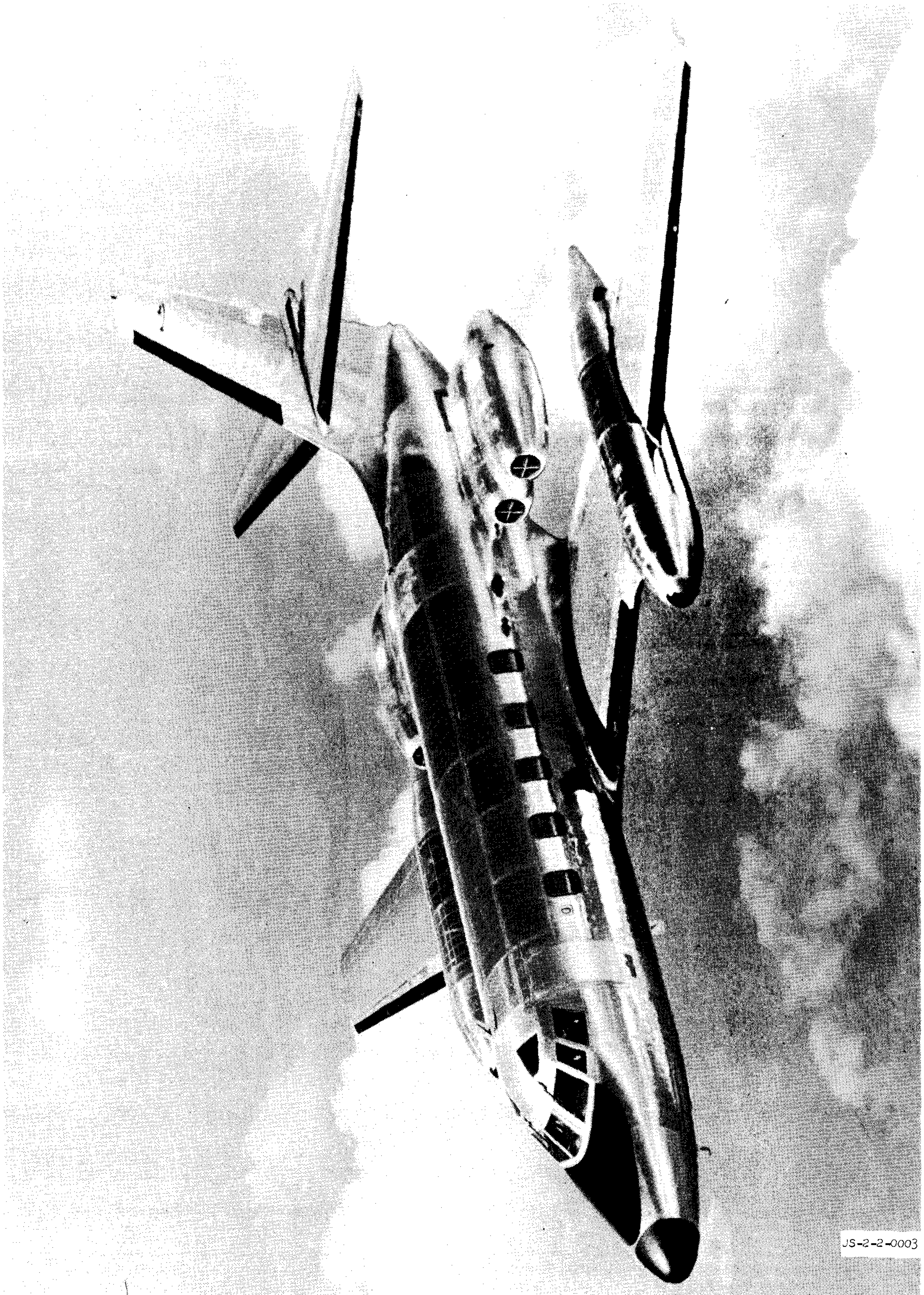


The Airplane

GENERAL AIRPLANE INFORMATION

GENERAL.

The JetStar is a high-performance, swept-wing, jet propelled monoplane. The airplane is a medium-range transport capable of operating from small and medium-sized airports. It is of all-metal construction, and is powered by four Pratt and Whitney JT12A turbojet engines, mounted well aft on the fuselage, two on each side. Engine thrust reversers are available to supplement the power brakes for exceptionally rapid deceleration power. Either an operable drag chute system or provisions for one are installed.

FUSELAGE.

The fuselage has a cylindrical cross section, and is divided into a nose electronic equipment compartment, pilots' compartment, passengers' compartment, and an aft fuselage equipment compartment. The pilots' and passengers' compartments can be pressurized to permit high cruise altitudes without discomfort. A hydraulically operated speed brake in the lower aft surface of the fuselage may be extended during flight to reduce speed. Normal entrance to the fuselage is through a cabin door on the left side. The door has a key-operated lock. Two escape windows, one on each side of the passengers' compartment, provide emergency exits and entrances.

EMPENNAGE.

The JetStar empennage is of conventional two-surface design. Normal pitch and yaw control is provided through conventional elevators and rudder. The elevators are hydraulically boosted to provide control over the entire operating range with minimum pilot effort. The rudder is aerodynamically boosted by a balance tab. Pitch trim is accomplished by movement of the empennage assembly, which is pivoted about the lower end of the vertical stabilizer rear spar by a dual, electrically driven actuator. The empennage leading edges are provided with flexible de-icing boots.

WINGS.

The wings have both leading and trailing edge flaps plus conventional ailerons for flight control. The ailerons are hydraulically boosted and are provided with electrically operated trim tabs. Flexible de-icing boots are attached to the leading edges. Each wing half incorporates two integral tanks and one permanent wing-mounted external tank.

LANDING GEAR.

The landing gear is of conventional tricycle arrangement with dual nose and main wheels. Hydraulic power is used for normal retraction and extension. Emergency extension is by free fall and compressed air bottles. The nose gear is hydraulically steerable, and retracts forward to a wheel well below the pilots' compartment. The main gears incorporate hydraulic anti-skid wheel braking, and retract inboard to wheel wells in the center fuselage.

TABLE OF CONTENTS

GENERAL AIRPLANE INFORMATION

INTRODUCTION

GROUP AIRCRAFT GENERAL

CHAPTER 5 TIME LIMITS/MAINTENANCE CHECKS
- Section 5.0 General
- Section 5.1 Pre-Flight Inspection
- Section 5.2 Post-Flight Inspection
- Section 5.3 100/300-Hour Inspection
- Section 5.4 Annual Inspection
- Section 5.5 Special Inspections
- Section 5.6 Overhaul and Replacement Schedule
- Section 5.7 Continued Airworthiness

CHAPTER 6 DIMENSIONS AND AREAS

CHAPTER 7 LIFTING AND SHORING
- Section 7.0 Lifting
- Section 7.1 Shoring
- Section 7.2 Methods of Handling a Damaged Airplane

CHAPTER 8 LEVELING AND WEIGHING
- Section 8.0 Leveling
- Section 8.1 Weighing

CHAPTER 9 TOWING AND TAXIING
- Section 9.0 Towing
- Section 9.1 Taxiing

CHAPTER 10 PARKING AND MOORING
- Section 10.0 Parking
- Section 10.1 Mooring
- Section 10.2 Storage During Periods of Inactivity

CHAPTER 12 SERVICING
- Section 12.0 General
- Section 12.1 Servicing the Fuel System
- Section 12.2 Servicing the Engine Oil System
- Section 12.3 Servicing the Engine Fire Extinguisher System
- Section 12.4 Servicing the Hydraulic System
- Section 12.5 Servicing the Batteries
- Section 12.6 Servicing the Landing Gear
- Section 12.7 Servicing the Pitot-Static System
- Section 12.8 Servicing the Environmental Systems
- Section 12.9 Servicing the Pitch Trim Actuator
- Section 12.10 Servicing the Oxygen System
- Section 12.11 Servicing the Lavatory Water System
- Section 12.12 Lubrication
- Section 12.13 Deleted
- Section 12.14 Exterior Cleaning

TABLE OF CONTENTS (Cont)

GROUP AIRFRAME SYSTEMS

CHAPTER 21 AIR CONDITIONING
Section 21.0 Air Conditioning System
Section 21.1 Pressurization System
Section 21.2 Electronic Equipment Cooling System
Section 21.3 Toilet Ventilation System

CHAPTER 22 AUTOPILOT
Section 22. 0 SP40 Sperry Autopilot
Section 22. 1 AP104 Collins Autopilot

CHAPTER 23 COMMUNICATIONS
Section 23. 0 General
Section 23. 1 Intercommunication and Passenger Address System
Section 23. 2 VHF Communication Systems
Section 23. 3 HF Communication Systems

CHAPTER 24 ELECTRICAL POWER
Section 24. 0 DC Power System
Section 24. 1 AC Power System (Airplanes 5001, 5002, 5004 through 5009)
Section 24. 2 AC Power System (Airplanes 5003, 5005, 5010 and Up)
Section 24.3 Recommended Wiring Practices

CHAPTER 25 EQUIPMENT AND FURNISHINGS
Section 25.0 Pilot's and Copilot's Seats
Section 25.1 Safety Belts and Inertia Reel
Section 25.2 Hand Fire Extinguishers

CHAPTER 26 FIRE PROTECTION

CHAPTER 27 FLIGHT CONTROLS
Section 27.0 Aileron Control System
Section 27.1 Elevator Control System
Section 27.2 Rudder Control System
Section 27.3 Trim Control and Mach Trim System
Section 27.4 Leading and Trailing Edge Wing Flap Systems
Section 27.5 Speed Brake System
Section 27.6 Stall Warning System (Dash 6 Jetstar)
Section 27.7 Stall Warning System (Dash 8 Jetstar)
Section 27.8 Control Cables - General Requirements

CHAPTER 28 FUEL
Section 28.0 Fuel Supply System
Section 28.1 Fuel Vent System
Section 28.2 Fuel Tanks
Section 28.3 Fuel Quantity Indicating System
Section 28.4 Defueling and Jettisoning Systems
Section 28.5 Fuel System Testing

CHAPTER 29 HYDRAULIC POWER
Section 29.0 Hydraulic Power General
Section 29.1 Normal (No. 1) and Auxiliary (No. 3) Hydraulic Systems
Section 29.2 Standby (No. 2) Hydraulic System

TABLE OF CONTENTS (Cont)

GROUP AIRFRAME SYSTEMS (Cont)

CHAPTER 30 ICE AND RAIN PROTECTION
- Section 30.0 General
- Section 30.1 Wing and Empennage De-Icing System
- Section 30.2 Windshield Anti-Icing System
- Section 30.3 Windshield Wiper System
- Section 30.4 Pitot Tube and Static Port Anti-Icing System
- Section 30.5 Windshield De-Fog System
- Section 30.6 Cabin Window De-Fog System
- Section 30.7 Engine & Nacelle Ice Protection

CHAPTER 31 INSTRUMENTS

CHAPTER 32 LANDING GEAR
- Section 32.0 Landing Gear and Landing Gear Doors
- Section 32.1 Emergency Gear Release System (Airplanes 5001 through 5085)
- Section 32.2 Landing Gear Indicating and Warning System
- Section 32.3 Brake and Anti-Skid Systems
- Section 32.4 Nose Steering System
- Section 32.5 Emergency Gear Release System (Airplanes 5086 and up)

CHAPTER 33 LIGHTING
- Section 33.0 Aircraft Lighting
- Section 33.1 Warning Circuits

CHAPTER 34 NAVIGATION SYSTEMS
- Section 34.0 General
- Section 34.1 VHP Navigation Systems (VOR)
- Section 34.2 Flight Director Systems
- Section 34.3 Glideslope Systems
- Section 34.4 Marker Beacon System
- Section 34.5 Automatic Direction Finder
- Section 34.6 Gyrosyn Compass Systems
- Section 34.7 Weather Radar
- Section 34.8 Distance Measuring Equipment System
- Section 34.9 Air Traffic Control Transponder System
- Section 34.10 Air Data, Attitude, and Direction Instrumentation (Pitot-Static System)
- Section 34.11 Alternate Electronic Systems
- Section 34.12 Doppler Navigation and Computer System

CHAPTER 35 OXYGEN SYSTEM

CHAPTER 38 WATER/WASTE

CHAPTER 39 DRAG CHUTE DELETED

GROUP STRUCTURES

CHAPTER 51 STRUCTURES
- Section 51.0 General
- Section 51.1 Alignment and Symmetry Check
- Section 51.2 Airframe Sealing

TABLE OF CONTENTS (Cont)

GROUP STRUCTURES (Cont)

Section 51.3 Penetrant Inspection
Section 51.4 Pressurization Check
Section 51.5 Mass Rebalance of Control Surfaces
Section 51.6 Chemical Identification of Materials
Section 51.7 Corrosion Control
Section 51.8 Finishes
Section 51.9 Depleted Uranium Counterweights
Section 51.10 Airframe Long Term Storage

CHAPTER 52 DOORS
Section 52.0 Entrance Door
Section 52.1 Emergency Escape Hatch
Section 52.2 Nose Cone
Section 52.3 Access and Inspection Provisions
Section 52.4 Entrance Stairs
Section 52.5 Landing Gear Doors

CHAPTER 53 FUSELAGE

CHAPTER 54 NACELLES AND PYLONS

CHAPTER 55 STABILIZERS

CHAPTER 56 WINDOWS
Description and Operation
Inspection
Removal and Replacement

CHAPTER 57 WINGS

CHAPTER 58 NON DESTRUCTIVE EVALUATION
Section 58.0 General
Section 58.1 Nondestructive Evaluation Methods
Section 58.2 Wing Group
Section 58.3 Empennage Group
Section 58.4 Fuselage Group
Section 58.5 Landing Gear Group
Section 58.6 Nacelle Group

CHAPTER 59 STRUCTURAL REPAIR INSTRUCTIONS
Section 59.0 Introduction
Section 59.1 General
Section 59.2 Wing Group
Section 59.3 Empennage Group
Section 59.4 Fuselage Group
Section 59.5 Landing Gear Group
Section 59.6 Nacelles
Section 59.7 Damage due to Landing Gear Failure
Section 59.8 Typical Repairs and Applications
Section 59.9 10,000 Hour Inspection

TABLE OF CONTENTS (Cont)

GROUP POWER PLANT

CHAPTER 71 POWER PLANT
CHAPTER 72 ENGINE
CHAPTER 73 ENGINE FUEL AND CONTROL
CHAPTER 74 IGNITION
CHAPTER 75 AIR
CHAPTER 76 ENGINE CONTROLS
CHAPTER 77 ENGINE INDICATING
Section 77.0 Engine Pressure Ratio
Section 77.1 Tachometer
Section 77.2 Exhaust Gas Temperature
Section 77.3 Fuel Flow
Section 77.4 Oil Pressure
Section 77.5 Oil Temperature
CHAPTER 78 EXHAUST
CHAPTER 80 STARTING

INTRODUCTION

GENERAL

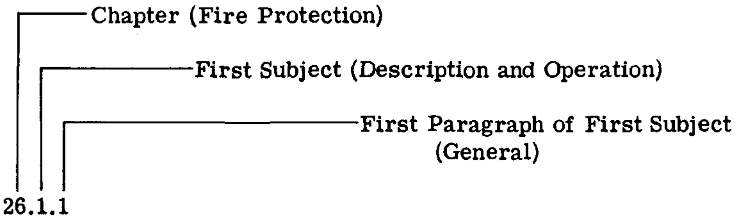
This Handbook of Operating and Maintenance Instructions contains information of the systems and components of the JetStar airplane and instructions for the maintenance of the airplane in an airworthy condition. While the maximum of useful maintenance information has been included in this handbook, detailed instructions for standard aircraft maintenance procedures or for use of common hand tools have been omitted or referenced. Detailed instructions for working on all airplane systems and components are provided. Repair shop operations for electronic packages, engines, and similar items, after removal from the airplane, are omitted, but a list of reference publications is included to assist the maintenance man in locating this type of information.

FORMAT

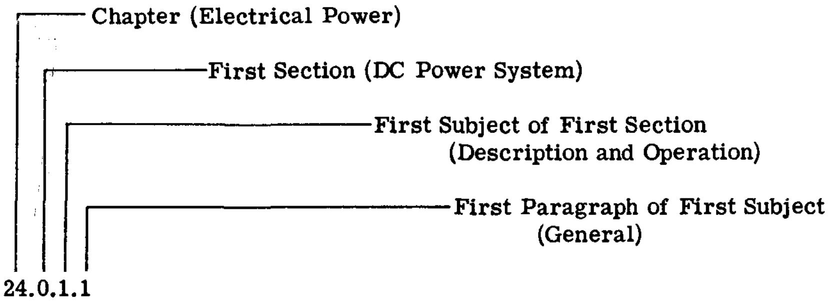
The handbook is divided into chapters numbered in accordance with A.T.A. Specification 100. The chapter breakdown is functional or by areas of the airplane, as applicable.

IDENTIFICATION OF SUBJECT MATTER

Subject matter is identified by a simple numbering system which can occur in two forms. Some chapters are not subdivided into sections and are broken down:


Chapter (Fire Protection)
First Subject (Description and Operation)
First Paragraph of First Subject (General)
26.1.1

Other chapters, due to the amount and diversity of the subject matter which they contain, are subdivided into sections:


Chapter (Electrical Power)
First Section (DC Power System)
First Subject of First Section (Description and Operation)
First Paragraph of First Subject (General)
24.0.1.1

Note that the first section of a chapter divided into sections is identified by ".0" following the chapter number, hence ".1" identifies the second section of the chapter, ".2" the third section, and so on.

INDEXING

A table of contents prefaces each chapter, listing the contents in the order in which they appear. In those chapters which are divided into sections, the table of contents lists only the sections, and a section table of contents prefaces each section. The divisions used within a chapter which is not sectionalized and those used within a section are similar:

DESCRIPTION AND OPERATION
OPERATIONAL CHECKOUT
TROUBLESHOOTING
REMOVAL AND REPLACEMENT
PRESERVATION
EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION

In some cases, the subject matter of the chapter or section does not require use of all these major subjects, and only those needed are used.

PAGE NUMBERING

The pages in each chapter are identified as dash numbers of the chapter number. The dash numbers run from "-1" (for the first page of the chapter) straight through to the highest page number in the chapter, without regard for the subdivision of some chapters into sections.

Chapter (Fuel)
Fifth Page in Chapter
28-5

FIGURE IDENTIFICATION AND LOCATION

The figures are identified by dash numbers of the chapter number in which they appear:

Chapter (Oxygen)
(Second Figure in Chapter)
35-2

Figure numbers run consecutively from -1 to the highest figure number in each chapter. In chapters not divided into sections, all figures are located at the end of the chapter. In chapters which are divided into sections, the figures pertaining to each section are located at the end of the section. In either case, the figure numbering within the chapter is consecutive, and does not identify the section number, where sections are used. The figure numbers, titles, and the pages on which they appear, are listed at the end of the chapter table of contents when the chapter is not sectionalized, or at the end of the section table of contents when the chapter is sectionalized.

EMPHASIS TERMS

In cases where it is necessary to emphasize an instruction or statement, the following methods are used:

"Note." An operating procedure, condition, etc., which it is essential to emphasize. Example:

Note

The seal is installed with the holes facing the pressurized area (inboard).

"Caution." Operating procedures, practices, etc., which if not strictly observed will result in damage to equipment. Example:

CAUTION

Do not use a screwdriver or similar tool to try and pry the pane loose. This can damage the pane.

"Warning." Operating procedures, practices, etc., which will result in personal injury or loss of life if not carefully followed. Example:

WARNING

Do not strike or bump the fuselage while it is under pressure.

SCHEMATIC DIAGRAMS

The schematic diagrams used in this manual contain symbols representing common mechanical and electrical items, such as valves, relays, etc. These symbols are shown in figures i-1 and i-2. Areas of the airplane in which items are located, such as instrument panels, junction boxes, etc., are defined by broken lines surrounding the items in the specific area, with the applicable nomenclature to identify the area. The arrangement of the schematic diagrams is not necessarily the same as the system arrangement on the airplane, but is devised for the best presentation of component sequence and power flow within the system.

SWITCH AND CONTROL NOMENCLATURES AND POSITIONS

The text of this handbook presents switch and control nomenclatures and positions as they appear on the airplane. The nomenclature is printed in capital letters; positions are printed in capital letters and enclosed in quotation marks. Where there is no marking on the airplane, this does not apply, and lower case letters are used.

MODELS COVERED

This handbook covers the components and systems of JetStar Models 1329-23A, 1329-23D, and 1329-23E. Supplements to this handbook are issued as necessary to give line maintenance instructions for components and systems which differ from those of Jet Star Models 1329-23A, 1329-23D, and 1329-23E. The appropriate supplement should be used in conjunction with this handbook to obtain complete line maintenance instructions for any JetStar models other than these.

REVISIONS

Revisions to this handbook are issued to cover changes in the JetStar Models 1329-23A, 1329-23D, and 1329-23E, and improved maintenance procedures. Each revision includes a list of effective pages. Check list to be sure your copy is up to date. Pages added by revision which fall within a chapter are identified as lettered suffixes of the original page number preceding them, and should be inserted to follow the existing page of the same basic number. Pages added by revision which fall at the end of a chapter are identified as continuing numbers of the existing page numbers, and should be inserted to follow the existing pages.

STANDARD TORQUE VALUES

Bolt and screw torque values are specified in Section 51.0 of this manual.
Torque values for flareless tubing are specified in Section 29.0 of this manual.
Torque values for wire terminal installation are specified in Section 24.3 of this manual.
The flexible Wiggins type fuel line coupling torque values are specified in Section 28.0 of this manual.
The torque values for oxygen tubing are specified in Chapter 35 of this manual.
Flared tube fitting torque values are specified in figure 28-4A in this manual. These include the rigid NAS connectors used primarily in the fire extinguishing system.
These torque values should not be construed to supersede any special torque requirements that may be specified elsewhere in this manual.

TECHNICAL INFORMATION

All technical inquiries and other general information questions should be directed to:

Lockheed Martin Aircraft & Logistics Centers
107 Frederick Street B270
Greenville, SC 29605

Technical support is available Telephone: (864) 422-2179 Hot Line: (864) 422-6265
Fax: (864) 422-2184

Services: Troubleshooting assistance and engineering,
OMR, Service Bulletin update, HOMI revision,
JetStar program matters.

The above phone hot line number is also good for emergency after-hours technical support.

Order Technical Manuals and HOMI revisions:
Telephone: (864) 422-2171 Fax: (864) 422-2183

SPARE PARTS/MATERIALS

All inquires concerning parts availability and/or pricing should be directed to:

Hi-Tech Aero Spares
3436 Meridian Way
West Chester, OH. 45069

Spares Support is available during normal business hours (8:00am to 5:00pm Eastern Time) and AOG at:
Telephone: (513) 942-4541, Ext. 120

REFERENCE PUBLICATIONS

In addition to this handbook, other Lockheed Martin Aircraft & Logistics Center publications concerning the JetStar airplane are:

FAA Approved Airplane Flight Manual for Lockheed Model 1329 JetStar (Publication No. SMR-150)
JetStar Illustrated Parts Breakdown

JetStar Weight and Balance Data
JetStar Electrical and Electronic Wiring Diagram
JetStar Electrical Load Analysis

The following list includes those equipment manufacturers' publications referenced in this handbook, and others which may be helpful in maintaining the JetStar airplane and its systems and equipment:

ADAMS RITE SABRE INTERNATIONAL, FORMERLY SABRE INDUSTRIES, WOODSTOCK, CT 06281

Oxygen components

AEROQUIP CORPORATION, JACKSON, MI 49203

Description of Hose of Teflon; Catalog No. 104

AIRCRAFT POROUS MEDIA INC., PINELLAS PARK, FL 34665-5798

Overhaul Manual APM-TM-71-6, Hydraulic Filter

AIRCRAFT TECHNICAL PUBLISHERS (ATP), BRISBANE, CA 94005-1203

JetStar technical manuals on microfiche are available

ALCO CONTROLS DIVISION, EMERSON ELECTRIC CO., ST. LOUIS, MO 63141

Overhaul Manual Breakdown for Electric Windshield Wiper System, XW 20101-707

AUTO-VALVE, INCORPORATED, DAYTON, OH 45427

Blueprints of Condensate Drain Valves, Part No. 750B-1, 750B-2, and 750B-3

BARRY WRIGHT CORP., BURBANK, CA 91505

Overhaul Manual TK1846-7, JetStar Engine Mount Assemblies

BENDIX, INSTRUMENTS AND LIFE SUPPORT DIVISION; DAVENPORT, IA 52808

Service Bulletin 7631, Capacitance Test Set Models TF-20, TF-20-1 and TF-20-3 Modification

BENDIX RADIO, BALTIMORE, MD 21204

Description, Bendix Radar Antenna (ANT-1G); Publication No. IB-735
Description, Radar Control Panel (CON-1P); Publication No. IB-734
Description, Bendix Weather Scope (PPI-IE); Publication No. IB-740
Instruction Book, Synchronizer-Amplifier-Power Supply; Publication No. IB-739
Instruction Book, Weather Radar System; Publication No. IB-732B
Instruction Book, ANT-1(G) Antenna; Publication No. IB-735-1
Instruction Book, Bright Display Indicator; Publication No. IB-740-1
Instruction Book, RDR-1D (Weather) Radar System; Publication No. IB-765

BERTEA PRODUCTS, PASADENA, CA 92664

Overhaul Instructions with Parts List, Nose Landing Gear Steering Valve, Part No. 50100

CARLETON TECHNOLOGIES INC., ORCHARD PARK, NY 14127-4195

Oxygen components, formerly the Aro corp.

COLLINS RADIO COMPANY, CEDAR RAPIDS, IA 52406

Installation Book, 17L-7 VHF Communications Transmitter, Publication No. 523-0021-006
Maintenance Book, 17L-7 VHF Communications Transmitter, Publication No. 523-0021-004
Overhaul Book, 17L-7 VHF Communications Transmitter, Publication No. 523-0021-003
Illustrated Parts Catalog, 17L-7 VHF Communications Transmitter, Publication No. 523-0021-005

Use and/or disclosure is governed by the statement on the title page of this document.

Installation Book, 51X-2 VHF Communications Receiver, Publication No. 523-0022-006
Maintenance Book, 51X-2 VHF Communications Receiver, Publication No. 523-0022-004
Overhaul Book, 51X-2 VHF Communications Receiver, Publication No. 523-0022-003
Illustrated Parts Catalog, 51X-2 VHF Communications Receiver, Publication No. 523-0022-005
Instruction Book, 344B-1/1A VOR Instrumentation, Publication No. 520-5598-000
Installation Book, 51RV-1 VOR/ILS Receiver, Publication No. 523-0736-006
Maintenance Book, 51RV-1 VOR/ILS Receiver, Publication No. 523-0736-004
Overhaul Book, 51RV-1 VOR/ILS Receiver, Publication No. 523-0736-003
Illustrated Parts Catalog, 51RV-1 VOR/ILS Receiver, Publication No. 523-0736-005
Instruction Book, 51Z-2/2A Marker Beacon Receiver, Publication No. 520-5543-006
Installation Book, 51Z-3 Marker Beacon Receiver, Publication No. 523-0092-006
Maintenance Book, 51Z-3 Marker Beacon Receiver, Publication No. 523-0092-004
Overhaul Book, 51Z-3 Marker Beacon Receiver, Publication No. 523-0092-003
Illustrated Parts Catalog, 51Z-3 Marker Beacon Receiver, Publication No. 523-0092-005
Installation Book, DF-202 ADF System, Publication No. 523-0020-006
Maintenance Book, DF-202 ADF System, Publication No. 523-0020-004
Overhaul Book, DF-202 ADF System, Publication No. 523-0020-003
Illustrated Parts Catalog, DF-202 ADF System, Publication No. 523-0020-005
Installation Book, 860E-1 DME Transceiver, Publication No. 523-0050-006
Maintenance Book, 860E-1 DME Transceiver, Publication No. TD 519A
Overhaul Book, 860E-1 DME Transceiver, Publication No. 523-0050-003
Illustrated Parts Catalog, 860E-1 DME Transceiver, Publication No. 523-0050-005
Installation Book, FD-105 Flight Director, Publication No. 523-0019-006
Maintenance Book, FD-105 Flight Director, Publication No. 523-0019-004
Overhaul Manual, FD-105 Flight Director, Publication No. 523-0019-003
Illustrated Parts Catalog, FD-105 Flight Director, Publication No. 523-0019-005
Installation Book, 621A-2/2A Radar Beacon Transponder, Publication No. 523-0091-006
Maintenance Book, 621A-2/2A Radar Beacon Transponder, Publication No. 523-0091-004
Overhaul Book, 621A-2/2A Radar Beacon Transponder, Publication No. 523-0091-003
Illustrated Parts Catalog, 621A-2/2A Radar Beacon Transponder, Publication No. 523-0091-005
Installation Book, 51V-3 Glideslope Receiver, Publication No. 520-5444-006
Maintenance Book, 51V-3 Glideslope Receiver, Publication No. 520-5444-004
Overhaul and Illustrated Parts Catalog, 51V-3 Glideslope Receiver, Publication No. 520-5444-003

DECOTO AIRCRAFT, INC., YAKIMA, WA 98901

Overhaul Manual, Speed Brake Actuator, Part No. 2-2391110

DELAVAL SPECIAL PRODUCTS DIVISION, CLEVELAND, OH 44121

Overhaul Manual, Thrust Reverser Actuator, Part No. 5610-2-3

GABB SPECIAL PRODUCTS, WINDSOR LOCKS, CT 06096

Overhaul Manual for FC3500 Cap and Adapter Assembly, 28-10-1

GENERAL ELECTRIC COMPANY, AIRCRAFT EQUIPMENT DIV., WILMINGTON, MA 01887

Overhaul Manual, G-E Model 8AW59FAC1 Frequency Meter, Publication No. GEI-73304

B. F. GOODRICH AVIATION PRODUCTS; AKRON, OH 44318

Repair Manual, De-Icer Boots; Publications No. 55-169 and 59-131 Maintenance Instructions, Report 61-128C, High Pressure Pneumatic Deicers

B. F. GOODRICH, AEROSPACE AND DEFENSE DIVISION, P.O. BOX 340, TROY, OH 45373

Maintenance and Overhaul Manual for 2-798-4, -5 Multiple Disk Brakes
Maintenance and Overhaul Manual 567 for 2-1123-2 Multiple Disk Brakes
Maintenance and Overhaul Manaul for 3-1184 MLG Wheel
Maintenance and Overhaul Manual 667 for 3-1274 MLG Wheel
Maintenance and Overhaul Manual for 3-999 NLG Wheel

Use and/or disclosure is governed by the statement on the title page of this document.

GOODYEAR TIRE AND RUBBER COMPANY; AKRON, OH 44316

Description and Specifications, Anti-Skid Control Box; Drawings No. 5D1-1071 and 5D4-226 (current revisions)
Maintenance Instructions, Goodyear Anti-Skid System Used on Lockheed JetStar; AP-139

HTL ADVANCED TECHNOLOGY; MONROVIA, CA 91016

Overhaul Manual 26-20-00, 2241-005 Fire Extinguisher Container

WALTER KIDDE CO., INC.; BELLEVILLE, NJ 07109

F-40826 Recharge Instructions, 891471-02 Fire Extinguisher Container
Service Bulletin 26-20-240, Water Fire Extinguisher
Service Bulletin 26-20-245, Water Fire Extinguisher Conversion
Service Directive 7301, CO_2 Fire Extinguishers 870883 and 870887
Service Memo No. 46, Discontinued Aircraft Water Fire Extinguishers
Overhaul Manual, Fire Detector Control Unit, 891491-120-01,-02

MARATHON BATTERY CO., WACO, TX 76710

Description of Nickel-Cadmium Storage Batteries, T.O. 8D2-3-1
Service Manual, Nickel-Cadmium Storage Batteries

MIDLAND-ROSS CORPORATION, JANITROL AERO DIV., COLUMBUS, OH 43204

Maintenance Instructions, Electronic Cooling and Cabin Pressure System, Publication No. 39D69.
Maintenance Instructions, Pressure Regulation and Safety Valve Assembly, Part No. E32C84. Publication No. 39D70.
Maintenance Instructions, Shutoff Valve Assembly, Part No. D70C19. Publication No. 39D71.
Maintenance Instructions, Shutoff Valve Assembly, Part No. B74C94. Publication No. 39D72.

NATIONAL WATER LIFT CO., ABVEX, NWL Division, KALAMAZOO, MI 49001

Technical Telephone: (616) 384-3436
Fax: (616) 384-3436
Maintenance Manual with Parts List for JSM170 and JSM190 Elevator and Aileron Boost Assembly
Service Bulletin No. 175, Modification of JSM170 and JSM190 Boost Assemblies

THE NORDAM GROUP, TULSA, OK 74101-3365

Overhaul Manual, Thrust Reverser 208-002, 78-30-04
Thrust Reverser Fairing Instl., Procedure 208D0006

PARKER-HANNIFIN CORP., IRVINE, CA 92664

Overhaul Instructions, Accumulator Assembly

PURITAN BENNETT AERO SYSTEMS, LENEXA, KS 66215

Oxygen components

ROBERSHAW CONTROLS COMPANY, ANAHEIM CA

Overhaul manual, 900-700-035, Portable Oxygen

ROCKER INDUSTRIES. HARBOR CITY, CA 90710

Overhaul Manual, Thrust Reverser Actuator, Part No. 51009

ROHR INDUSTRIES INC. (See THE NORDAM GROUP)
SARGENT CONTROLS, TUSCON, AZ 78062

Technical information, Pitch Trim Actuators.

Use and/or disclosure is governed by the statement on the title page of this document.

SIMMONS PRECISION, VERGENNES, VT 05491

Bulletin No. 304E, Installation and Maintenance Data for Simmons Pacitron Capacitance Fuel Quantity Gage System on Lockheed JetStar
Service Bulletin No. 1096-29 Modification of All Lockheed JetStar Indicators

SOUTHWESTERN INDUSTRIES INC., LOS ANGELES, CA 90045

ES 151 Maintenance Instructions, Flap Actuator

SPERRY CORP., GREAT NECK, N.Y. 11020

Description of C-10 Gyrosyn Compass System,,Publication Nos. 15-3201 and 60-15-30
Maintenance Manual, SP-40 Automatic Flight Control System, Publication No. LJ-15-0142
Overhaul Manual, Gyrosyn Compass Indicators, Publication No. 15-3120
Overhaul Manual, Compass Rack, Publication No. 15-3108
Overhaul Manual, Servo Amplifier, Publication No..15-3106
Overhaul Manual, Directional Gyro, Publication No. 15-3203
Overhaul Manual, Flux Valve and Compensator, Publication No. 15-3210

THE TALLEY CORPORATION, NEWBURY PARK, CA 91320

Overhaul Manual and Illustrated Parts List, Trim Actuators, Part Number 718T100 and 749T100

TAVCO INC., CHATSWORTH, CA 91311

Overhaul Manual, Air Bottle/Reservoir, 23111108-1, 23111188 and 23111111-1

TECHNICAL ORDERS

Inspection, Maintenance, Storage, and Disposition of Aircraft Tires, T.O. 4F-1-3
Assembly and Installation Instructions for MS Flareless Tube Fittings, T.O. 44E-1-10

TELEDYNE AVIONICS, CHARLOTTESVILLE, VA 22902

Overhaul Manual, MS-927 Vertical Speed Indicator

TEXTRON CORP., PACOIMA, CA 91331

Maintenance Manual 26-20-1, Fire Extinguisher Directional Control Valve
Overhaul Manual and Recharge Instructions, 26-20-75 Fire Extinguisher Container
Overhaul and IPC, 28-10-39, Air Pressure Regulator

WHITTAKER CORP., LOS ANGELES, CA 90024

Overhaul Instructions, Firewall Fuel Shutoff Valve, Part No. 134595, Publication No. O.I. 28-13
Overhaul Instructions, Fuel Jettison Shutoff Valve, Part No. 134605, Publication No. O.I. 28-12
Overhaul Instructions, Manual Defueling Shutoff Valve, Part No. 134615, Publication No. O.I. 28-14

Note

Unclassified Technical Orders may be ordered from the nearest Field Office of the U.S. Department of Commerce.

Note

JT12 Engine Manuals and Service Bulletins are available from:

Pratt and Whitney Aircraft
Product Support Department
400 Main Street
East Hartford, CT 06108

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ADDITIONAL SUPPORT

ENGINES SUPPORT

Allied Signal Aerospace
Telephone 602-365-2493
Fax 602-365-5577
TFE 731 Engines

731 JETSTAR PROGRAM

Garrett Aviation Services
Los Angeles, CA
Telephone 310-568-3845
Fax 310-568-3701
731 Coordinator, Jeff Holm

RECORD KEEPING

Aircraft Technical Publishers (ATP), Microfiche
Brisbane, CA 94005
Telephone: (415) 468-1705
Fax: (415) 468-1596

Camps Systems Inc.
Ronkonkoma, New York
Telephone: (516) 588-6064
Fax: (516) 588-3294
Maintenance and record keeping

RVSM EVALUATION

Aeromech
1309 Bonneville Ave
Snohomish, Washington 98367
Telephone 425-252-3236

Rockwell Collins
Cedar Rapids, IA 52498
Technical Telephone 319-295-3236
Certification Telephone 319-295-2185

THRUST REVERSERS

Nordam "Purchased program from Rohr"
Telephone 918-560-8650
Fax 918-560-2604

TRAINING

Flight Safety International, Inc.
College Park, GA
Telephone 770-991-6064
Fax 770-991-5959

URANIUM COUNTERWEIGHT REPAIR

Starmet CMI
Barnwell, SC 29812
Repair of depleted uranium counterweights
Telephone 803-259-2321
Fax 803-259-3622

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(Pages Introduction -9 thru Introduction-10B - Deleted.)

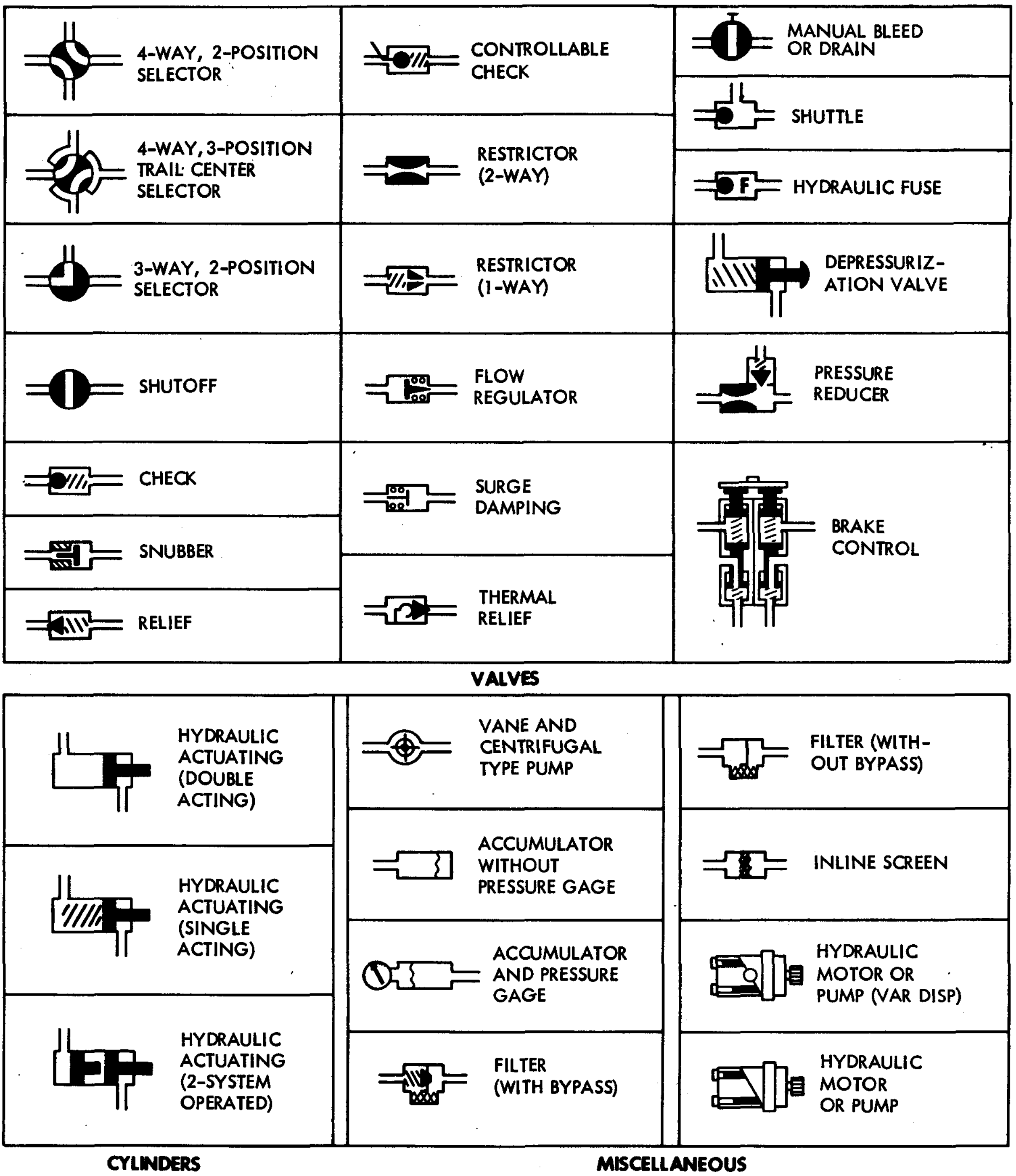

4-WAY, 2-POSITION SELECTOR
4-WAY, 3-POSITION TRAIL CENTER SELECTOR
3-WAY, 2-POSITION SELECTOR
SHUTOFF
CHECK
SNUBBER
RELIEF
CONTROLLABLE CHECK
RESTRICTOR (2-WAY)
RESTRICTOR (1-WAY)
FLOW REGULATOR
SURGE DAMPING
THERMAL RELIEF
MANUAL BLEED OR DRAIN
SHUTTLE
F
HYDRAULIC FUSE
DEPRESSURIZATION VALVE
PRESSURE REDUCER
BRAKE CONTROL
VALVES
HYDRAULIC ACTUATING (DOUBLE ACTING)
HYDRAULIC ACTUATING (SINGLE ACTING)
HYDRAULIC ACTUATING (2-SYSTEM OPERATED)
VANE AND CENTRIFUGAL TYPE PUMP
ACCUMULATOR WITHOUT PRESSURE GAGE
ACCUMULATOR AND PRESSURE GAGE
FILTER (WITH BYPASS)
FILTER (WITHOUT BYPASS)
INLINE SCREEN
HYDRAULIC MOTOR OR PUMP (VAR DISP)
HYDRAULIC MOTOR OR PUMP
CYLINDERS
MISCELLANEOUS

Figure i-1. Mechanical Symbols

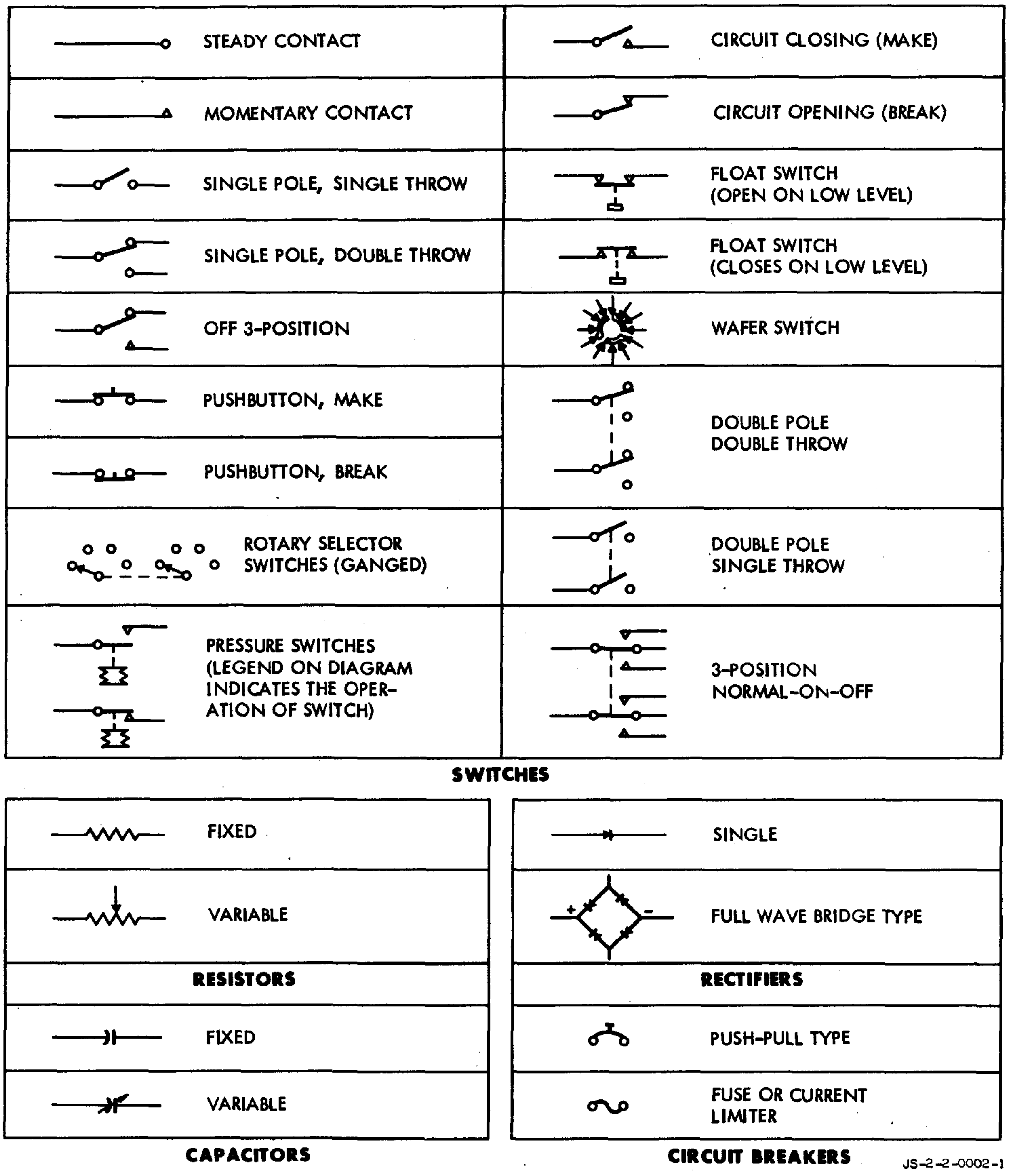
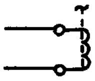
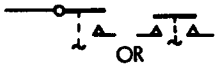
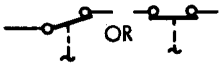
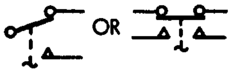
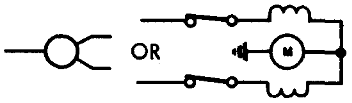
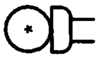
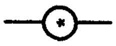


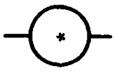
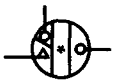
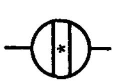
Figure i-2. Electrical Symbols (Sheet 1 of 3)

	COIL
	NORMALLY OPEN (CONTACT MAKE)
	NORMALLY CLOSED (CONTACT BREAK)
	TRANSFER CONTACT

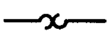
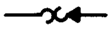
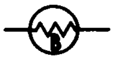
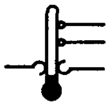
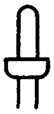
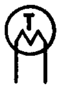
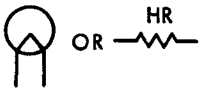
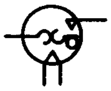
RELAYS

	REVERSIBLE MOTOR
	* INDICATES TYPE PI - POSITION OP - OIL PRESSURE IND XMTR - POS TRANSMITTER
	* INDICATES TYPE M - MOTOR GEN - GENERATOR MG - MOTOR GENERATOR

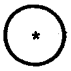
ROTARY EQUIPMENT

	NON-JEWELED INDICATOR OR WARNING LIGHT
	JEWELED INDICATOR WITH PUSH TO TEST CIRCUIT
	JEWELED INDICATOR OR WARNING LIGHT * INDICATES COLOR A - AMBER, R - RED B - BLUE, W - WHITE C - CLEAR, Y - YELLOW G - GREEN, O - ORANGE NE - NEON, P - PURPLE

LAMPS

	BIMETALLIC ELEMENT THERMAL SWITCH
	FLASHER
	BALLAST LAMP
	MERCURY-TYPE THERMAL SWITCH WITH HEATER
	RESISTANCE - TYPE TEMPERATURE BULB
	THERMISTOR
	HEATER ELEMENT
	BIMETALLIC ELEMENT THERMAL SWITCH WITH HEATER

HEATERS

	* INDICATES TYPE A - AMMETER V - VOLTMETER W - WATTMETER VA - VOLT AMMETER CRO - OSCILLOSCOPE F - FREQUENCY METER T - TEMPERATURE METER

METERS

Figure i-2. Electrical Symbols (Sheet 2 of 3)

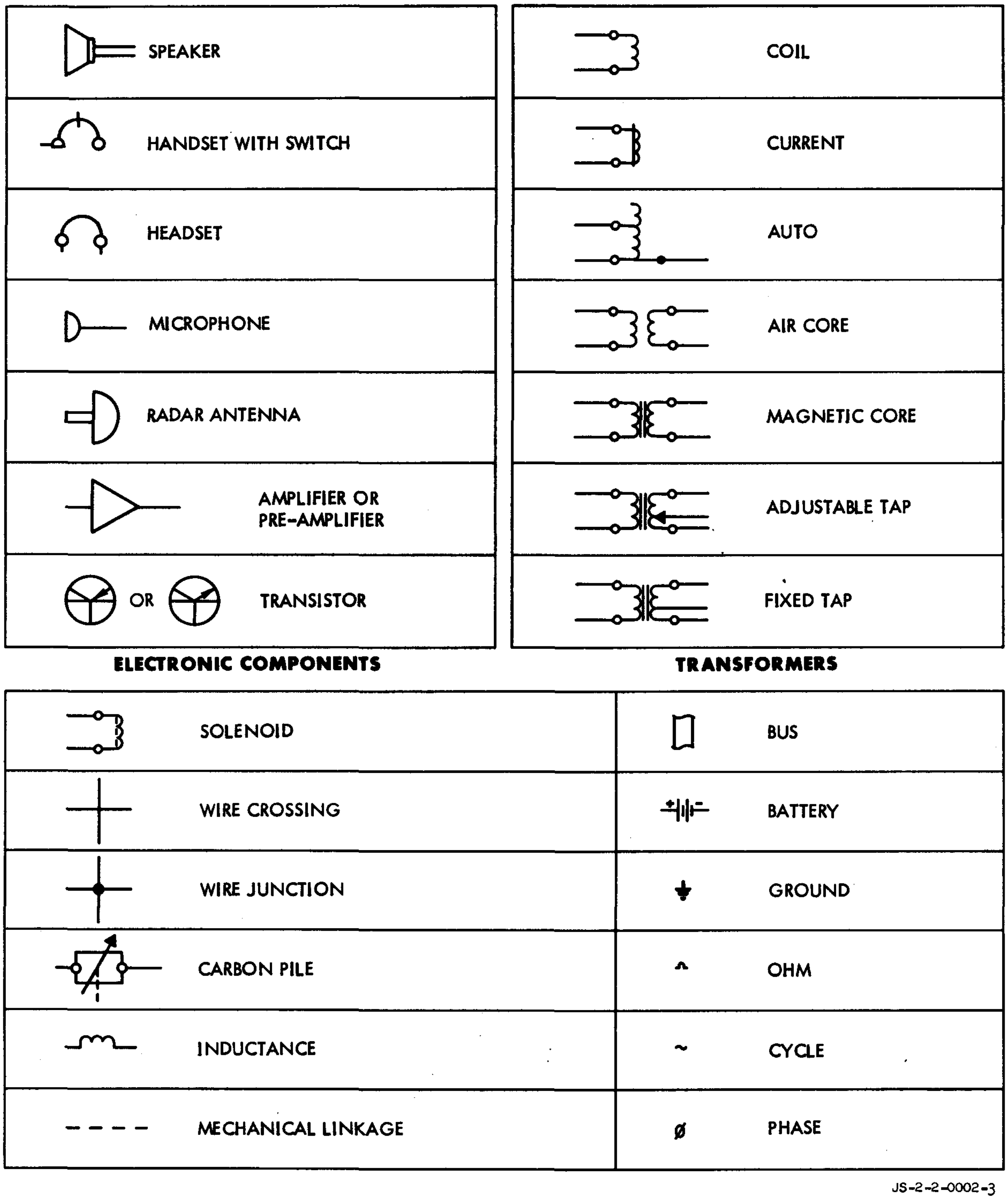


Figure i-2. Electrical Symbols (Sheet 3 of 3)

CHAPTER 5
TIME LIMITS/MAINTENANCE CHECKS

TABLE OF CONTENTS

Section		Page
5.0	GENERAL	5-3
5.1	PRE-FLIGHT INSPECTION	5-5
5.2	POST-FLIGHT INSPECTION	5-7
5.3	100/300-HOUR INSPECTION	5-9
5.4	ANNUAL INSPECTION	5-15
5.5	SPECIAL INSPECTIONS	5-19
5.6	OVERHAUL AND REPLACEMENT SCHEDULE	5-23
5.7	CONTINUED AIRWORTHINESS	5-37

SECTION 5.0

GENERAL

5.0.1 MANUFACTURER'S RECOMMENDATIONS. These inspections are Lockheed's recommendations to the operator and are based on average operating conditions. They have been derived to comply with the current requirements of FAR Part 91, Subpart E, to serve as "a current inspection program recommended by the manufacturer." The listing of an inspection here does not make it mandatory; however, it may be invoked as mandatory by the FAA at the time the individual operator files his program or at some later time. Possible discrepant conditions in systems, components and areas have been defined in general terms rather than by detail listing. These data should enable an operator to establish a record system or checklist for each inspection to suit his own operation. When setting up such a program or planning for a particular inspection, the operator should be cognizant of the information given in Section 5.6 Overhaul and Replacement Schedule, and include relevant items in his program to avoid unnecessary duplication of maintenance work. Routine servicing and lubrication as specified in Chapter 12 can also be phased into an inspection work package where applicable. The omission of an inspection from this program does not relieve the operator from complying with all current requirements and future revisions of FAR Part 43 and FAR Part 91.

Note

> An individual operator may elect to use the information contained in this program to develop his own program of another type, such as a progressive inspection program in accordance with FAR Part 91 and approved by the FAA.

5.0.2 OWNER/OPERATOR RESPONSIBILITIES. The JetStar owner/operator has the responsibility to maintain the airplane in an airworthy condition, including compliance with FAA FAR Part 39. Maintenance should be performed in accordance with FAA FAR Part 91, Subpart E, and any other applicable regulation. Refer to FAA FAR Part 91.403, FAR Part 121.363, and FAR Part 135.413, as applicable. The airplane should be maintained in accordance with instructions in the HOMI and/or other applicable documents.

5.0.3 REPLACEMENT PARTS. Hi-Tech Aero Spares maintains a reasonably adequate stock of parts for maintenance and repair of JetStars. Some JetStar functional components are available from holders of Parts Manufacturer Approval (PMA). Used and repaired parts and assemblies are available from Hi-Tech Aero Spares and other sources. The supplier of new/used/surplus/spare JetStar components should execute an FAA Form 8130-3, "Authorized Release Certificate Approval Tag", for each part or assembly; several parts may be listed on a single form provided each is fully documented. "User/Installer Responsibilities" are listed on the FAA Form 8130-3.

Useful guidance material is provided by FAA Advisory circulars AC No. 20-62C, "Eligibility, Quality, and Identification of Approved Aeronautical Replacement Parts", and AC No. 21-29A, "Detecting and Reporting Suspected Unapproved Parts".

SECTION 5.1

PRE-FLIGHT INSPECTION

Section II of the JetStar Airplane Flight Manual details a pre-flight inspection check list and Lockheed recommends that this be followed or used as a guide by the operator to compile his own.

SECTION 5.2

POST-FLIGHT INSPECTION

As operating and environmental conditions will vary widely according to an operator's route structure and location, it is Lockheed's recommendation that each operator determine the post-flight inspection necessary for his own operation.

SECTION 5.3

100/300-HOUR INSPECTION AT FIRST 100 HOURS AND EVERY 300 HOURS THEREAFTER EXCEPT AS NOTED

Contents Page

DESCRIPTION 5-9
FUSELAGE EXTERIOR 5-10
NOSE ELECTRONICS COMPARTMENT 5-10
FLIGHT STATION 5-10
MAIN POWER SHIELD 5-10
OXYGEN COMPARTMENT 5-10
BATTERY COMPARTMENT 5-11
CABIN 5-11
  Galley 5-11
  Lavatory 5-11
AFT EQUIPMENT COMPARTMENT 5-11
NOSE LANDING GEAR WHEEL WELL 5-12
MAIN LANDING GEAR WHEEL WELLS 5-12
EMPENNAGE 5-13
WINGS 5-13
  External Wing Tanks 5-13
WING ROOT AREA 5-13
ENGINES 5-14
NACELLES 5-14
OPERATIONAL CHECKS 5-14

Note

Substitution of the 300 hour inspection for the 100 hour inspection is not applicable to aircraft carrying any person for hire such as operations under FAR Part 121.

5.3.1 DESCRIPTION. This is a thorough visual inspection of the airplane with operational checkouts. For location of any panel or door which needs to be removed for this inspection, refer to Chapter 52 for details. Some of the conditions to look for which apply in general to the area breakdowns are as follows:

a. SKINS AND FAIRINGS - Scratches, dents, punctures, cracks, loose or missing attachments.

b. STRUCTURE - Cracks, dents, distortion, loose or missing attachments and corrosion.

c. WINDOWS AND WINDSHIELDS - Scratches, cracks, crazing, evidence of leakage, security and cleanliness.

d. PLUMBING - Evidence of leakage, dents, chafing, loose clips, loose connections and cleanliness.

e. CONTROL CABLES - Wear, fraying, chafing, incorrect pull-off, lack of necessary lubricant and cleanliness.

f. ELECTRICAL WIRING AND CONNECTIONS - Fraying, condition of protective coverings of wire bundles in specified areas (condensation, etc.) loose connections and cleanliness. Inspect wiring in the main landing gear trunnion joint area in accordance with instructions in Chapter 32.

g. ELECTRONIC UNITS - Cleanliness, condition, security (shock mounts, etc.).

h. ANTENNAS - Security, damage, condition.

i. LIGHTS - Burned out or loose lamps and cracked covers or shields.

j. HYDRAULIC, MECHANICAL, AND ELECTRICAL COMPONENTS - Damage, security, leakage, evidence of overheating, lack of lubrication, loose connections and cleanliness.

k. OXYGEN EQUIPMENT - Cleanliness: Presence of grease, oil or moisture, proximity of electrical connections or wiring.

l. INLETS, OUTLETS AND DRAINS - Obstructions, cleanliness, and correct operation of all flapper and other type valves.

m. SEALS - Deterioration, cleanliness, security, need for lubrication; wear.

n. MANDATORY MARKINGS - Legibility and condition of paint and decals.

5.3.1 (Cont).

o. CORROSION - Discoloration, absence of protective finish, roughening or blistering of paint and deposits.

p. HEAT DAMAGE - Distortion, discoloration and corrosion.

q. CONDENSATION - Dampness, condition of drain valves, discoloration and corrosion.

In the following area breakdowns specific components, systems and details are given together with special checks where applicable, and should be checked for the conditions listed above.

5.3.2 FUSELAGE EXTERIOR.

a. Fuselage skins and all fairings.

b. All fuselage inlets, outlets, and drains.

c. Radome including lightning protection foil strips for continuity and contact.

d. Windshield and windshield wipers. Wiper blades and arms for damage and wear.

e. Cabin windows.

f. All antennas.

g. Cabin entrance door for operation, upper track and rollers, lower rail and "A" frame, locking bayonets for excessive wear and door seal for correct seating in retainer.

h. Entrance ladder and ladder track.

i. Lower anti-collision light and wing illumination light including presence of condensation.

j. Drag chute compartment doors and hook mechanism. (Airplanes with operable drag chute system only.)

k. Mandatory markings.

l. Pitot mast drain points.

5.3.3 NOSE ELECTRONICS COMPARTMENT.

a. Nose section latches, including drain flapper seals.

b. Proper closure of nose section seal.

c. Locking bolts for excessive wear, track and rollers.

d. All electronic units, weather radar antenna, and glide slope antenna.

e. Pitot and static lines and check drain points for condensation.

f. All accessible wiring and connections.

g. Flow limiter valve.

h. Ducting components to the fan flow regulator valve.

i. Fan flow regulator (forward and aft side of Fuselage Station 164 bulkhead).

j. The windshield heat control box and related relays and contactors, including evidence of overheating.

k. Forward side of F.S. 164 bulkhead structure.

5.3.4 FLIGHT STATION.

a. Crew oxygen masks, mask hose, crew regulators (breathe in hose to check diaphragm), mask microphone, hand microphone, magnetic head set and smoke goggles.

b. Portable oxygen cylinder for correct pressure. (1800 PSI)

c. Clear vision windows for correct operation and hinge latch mechanism for wear. Windshield seal and striker.

d. Crew seats for correct operation and seat tracks including locking pin holes for cleanliness.

e. Seat belts, harnesses and inertia reels for operation.

f. All instruments.

g. All lights, lenses, switches and circuit breakers.

h. Weather radar scope.

i. All control panels for damage, loose knobs and illegible markings.

j. Parking brakes and rudder lock for correct operation.

k. The flight station temperature sensor. (Airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, only).

5.3.5 MAIN POWER SHIELD.

a. All current limiters and fuses.

b. All relays and contactors.

c. All accessible wiring and terminals.

5.3.6 OXYGEN COMPARTMENT.

a. Oxygen cylinder mounts.

5.3.6 (Cont).

b. Oxygen lines and bond jumpers.

c. Oxygen cylinder shut-off valves and oxygen selector valves.

d. Spare current limiter and fuse supply.

e. Compartment door latches.

f. Oxygen cylinders for correct pressure and service to operator's specifications.

5.3.7 BATTERY COMPARTMENT.

Note

The battery compartment shall be inspected every 100 hours.

a. Battery leads and connectors. (Refer to paragraph 24.0.4.5)

b. Battery mounting.

c. Remove, service, and re-install batteries in reversed positions. (See Note in Section 12.5.)

d. Sump jars, moist pad, and connecting tubing.

e. All wiring, relays, and electronic units in area.

f. Battery box for damage and evidence of leakage.

g. Battery vent for obstruction.

5.3.8 CABIN. (Fuselage Station 232 to Fuselage Station 570)

a. All interior doors for binding and defective latch mechanism.

b. Passenger seats and seat belts for operation and seat tracks including locking pin holes.

c. CO_2 and water fire extinguishers for service.

d. Fire axe.

e. All lights and air conditioning gasper outlets.

f. Escape hatches, operation, lights and seals.

g. Stair and gearbox.

h. The cabin temperature sensor.

i. Check out-flow valves - Airplanes 5001 thru 5045: between F.S. 330 and F.S. 350 under center aisle floor; airplanes 5046 and up, and those incorporating Service Bulletin 329-61: forward face of F.S. 570 bulkhead.

Note

Although galley and lavatory locations may vary according to a customer's interior configuration, for the purpose of this inspection their location is considered as follows:

Galley F.S. 490-F.S. 526
Lavatory F.S. 526-F.S. 570

5.3.8.1 GALLEY

a. Remove all drawers and check for spillage and loose articles.

b. Hot cups, hot plate, oven, and thermos jug.

c. All switches and circuit breakers.

5.3.8.2 LAVATORY

a. Remove toilet, check area for spillage and corrosion, service and re-install.

b. Sink drain mechanism. Check under vanity for spillage and corrosion.

c. Shaver power supply.

d. All lights and switches.

5.3.9 AFT EQUIPMENT COMPARTMENT.

a. Fuel compartment including all valves, lines and compartment covers.

b. Fire bottles, lines and connections.

c. All accessible hydraulic and fuel lines.

d. All accessible control cables and pulleys.

e. All bleed air lines and connections.

f. All air conditioning units and ducting. With APU installed, air conditioning unit oil reservoir should be checked more often with ground operation in areas of high humidity.

g. All accessible wiring, terminals, relays, and connections.

h. Pitch trim actuator including electrical connections. Check for water accumulation. (See Section 12.9.)

i. No. 1 and No. 2 system hydraulic filters for brass particles, segments of O-rings, etc.

j. Doppler radar antenna and R.T. unit if fitted.

5.3.9 (Cont).

k. Speed brake including blow-in doors, hinge points, actuators and their attachment points for wear.

l. Speed brake uplock and switch for wear.

m. Compartment overheat warning.

n. Fire extinguisher control valves.

o. Deleted.

p. Air pressure regulator.

q. Clean water separator filter and drain line filter.

r. Stowage box if fitted.

s. Hydraulic system accumulators and thrust reverser accumulator. No. 1 and No. 2 system accumulators 1700 PSI. Thrust reverser accumulator 1200 PSI. (Nitrogen or dry air.)

t. The following hydraulic system valves: Engine-driven pump inlet and outlet shut-off valves and relief valves for system No. 1 and No. 2, speed brake selector valve and thrust reverser selector valves.

u. De-icer distributor valve and the regulator and pressure relief valve.

v. Check reservoir of turbine fan assembly for water contamination. By using syringe and tube remove water as required and replace with oil per instructions in Section 12.8, paragraph 12.8.1.1.

w. On airplanes 5126 and up, check safety valve static sense lines (Fuselage Station 570 Bulkhead). Openings shall be free of obstruction.

5. 3.10 NOSE LANDING GEAR WHEEL WELL.

a. Scissors and disconnect pin, including excessive wear.

b. Nose wheel steering, including excessive wear.

c. All hydraulic lines, fittings and units.

d. All electrical wiring, switches and connectors.

e. Brake accumulator, including correct pre-load. (800 ±50 PSI nitrogen or dry air).

f. Wheel well doors and hinges and all linkages.

g. Nose wheel emergency bottle for service. 1800 to 2500 PSI (Nitrogen or dry air.)

h. Tires including correct pressure. Dash 6 aircraft 180 ±5 PSI, Dash 8 aircraft 220 ±5 PSI. (Nitrogen or dry air.).

i. Landing light.

j. Shock strut, exposed surface for damage and service.

k. Remove at least one right and one left vertical sidewall access plate and check brake valves, cables, pulleys, plumbing, and wiring.

l. Wheel for damage and need of cleaning.

m. Gear actuator, steering actuator and gear door actuator including damage to the sliding surfaces.

n. Gear supporting structure for cracks and loose or missing attachments.

o. Brake selector valve and steering selector valve.

p. Pressure lines from the emergency extension air bottle.

q. Aft side of FS 164 bulkhead structure.

r. Static lines and check drain point for condensation.

s. Check insulation.

t. Visually inspect steering actuator for cracks, distortion, corrosion and hydraulic leakage.

5.3.11 MAIN LANDING GEAR WHEEL WELLS.

a. All landing gear components and all wheel components for evidence of damage and obstruction.

b. All hydraulic and fuel lines.

c. All cables and pulleys.

d. All landing gear doors and their hinge and attachment points for excessive wear.

e. Landing gear and landing gear door actuators.

f. Landing gear uplock mechanism for wear and cracks.

g. Landing gear door uplock mechanism for free release and lack of binding.

h. Emergency gear release system including cables, pulleys, and air bottle for service. 1800 to 2500 PSI. (Nitrogen or dry air.)

i. Shock struts, exposed surface for damage and service.

j. Brakes: Wear indicators and leakage.

k. Tires including pressure. Dash 6, 205 ±5 PSI; Dash 8, 220 ±5 PSI. (Nitrogen or dry air.)

l. Pressure lines from the emergency extension air bottle.

m. Side brace actuators. Check moisture drain for obstructions.

n. Drain safety valve static line (left wheel well, Airplanes 5001 through 5125 only). See Section 21.1.

Note

Static line should be drained every 100 hours.

o. Inspect wiring in trunnion joint area in accordance with Section 32.0.

p. Main gear uplock accumulators, 500 PSI on Aircraft 5041 and up and aircraft modified by Service Bulletin 329-25B.

5.3.12 EMPENNAGE.

a. Vertical and horizontal stabilizer de-icer boots.
b. Vertical stabilizer to fuselage fairing for excessive wear.
c. Rudder and elevator hinges for excessive wear.
d. Rudder and elevator for correct clearance to shroud.
e. Rudder tab hinge for excessive wear.
f. Rudder tab actuator.
g. Rudder tab operation.
h. Elevator boost package.
i. Elevator torque tube attachment points.
j. Vertical stabilizer attachment.
k. Fuel jettison mast including screen and grommet.
l. Upper anti-collision light and tail navigation light.
m. Mandatory markings.
n. Static dischargers, if installed.
o. Two-inch fuel jettison line for damage and chafing where the line passes through the former. (See figure 52-5, item 26.)
p. Inspect JE 24-1, -3 or -5 empennage pivot fitting assembly in accordance with paragraph 58.3.10. Transmit information on defects found to: Lockheed Western Export Company (see page Introduction-3).
q. Perform visual inspection of fasteners common to empennage pivot fitting and lower vertical stabilizer rear beam. Check for loose, missing or broken fasteners.

5.3.13 WINGS.

a. All wing panels, skins, and fairings.
b. All wing inlets, outlets and drains.
c. De-icer boots and hoses.
d. Wing tip navigation lights.
e. External tanks including electrical wiring and fuel lines.
f. Aileron hinge pins for excessive wear. (Refer to Paragraph 27.0.6.2.)
g. Aileron for correct clearance to shroud.
h. Aileron tab hinge for excessive wear.
i. Aileron operation through extremes of travel with boost off.
j. Leading edge flap attachment and hinge points for wear. (Refer to Paragraph 27.4.6.9.) Trailing edge flap attachment and hinge points for wear. (Refer to Paragraph 27.4.6.7.)
k. L.E. and T.E. flap asymmetry cables and pulleys.
l. L.E. flap drive torque tubes and attachments.
m. T.E. flap closure doors for evidence of binding, missing, or damaged chafing strips and correct rigging in faired position.
n. T.E. flap drive torque tubes and attachments.
o. Tank and boost pump access panels.
p. Mandatory markings.
q. Static dischargers, if installed.
r. Fuel tank drain valves for damage and evidence of leakage.
s. Fuel tank filler caps for evidence of leakage and broken or loose retaining chains.
t. Fuel tank vent for damage and obstruction; and flame arrestor for damage.

5.3.13.1 EXTERNAL WING TANKS.

a. Fuel tank drain valves for damage and evidence of leakage.
b. Fuel tank filler caps for evidence of leakage and broken or loose retainer chains.
c. Fuel tank vent for damage and obstruction, and flame arrestor for damage.
d. Fuel boost pump (pump removal not required).
e. Landing and taxi lights.

5.3.14 WING ROOT AREA.

a. Aileron boost unit.
b. Fuel interconnect valves and all fuel lines.
c. Leading edge flap motor-driven gear box (forward-left hand side) and torque tubes.
d. Trailing edge flap motor-driven gear box (aft-right hand side) and torque tubes.
e. Leading edge flap selector valve - No. 1 system - (forward-left hand side).
f. Trailing edge flap selector valve - No. 1 system - (aft-left hand side).
g. Leading edge and trailing edge flap selector valve - No. 2 system - (aft-right hand side)
h. Anti-skid control boxes for Dash 6 and 6A installations located inside the lower right wing to fuselage fairing between stations 450 and 470. Anti-skid control box for Dash 8 engine installation located centrally under walk way between stations 330 and 350.

5.3.14 (Cont).

i. De-icer units: Distributor valve, pressure transmitter, low vacuum warning switch (right hand side).

j. Fuel boost pumps. (pump removal not required)

k. At first 100-Hour Inspection, and thereafter at each Annual Inspection: wing to fuselage attachment bolts for proper torque (see Figure 51-3, Chapter 51) (requires fairing removal).

5.3.15 ENGINES.

a. This inspection is to be accomplished in accordance with the requirements for the minor periodic inspection in the Pratt & Whitney JT12A Maintenance Manual.

b. Engine mounts for damage and loose attachments.

5.3.16 NACELLES.

a. All exterior skins and fairings.

b. Access doors including hinges for excessive wear, cracks, binding and loose or missing attachments.

c. Tail pipe and exhaust cone for distortion and cracks.

d. Thrust reverser doors, tracks, and actuation linkage for damage, excessive wear, and loose or missing attachments.

e. Pressure ratio pitot tube, starter-generator, tachometer generator and inlet duct anti-icing valve.

f. Oil temperature bulb, oil pressure transmitter, oil pump, and oil tank.

g. Fuel pump, fuel control, fuel pressure switch and fuel flow transmitter.

h. Hydraulic pump.

i. Oil, fuel and hydraulic lines.

j. Accessible fire sensing elements for breaks, chafing, and damaged insulation; and insulated clamps for damage.

k. At first 100-Hour Inspection, and thereafter at each Annual Inspection: nacelle to fuselage attachment bolts for proper torque (see Figure 51-1, Chapter 51) (requires fairing removal) (1000 to 1300 inch-pounds).

5.3.17 OPERATIONAL CHECKS.

a. All cockpit lights including personnel warning lights, instrument panel lights, and press-to-test lights; all passenger compartment lights; and all exterior lights for correct operation.

b. Landing gear position indicators.

c. Fuel quantity gages for readings compatible with the known contents of the fuel tanks.

d. Oil temperature indicators.

e. Oxygen warning system. Manually drop passenger masks. (See Chapter 35.)

Note

Return mask regulators to 100% setting before restowing.

f. De-icing system through one heavy icing cycle.

g. Nacelle overheat and fire warning switches.

h. Fire extinguishing containers (pressure vs. temperature check), directional control valves and fire detector control units.

i. Smoke detector(s) for correct operation.

j. Windshield wipers (on wet windshield). Check wipers for correct "Park" position. (See Section 30.3.)

k. Windshield heat.

l. Inverters for correct voltage and frequency. (115 volts; 400 hertz.)

m. Pitot and static heating elements.

n. All control surfaces and trim tabs.

o. L.E. and T.E. flaps on No. 1 and No. 2 systems including correct operation of L.E. downstops and asymmetry.

p. Stabilizer pitch trim. (See Section 27.3.)

q. Electrically-operated hydraulic pump.

r. Fuel jettison for correct operation of each valv

s. All bleed air valves.

5.3.17 (Cont).

t. Perform operational check of electrical power system as outlined in Section 24.0 (including ground fault protection circuit check per paragraph 24.0.1.7).

u. VHF transmitter and receiver, ADF receiver, and VOR receiver. (Check these items in both Systems 1 and 2.)

v. Operate drag chute. (Airplanes with operable drag chute system only.)

w. Auto-pilot engage coupler. (Refer to Chapter 22.)

x. All electrically-operated gyro instruments.

y. Cycle landing gear on normal system.

z. Free-fall landing gear by emergency release system (activation of blow-down bottle not required).

aa. Operate emergency standby release system on Airplanes 5041 and up and those incorporating Service Bulletin 329-25 (activation of blow-down bottle not required).

ab. Speed brake. (Functionally check operation of both retract solenoids.)

ac. Stair and gearbox.

ad. Spot check anti-skid system. (Audible "click" is heard when system is activated and skid detectors are flipped into forward rotation.) (See Section 32.3.)

SECTION 5.4

ANNUAL INSPECTION

Contents	Page
DESCRIPTION	5-15
FUSELAGE EXTERIOR	5-15
NOSE ELECTRONICS COMPARTMENT	5-15
FLIGHT STATION	5-15
MAIN POWER SHIELD	5-15
OXYGEN COMPARTMENT	5-15
BATTERY COMPARTMENT	5-15
CABIN	5-15
AFT EQUIPMENT COMPARTMENT	5-16
NOSE LANDING GEAR WHEEL WELL	5-16
MAIN LANDING GEAR WHEEL WELLS	5-16
EMPENNAGE	5-17
WINGS	5-17
External Wing Tanks	5-17
WING ROOT AREA	5-17
NACELLES	5-18
OPERATIONAL CHECKS	5-18

5.4.1 DESCRIPTION. This inspection fulfills the normal requirements of the Periodic Inspection required for FAA annual recertification and consists of the complete 100/300-Hour Inspection plus the extra requirements listed in this section. The same definitions of specific conditions as described in the 100/300-Hour Inspection apply to the following area breakdowns.

Note

Perform any special inspections in Section 5.5 that may be required during this annual.

5.4.2 FUSELAGE EXTERIOR. Same as paragraph 5.3.2.

5.4.3 NOSE ELECTRONICS COMPARTMENT.

a. Interior structure (including Fuselage Station 164 bulkhead).

5.4.4 FLIGHT STATION.

a. Windshield wiper motor and flexible drive.

b. Accessible interior structure including forward and aft face FS 164 bulkhead.

c. Throttle quadrant cables and mechanism (including thrust reverser levers).

d. Throttle friction locks for correct operation.

e. Cockpit speaker and interphone amplifier.

f. Cabin pressurization filters for excessive dirt. (See Section 21.1 and Section 21.2.)

g. Autopilot servos.

h. Remove floor boards for detailed inspection. Check floor panels and tracks.

i. Check behind all circuit breaker panels.

j. Check behind overhead console panels.

k. Check behind center stand panels.

l. Check behind side console panels.

m. Control column for excessive play.

n. Condition of all placards, decals and markings.

5.4.5 MAIN POWER SHIELD. Same as paragraph 5.3.5.

5.4.6 OXYGEN COMPARTMENT. Same as paragraph 5.3.6.

5.4.7 BATTERY COMPARTMENT. Same as paragraph 5.3.7.

5.4.8 CABIN.

a. Lift all floorboards, including vestibule (not in ladder enclosure), galley and lavatory (inspection plates, if fitted). Check floor panels and tracks.

b. L.E. flap 120° gear box and connecting torque tubes.

5.4.8 (Cont).

c. Anti-skid selector valves.

d. Main, standby and windshield inverters.

e. Check all drain holes under floor, including thru longeron and frames.

f. Cabin pressurization pneumatic relay and filter, outflow valves, jet pump and jet pump pressure regulator and pneumatic relay. Clean outflow valves (nicotine on knife edges). Airplanes 5001 through 5045: Located between FS 330 and FS 350 under center aisle floor; Airplanes 5046 and up, and those incorporating Service Bulletin 329-61: Forward face of FS 570 bulkhead.

g. Check alumina desiccant in cabin window de-mist tubes.

h. Check installation of the flight station high limit thermostat and the flight station temperature anticipator (over right-hand wheel well).

i. Pitch-trim relays.

j. Aileron tension regulator.

k. All air-conditioning ducting.

1. Insulation and sound-proofing for moisture.

m. Check forward face of Fuselage Station 570 bulkhead structure.

n. Inspect cabin windows for cracks or separation in or near the rabbet radius in the outer panel. (See figure 56-11.)

5.4.9 AFT EQUIPMENT COMPARTMENT.

Note

As used in this paragraph, "Structural check" is defined as a visual check of subject structure for cracks, buckling, distortion, loose, damaged or missing attachments, and corrosion. When conditions warrant, refer to Chapter 58 for more detailed inspections.

a. Check aft face of FS 570 bulkhead structure.

b. All plugs, grommets and seals, etc. at pressure bulkhead, Fuselage Station 570.

c. Structural check of the engine support frames at Fuselage Stations 589 and 608, and the bulkheads at Fuselage Stations 682 and 739.

d. Structural check of generator control panel rack. Also, check for illegible markings and mounting of the attached equipment.

e. The following components of the air conditioning and pressurization system: Water separator, refrigerator assemblies, emergency pressurization heat exchanger, bleed air inflow control, cabin temperature control box, low limit thermostat (except on airplanes 5086 through 5092, and 5094 through 5096 not having Service Bulletin 329-234 incorporated), cabin high limit thermostat, cabin temperature anticipator and all of the system valves and ducting.

f. Refrigerator unit oil reservoir for need of servicing. Check for corrosion on bottom of refrigerator core.

g. Visual check of the electric motor-driven hydraulic pump and the hydraulic pressure transmitters.

h. Hydraulic reservoir pressure regulator.

i. Structural check of the underside of the shelf under the vertical stabilizer between Fuselage Stations 690 and 760.

j. Doppler antenna, if fitted.

k. Elevator tension regulator.

5.4.10 NOSE LANDING GEAR WHEEL WELL.

a. Remove all vertical sidewall accessplates (2 left and 2 right) for detailed inspection including structural check.

b. Check side play in gear.

c. Upper trunnion mount.

d. Check clearance of landing gear to overhead floor support structure.

e. Inspect wheels and tie bolts per 5.5.21.

f. Visually inspect all outside surfaces of the shock strut. On aircraft 5001 through 5157 not provided with a JL1666 NLG cylinder, perform the NDE inspection in paragraph 58.5.4. If defects are found, replace the defective cylinder with a JL 1666 cylinder (7049-T73 alloy) or see Technical Information, page Introduction-3. This NDE inspection is not required on JL1666 cylinders.

g. Inspect upper drag link in accordance with paragraph 58.5.2. To report defects see Technical Information, page Introduction-3. Note: This inspection is not required on link Part No. JL1805 (7049-T73 alloy).

h. At each third (3-year intervals) annual inspection, fluorescent penetrant inspect the shock strut internal surfaces per sections 58.5.4.4 and 58.5.4.5. Refer to figure 32-10 for packing instructions with emphasis on step 17.

i. Visually inspect steering actuator for cracks, distortion, corrosion and hydraulic leakage.

5.4.11 MAIN LANDING GEAR WHEEL WELLS.

a. Gear support structure.

5.4.11 (Cont).

b. Landing gear selector valve and gear doors selector valve.

c. Fore and aft play in gear through shimmed upper attach bearings. (See paragraph 32.0.4.4.)

d. Uplock adjustment: Check landing gear uplock and landing gear door (Reference: Section 32.0).

e. Play in side brace actuators.

f. Remove anti-skid generator and check for axle corrosion.

g. Wheel well box structure. On aircraft 5001 through 5146 not modified by Service Bulletin 329-265, perform the NDE inspection in paragraph 58.4.2. If defects are found, install the kit provided by Service Bulletin 329-265.

h. Inspect wheels and tiebolts per 5.5.21.

i. Visually inspect all outside surfaces of the shock strut. On aircraft 5001 through 5159 not provided with a JL1174 MLG cylinder, perform the NDE inspection in paragraph 58.5.4. If defects are found, replace the defective cylinder with a JL1174 cylinder (7049-T73 alloy) or see Technical Information, page Introduction-3. This inspection is not required on JL1174 cylinders.

j. At each third (3-year intervals) annual inspection, fluorescent penetrant inspect the shock strut internal surfaces per sections 58.5.4.4 and 58.5.4.5. Refer to figure 32-11 for packing instructions with emphasis on step 17.

5.4.12 EMPENNAGE.

Note

As used in this paragraph, "Structural check" is defined as a visual check of subject structure for cracks, buckling, distortion, loose, damaged or missing attachments, and corrosion. When conditions warrant, refer to Chapter 58 for more detailed inspections.

a. De-icing system flexible air supply tubing.

b. Structural check of the vertical and horizontal stabilizer rear spars. If any indication of structural or corrosion damage is found, remove leading edge and perform structural check of front spars. Check left and right vertical stabilizer spar caps for cracks.

c. Rudder and elevator counterbalance weights for secure mounting.

d. Structural check of the entire exterior surface of the vertical and horizontal box structure and leading edges.

e. Rudder and elevator hinge pins for damage and excessive wear and the hinge support structure.

f. Elevator support bearings for wear and the bearing support structure.

g. Check all skins in jet blast areas for cracks.

h. Bonding jumpers - all control surfaces.

i. Structural check of empennage pivot fitting (JE24) at lower end of vertical stabilizer rear beam through the access panel. (See Figure 52-5, Item 24, C.)

5.4.13 WINGS.

a. Drop L.E. flaps for thorough inspection of wing L.E.

b. Wing front and rear beams for cracks, loose and missing attachments, and evidence of fuel leakage.

c. All electrical wiring along front beam for evidence of chafing.

d. L.E. and T.E. flap jackscrew actuators and support fittings.

e. L.E. and T.E. flap torque tubes for clearance to structure and universal mechanisms for condition.

f. Remove wing tip: Check flux valve and end rib for cracks.

g. Aileron trim tab position transmitter.

h. Aileron counterbalance weights for loose or missing attachments.

i. Aileron push-pull rods and supports for binding, looseness and need of cleaning.

j. Check for excessive play in ailerons, and L.E. and T.E. flaps.

k. Inspect Part No. JW290-1L/R, -2L/R, -3L/R, -4IL/R and -5L/R; -11L/R, -12L/R. -13L/R, -14L/R, -15L/R, -21L/R, -22L/R, -23L/R, -24L/R and -25L/R wing root rib lower shear fittings, in accordance with paragraph 58.2.3 on airplane serial 5001 through 5059, 5061 and 5064. Defects reported are to be repaired or pans replaced in accordance with Service Bulletin 329-190.

l. Check samples of liquid from fuel sumps for presence of contamination or microbiological growth (Refer to Section 51.7 Corrosion Control and Protection).

m. MLG actuator and auxiliary spar outboard support fitting per paragraph 58.2.6. This inspection is not required on parts with less than 5000 hours service or on replacement 7075-T73 alloy parts (Part No. JW60-5L/R or JW60-7L/R). Refer to Section 59.2 for defect repairs.

n. Make inspection as required by paragraph 5.5.8.1.

o. MLG aft support spar inboard (clevis) end in the area of bushed bolt holes (forward and aft faces). If conditions warrant, refer to paragraph 58.2.4 for more detailed inspection.

5.4.13.1 EXTERNAL WING TANKS.

a. External tank to wing attachments for secure mounting (requires fairing removal).

5.4.14 WING ROOT AREA.

a. Remove access panel on boost pump door and check area for corrosion and fungus.

b. Wing to fuselage attachment bolts for proper torque (See Figure 51-3 Chapter 51) (requires fairing removal).

c. Perform NDE inspection on fuselage main frames in accordance with paragraph 58.4.3 and 58.4.4. This inspection is required except for the following 7075-T73 alloy replacement main frames, in accordance with Service Bulletin 329-269.

Part No.	**Position**
JF 265-2, -3	FS 410
JF 266-2, -3	FS 430
JF 267-2, -3	FS 450
JF 268-2, -3	FS 470
JF 269-3, -4	FS 490

To report defects, see Technical Information, page Introduction-3.

5.4.15 NACELLES.

a. Nacelle to fuselage attachment bolts for proper dry torque, 1000 to 1300 inch-pounds. (Requires fairing removal.)

b. Inspect by visual and/or penetrant inspection methods the upper skin panels of the nacelle box beam to determine extent of cracking and growth, if any, of existing cracks. If the total cross-sectional area of either skin panel in any buttock line plane is reduced by 30 percent or more, a repair to the damaged region must be made.

The repairs, if required, will be handled on an individual basis depending on the exact location. For coordination of repairs, see Technical Information, page Introduction-3.

c. Hydrostatic press test to 750 psi required on fire extinguishing flex hoses between nacelle and fuselage.

5.4.16 OPERATIONAL CHECKS.

a. Cycle landing gear on normal system.

b. Free-fall landing gear by emergency release system, including activation of blow-down bottles.

c. Operate emergency standby release system including activation of blow-down bottles on airplanes 5041 and up, and those incorporating Service Bulletin 329-25.

d. Pressurization system. While airplane is pressurized, check exterior contour of entrance door and emergency exits for step condition. Check for any excessive pressure leaks.

e. Fuel jettison system per paragraph 28.5.3. Check for jettison line leakage.

f. Fire extinguisher system operational checkout as outlined in Section 26.2.1.

g. Make ground fault protection circuit check per paragraph 24.0.1.7.2.

SECTION 5.5
SPECIAL INSPECTIONS

Contents Page

SPECIAL INSPECTION - AFTER EXTREME TURBULENCE OR VIOLENT MANEUVERS 5-19
SPECIAL INSPECTION - AFTER LIGHTNING STRIKE 5-19
SPECIAL INSPECTION - AFTER OVERWEIGHT OR HARD LANDING 5-20
SPECIAL INSPECTION - AT ENGINE REMOVAL 5-21
SPECIAL INSPECTION - ENGINE VIBRATION 5-21
SPECIAL INSPECTION - ENGINE, EXCESSIVE EGT 5-21
SPECIAL INSPECTION - ENGINE, EXCESSIVE RPM 5-21
SPECIAL INSPECTION - FUEL TANKS 5-21
SPECIAL INSPECTION - AT CABIN INTERIOR TRIM REMOVAL 5-21
750 HOUR INSPECTION OF WINDSHIELD POSTS 5-22
600 HOUR INSPECTION (THRUST REVERSERS) 5-22
900 OR 1500 HOUR INSPECTION (JT12A-6A ENGINE) 5-22
1500 HOUR INSPECTION (JT12A-8 ENGINE) 5-22
1000 HOUR OR 2 YEAR INSPECTION 5-22
1500 HOUR INSPECTION OF WINDSHIELD POSTS 5-22
2000 HOUR OR 4 YEAR INSPECTION 5-22
5000 HOUR INSPECTION 5-22A
8000 HOUR INSPECTION 5-22A
60 DAY INSPECTION (DRAG CHUTE) 5-22A
10,000 HOUR INSPECTION 5-22A
5-YEAR CABIN WINDOW INSPECTION 5-22A
SPECIAL INSPECTION-WHEELS 5-22A
3,000 HOUR INSPECTIONS 5-22A
6,000 HOUR INSPECTIONS 5-22B
2,500 HOUR, 5 YEARS MAXIMUM, INSPECTIONS 5-22B

Note

> As used in this section, a structural check or inspection is defined as a visual check of subject structure for cracks, buckling, distortion, loose, damaged, or missing attachments, and corrosion. When conditions warrant, refer to Chapter 58 for more detailed inspection.

5.5.1 SPECIAL INSPECTION-AFTER EXTREME TURBULENCE OR VIOLENT MANEUVERS. A thorough structural inspection should be made in the areas specified below, paying particular attention to the detection of cracks, distortion, buckling, loose, damaged, or missing attachments, security of components and fuel leakage. When conditions warrant, refer to Chapter 58 for more detailed inspections.

a. With electrical power on the airplane, operate all control surfaces and trim surfaces and check for range of travel and binding.

b. Fuselage lower surface and side skins along the entire fuselage length. Pay particular attention to areas around the doors and windows.

c. Fuselage upper skin, especially in the area above the wings.

d. Wing leading edge.

e. Lower wing surface.

f. Upper wing surface.

g. Wing front and rear spars.

h. Wing-to-fuselage fillets.

i. L.E. and T.E. flaps.

j. Aileron skin and hinges.

k. Aileron counterbalance weights.

l. Wing-to-fuselage attachment fittings and bolts; retorque the bolts.

m. Horizontal and vertical stabilizer skins.

n. Fillets between the horizontal and vertical stabilizers.

o. Rudder and elevator skins.

p. Elevator and rudder hinges and supporting structure.

q. Elevator and rudder counterbalance weights.

5.5.2 SPECIAL INSPECTION-AFTER LIGHTNING STRIKE.

a. With electrical power on the airplane, operate all control surfaces and trim surfaces and check for free and smooth operation.

b. Inspect the radome for pin holes. These holes may be too small to be readily visible, in which case, shine a bright light on the opposite surface to increase their visibility, or use a soap bubble type test with the radome under pressure.

c. Inspect the electrical conductor foil strips just beneath the radome surface for damage.

5.5.2 (Cont).

d. Inspect the radome rollers, tracks, and latches for damage.

e. Visually inspect all protruding antennas for damage.

f. Inspect the entire exterior surface of the airplane for pitting, dents, holes and popped rivets.

g. If popped rivets are found in the exterior surface, also inspect the interior structure in the affected area for damaged attachments.

h. Inspect all leading edge de-icing boots for pin holes.

i. Inspect the fuel vents, drains and flame arrestor for damage.

j. Inspect all bonding wires to ensure that they are intact.

k. Inspect all exterior lights for burned bulbs.

5.5.3 SPECIAL INSPECTION-AFTER OVERWEIGHT OR HARD LANDING.

5.5.3.1 Inspect all exterior skin and the areas specified below for cracks, distortion, buckling, loose, damaged or missing attachments, security of components, and evidence of fuel leakage. When conditions warrant, refer to Chapter 58 for more detailed inspections.

a. Wing rear beam, particularly from the inboard end to a point approximately 12 inches outboard of the main gear aft beam attachment fitting, with emphasis at attachments for the main gear forward trunnion fitting.

b. Main gear support beam entire length, with emphasis around the inboard and outboard end attachments and at the main gear aft trunnion fitting.

c. Forward and aft trunnion fittings for the main gear.

d. Inboard and outboard attachment fittings for main gear aft support beam. Inboard attachment fitting is on the fuselage and the outboard fitting is on the wing rear beam.

e. Main landing gear wheel well structure aft wall, fuselage station 526, inside and outside of aircraft.

f. Visually inspect the main gear for damage. If damage is in evidence or is suspected, disassemble and inspect per paragraphs 58.5.3 and/or 58.5.4.

g. Inspect the wing trailing edge alignment pin for sheared pin, and inspect the alignment pin fittings for sheared fasteners.

h. Wing-to-fuselage attachment fittings at fuselage stations 410, 430, 450, 470, and 490 for cracks and attachment failure.

i. Wing upper surface from wing station 41.5 to wing station 120 between rear and mid beams.

j. Wing lower surface from wing station 41.5 to wing station 120.

k. Visually inspect the nose gear and drag link for damage. If any evidence of damage is suspected, disassemble and inspect per paragraphs 58.5.3. and/or 58.5.4.

l. Entire nose gear wheel well structure with emphasis around the nose landing gear trunnion fitting and trunnion support structure.

m. Nacelle-to-fuselage attachment fittings for cracks and attachment failure.

5.5.3.2. In the event that the landing was unduly severe or if considerable damage is evident, it is recommended that all the NDE inspections referenced in paragraphs 5.5.3.1. be accomplished and the following additional operations and inspections be performed.

a. Alignment and symmetry check. Refer to Section 51.1.

b. Inspect both fuselage lower longerons from fuselage station 490 to 600.

c. Inspect wing-to-fuselage attach bolts (40 places) per paragraph 58.2.2.

d. External tank forward and aft attach bolts and fittings. See paragraph 58.2.8.

e. Inspect landing gear wheels per paragraph 58.5.5. Refer to B.F. Goodrich Maintenance and Overhaul Manual 667.

f. Perform inspection of nacelle-to-fuselage attachment fittings and bolts. See paragraph 58.6.2.

g. Inspect engine mount fittings and bolts. See paragraph 58.6.3.

h. All fuselage-to-empennage attachments including pitch trim actuator. See paragraphs 58.3.2, 58.3.3, and 58.3.4.

i. Horizontal stabilizer box beam to horizontal stabilizer center section attachments. See paragraph 58.3.5.

j. Inspect rudder bearings and support structure.

k. Contact engine manufacturer's representative for any special engine inspection.

1. It is recommended that the NDE inspections of paragraph 58.2.4 MLG aft support to fuselage attachment and paragraph 58.2.6 MLG actuator to auxiliary spar outboard support fitting be accomplished.

m. Visually inspect steering actuator for cracks, distortion, corrosion and hydraulic leakage.

5.5.4 SPECIAL INSPECTION-AT ENGINE REMOVAL.

a. Inspect the thrust reverser assembly for cracks, loose or missing attachments, excessive wear and corrosion.

b. Inspect all lines, ducts, plumbing and fittings for evidence of leakage, damage and chafing.

c. Inspect all electrical harnesses for loose connections, worn insulation and chafing.

d. Inspect all insulation blankets for damage

e. Inspect the nacelle interior structure at engine attachment points. When conditions warrant, refer to paragraph 58.6.3.

f. Inspect by visual and/or penetrant inspection methods, the lower skin panels of the nacelle box beam to determine extent of cracking and growth if any, of existing cracks. In case of premature engine removal, however, this inspection is not required more than once a year. If the total sectional area of either skin panel in any buttock line plane is reduced by 30 percent or more, a repair to the damaged region must be made. The repairs, if required, will be handled on an individual basis depending on the exact location. For coordination of repairs, see Technical Information, page Introduction-3.

5.5.5 SPECIAL INSPECTION-ENGINE VIBRATION.

a. When excessive engine vibration is suspected, perform a vibration check. (See Chapter 72.)

5.5.6 SPECIAL INSPECTION-ENGINE, EXCESSIVE EGT.

See Chapter 72.

5.5.7 SPECIAL INSPECTION-ENGINE, EXCESSIVE RPM.

See Chapter 72.

5.5.8 SPECIAL INSPECTION - FUEL TANKS.

5.5.8.1 The inspection in this paragraph is required at the same time the inspection in paragraph 5.5.8.2 is required. In addition to this, it is also required annually on aircraft 5001 through 5055 by AD 68-09-02. On aircraft 5056 and up, it should be performed every 1-1/2 - 2-1/2 years at the midpoint between the 5.5.8.2 inspections. Wing panels that have not been replaced with panels made of 7075-T76511 alloy are to be inspected in accordance with Primary NDE Procedure No. 1 of paragraph 58.2.9. All defects found should be marked on drawing JWK20 (or equivalent) showing crack location, i.e., riser number, left or right wing, end of crack and length of crack. To transmit drawing, see Technical Information, page Introduction-3. Lockheed Engineering will promptly evaluate this information and provide, with FAA concurrence, any requirements for more frequent inspections and/or repairs that may be necessary.

Note

The primary NDE Procedure No. 1 of paragraph 58.2.9 is considered equivalent/superior to the Phase I and Phase II inspections described in Service Bulletin 329-257E.

Note

In the event that a crack indication is recorded the discrepant area(s) should be reinspected at intervals not exceeding 180 days. Use the inspection techniques of primary NDE Procedure No. 1 of paragraph 58.2.9

5.5.8.2 Three/Five Year Tank Open Inspection. This inspection of the interior of each internal and external fuel tank is recommended every three years. Operators who obtain good results from the inspection in paragraph 5.5.8.1, use carefully supervised fuel with anti-micro-biological additives, make frequent fuel checks that show no fuel contamination and follow the recommendations of Service Bulletin 329-288, may safely reduce the frequency of this inspection to five years. This inspection will consist of the following:

Note

Prior to opening the tanks, a fuel tank leakage check should be made so that repairs may be made, if required, while the tank is open.

a. Perform a thorough structural inspection (with particular attention to the detection of cracks, buckling, distortion, corrosion, loose, damaged or missing fasteners) of tank interior, root area, aileron, bellcrank area, LE flap support area, external tank attachment area, box ribs, and forward, center, and aft spars.

b. Fuel low level warning switches and other tank mounted fuel system components shall be checked for security and condition of connections.

c. Ultrasonically inspect the wing panel risers in accordance with paragraph 58.2.9 using Primary NDE Procedure No. 2 and Backup NDE. Defects found by this inspection shall be reported in the same way as those in paragraph 5.5.8.1.

d. Prior to reinstallation of the panels, make a visual inspection of the holes and nuts used with the five internal wrenching bolts that are installed vertically at the inboard end.

5.5.9 SPECIAL INSPECTION-AT CABIN INTERIOR TRIM REMOVAL.

a. Inspect the fuselage interior structure for cracks, loose or missing attachments and corrosion.

b. Inspect all insulation and sound proofing for moisture.

5.5.10 750 HOUR INSPECTION OF WINDSHIELD POSTS.

Note

This inspection not required prior to 3750 flight hours or on post(s) that have been modified by Service Bulletin 329-275.

On aircraft with unpainted (exterior) windshield posts, clean all six posts and make a visual inspection. Using a magnifying glass, carefully check for skin cracks paying particular attention to the areas around countersunk screw holes. See figure 1 of Service Bulletin 329-275.

For aircraft with painted windshield posts, perform the eddy current NDE inspection procedure No. 2 of paragraph 58.4.5.

Some defects found may be corrected by installing the kit(s) in Service Bulletin 329-275. To report defects, see Technical Information, page Introduction-3.

5.5.11 600 HOUR INSPECTION (THRUST REVERSERS).

Perform 600 hour thrust reverser inspection. (See Chapter 78.)

5.5.12 900 or 1500 HOUR INSPECTION (JT 12A-6A ENGINE).

Note

Engines with 3000 hours T.B.O per 5.6.12, perform this inspection every 1500 hours. Engines with 1800 hours T.B.O. per 5.6.12, perform this inspection every 900 hours. Refer to Pratt and Whitney Service Bulletin 1508.

a. Perform major inspection. (See Chapter 72.)

b. Inspect the main engine mount isolators for wear and/or deterioration. (See Chapter 71.)

c. Perform operational checkout of exhaust gas temperature indicator, pressure ratio indicator, and tachometer in accordance with Chapter 77.

5.5.12.1 1500 HOUR INSPECTION (JT 12A-8 ENGINE.)

Note

Refer to Pratt and Whitney Service Bulletin 1508.

a. Perform major inspection. (See Chapter 72.)

b. Inspect the main engine mount isolators for wear and/or deterioration. (See Chapter 71.)

c. Perform operational checkout of exhaust gas temperature indicator, pressure ratio indicator, and tachometer in accordance with Chapter 77.

5.5.13 1000 HOUR OR 2 YEAR INSPECTION.

a. Remove the boots from the L.E. and T.E. flap actuators and inspect for dried lubricant on the screw. If the lubricant is dry, disassemble and relubricate the actuator per Section 27.4.

b. Inspect the rubber bulb type air seal at the leading edge flap hinge line for damage and/or deterioration.

c. Altimeter systems leak test.

d. Leak test pilot's pitot-static system. (Refer to Section 34.10.)

e. Leak test copilot's pitot-static system. (Refer to Section 34.10.)

5.5.14 1500 HOUR INSPECTION OF WINDSHIELD POSTS.

Note

This inspection is not required prior to 4500 ±500 flight hours or on post(s) that have been modified by Service Bulletin 329-275.

Inspect in accordance with eddy current NDE Inspection Procedure No. 2 and radiographic Procedure No.1 of Para. 58.4.5. To report defects, see Technical Information, page Introduction-3.

5.5.15 2000 HOUR OR 4 YEAR INSPECTION.

a. Disassemble, inspect and relubricate the L.E. and T.E. flap actuators per Section 27.4.

b. Inspect trailing edge flap actuator support (Flap Stations 19 and 58) in accordance with paragraph 58.2.7. Replace defective parts.

c. Inspect the upper and lower support bearings for the aileron tension regulator.

d. Inspect the upper and lower support bearings for the elevator tension regulator.

5.5.16 5000 HOUR INSPECTION.

Note

This is a structural check of certain major components in which particular attention should be given to the detection of cracks, buckling, distortion, loose, damaged, or missing attachments and corrosion. When conditions warrant, refer to Chapter 58 for more detailed inspections.

a. Visual inspection of main frames at fuselage stations 410, 430, 450, 470, 490, and 526 between the floor level and waterline 100.

b. Visual inspection of the upper surface of the shelf between fuselage stations 690 and 760 under the vertical stabilizer.

c. Visual inspection of the under-floor structure throughout the length of the fuselage.

d. Visual inspection of the entire outer surface of the wing box structure including the fittings and attachments to the forward and aft spars.

e. Visual inspection of the inside of the empennage upper vertical box structure for corrosion, and loose or missing attachments.

f. Visual inspection of the inside of the empennage lower vertical box structure.

g. Visual inspection of the inside of the horizontal stabilizer box structure.

h. Visual inspection of the inside and outside of the empennage center box structure.

i. Visual inspection of empennage leading edges.

j. Deleted.

k. Deleted.

l. Perform visual structural inspection of the vertical stabilizer scissors joint area. When conditions warrant, refer to paragraph 58.3.2

m. Perform visual structural inspection of the pitch trim actuator attach fittings. When conditions warrant, refer to the paragraph 58.3.3.

n. Perform visual structural inspection of the empennage hinge fitting area. When conditions warrant, refer to paragraph 58.3.4.

Note

The following inspections in steps o and p are not required prior to 20,000 hours.

o. Inspect vertical stabilizer actuator attach fitting attachment in accordance with paragraph 58.3.7.

p. Inspect empennage pivot fitting to vertical stabilizer rear beam attachment in accordance with paragraph 58.3.9.

5.5.17 8000 HOUR INSPECTION.

a. Inspect empennage scissors fitting, attaching fitting, and bolts per paragraph 58.3.2.

b. Inspect empennage pitch-trim attachment fittings and bolts per paragraph 58.3.3.

c. Inspect empennage pivot fittings, bushings, and hinge pins per paragraph 58.3.4.

5.5.18 60 DAY INSPECTION (DRAG CHUTE).

a. Unpack, inspect and repack the drag chute. (Airplanes with operable drag chute system only.)

5.5.19 10,000 HOUR INSPECTION. Refer to Section 59.9 for the 10,000 hour inspection requirements.

5.5.20 5-YEAR CABIN WINDOW INSPECTION. Inspect the cabin windows in accordance with paragraph 58.4.6 for cracks radiating from attachment holes in the outer window panel and cracks extending from the machined radius in the outer panel.

Visually check the inner panel for cracks radiating from attachment holes. Visually check the inner and outer panels for cracks and crazing in the panel surfaces. If any of the above conditions are found in the window panels, replace the window assembly with a JF764-7 window assembly.

5.5.21 SPECIAL INSPECTION-WHEELS. Inspect wheels at every tire change or annually, whichever occurs first. Inspect wheels and tiebolts in accordance with paragraph 58.5.5. Inspect tiebolt nuts in accordance with B.F. Goodrich Maintenance and Overhaul Manual 667 (or equivalent). Replace defective parts.

5.5.22 3,000 HOUR INSPECTIONS.

Note

The following inspection in step a is not required prior to 6,000 hours.

a. Inspect fuselage inner cap and channel at F.S. 589.5 in accordance with paragraph 58.4.9.

Note

The following inspection in step b is not required prior to 12,000 hours.

b. Inspect wing panel to mid beam lower cap spanwise splice, BB Sta. 55 to 187 in accordance with paragraph 58.2.13. This inspection should be coordinated with the inspections in paragraph 5.5.8.

Note

The following inspection in step c is not required prior to 20,000 hours.

c. Inspect horizontal stabilizer center section aft lower attachment in accordance with paragraph 58.3.8.

Note

The following inspection in step d is not required prior to 4,000 hours.

d. Inspect wing upper and lower panel to root rib attachment in accordance with paragraph 58.2.15. This inspection should be coordinated with the inspections in paragraph 5.5.8.

5.5.23 6,000 HOUR INSPECTIONS.

Note

The following inspections in steps a, b, c and d are not required prior to 12,000 hours.

a. Inspect wing panel to rear beam lower cap spanwise splice in accordance with paragraph 58.2.12. This inspection should be coordinated with the inspecttions in paragraph 5.5.8.

b. Inspect wing lower panel drain holes in accordance with paragraph 58.2.14. This inspection should be coordinated with the inspections in paragraph 5.5.8.

c. Inspect wing lower surface spanwise splices in accordance with paragraph 58.2.11.

d. Inspect frame outer cap at aft engine nacelle mount, F.S. 608, in accordance with paragraph 58.4.11.

Note

The following inspections in steps e, f, g and h are not required prior to 24,000 hours.

e. Inspect fuselage main frame to pressure skin attachment in accordance with paragraph 58.4.10.

f. Inspect emergency exit cutout upper corners in accordance with paragraph 58.4.12.

g. Inspect upper 30-degree longeron attachment, F.S. 483 to F.S. 496, in accordance with paragraph 58.4.13.

h. Inspect top longeron splice in accordance with paragraph 58.4.14.

5.5.24 2,500 HOUR, 5 YEARS MAXIMUM, INSPECTIONS.

a. Remove leading edge, and perform structural inspection of the horizontal stabilizer front beam.

b. Remove leading edge, and perform structural inspection of the vertical stabilizer front beam.

SECTION 5.6

OVERHAUL AND REPLACEMENT SCHEDULE

Contents	Page
GENERAL	5-23
AIR CONDITIONING AND PRESSURIZATION	5-24
AUTOPILOT	5-25
ELECTRICAL	5-26
FLIGHT CONTROLS	5-27
HYDRAULICS	5-28
FUEL SYSTEM	5-30
ICE AND RAIN PROTECTION	5-30
LANDING GEAR	5-32
OXYGEN	5-33
POWER PLANT	5-34
FIRE PROTECTION AND EXTINGUISHING	5-35
CABIN WINDOWS	5-36

5.6.1 GENERAL. This schedule lists certain functional and structural components together with the manufacturer's recommended TimeBetween Overhaul (T.B.O.). All the parts listed fall into one of two categories. Parts not listed may be considered "On Condition" items.

Note

> Superseded Lockheed parts will maintain the same T.B.O. For superseded vendor parts and parts not installed by Lockheed, consult with the part manufacturer or holder of the STC (Supplemental Type Certificate) for T.B.O intervals.

CATEGORY A. Compliance with the recommendations will normally assure trouble free operation of components, failure of which could result in severe crew or passenger discomfort or unsafe flight operation.

CATEGORY B. These parts can be treated as "On Condition" items; however, in the interest of preventive maintenance, observance of the recommended T.B.O. will help reduce unscheduled down time, prolong component life and alleviate premature replacement. Environmental conditions, particularly high humidity, can significantly affect the recommended T.B.O. Equally, an operator's particular maintenance procedures, route structure etc., could, upon negotiation with the component manufacturer through Lockheed, extend the T.B.O. time of items in this category.

ON CONDITION. Parts for which preventive maintenance is not applicable or are not economical to repair.

T.B.O. is given in flight-hours or calendar time. Where both are given, the first reached by the operator is recommended. Calendar time should be measured from the time the component/unit is first placed in service.

5.6.2 AIR CONDITIONING AND PRESSURIZATION

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Bleed Air Flow Control Valve	Garrett Pneu-Tech	106756 31013	3000 Hrs.		B
Cabin Press. Safety Valve	Garrett	103112-930		△16	B
Electronic Cooling Fan and Regulator Assy.	Janitrol	A71C09 B71C09 C71C09 D71C09 E71C09 F71C09	4000 Hrs. or 6 Yrs.		B
Turbine Fan Assy.	Hamilton Standard	548395 726638-1	5000 Hrs.	△17	B
Flight Station Air Cond. Shut-Off Valve	Barber-Colman	PYLB 7368	2500 Hrs. or 5 Yrs.		B
Emerg. Press. Shut-Off Valve	Barber-Colman	BYLB 8453	4000 Hrs.		B
Emerg. Press. Actuator	Garrett	540112-1	4000 Hrs.		B
Engine Bleed Air Shut-Off Valve	Janitrol	B70C19 CA70C19 D70C19	3000 Hrs.		B
Cabin Press. Outflow Valve	Garrett	102358 102358-3		△16	B
Cabin Press. Needle Valve	Garrett	131004-3	4000 Hrs. or 6 Yrs.		B
High Temp. Check Valve 1.00 Dia.	DeLaval	M522		△1	A
Check Valve 1.50 Dia.	DeLaval	M934		△1	A
Check Valve 2.00 Dia.	DeLaval	M921		△1	A
High Temperature Check Valve 2.00 Dia.	DeLaval	M942-1		△1	A
Check Valve 2.75 Dia.	DeLaval	M979 9790-2-1		△1	A
Check Valve 3.00 Dia.	DeLaval	M947		△1	A
Check Valve 3.50 Dia.	DeLaval	M928		△1	A

NOTES: △1 REPLACE EVERY 5000 HOURS.

△16 INSPECT ANNUALLY, REPLACE AS REQ'D (REF. PARA. 5.4.8 f).

△17 AT 300 HR INSPECTION CHECK RESERVIOR FOR WATER CONTAMINATION. REMOVE WATER AS REQ'D AND REPLACE WITH OIL (REF. PARA. 5.3.9 v).

5.6.3 AUTOPILOT

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Vertical Gyro	Sperry	1782356-111 2586137-1	2500 Hrs. or 2 Yrs.	△18	A
Altitude Control	Sperry	620399-1	2500 Hrs. or 2 Yrs.	△18	B
Aileron Servo Drive	Sperry	2585177-21 615743-21	2500 Hrs. or 2 Yrs.	△18	B
Elevator and Rudder Servo Drive	Sperry	2585177-22 615743-22	2500 Hrs. or 2 Yrs.	△18	B
Flight Control Computer without Doppler	Sperry	1781218-21	2500 Hrs. or 2 Yrs.	△18	B
Flight Control Computer with Doppler	Sperry	1781218-24	2500 Hrs. or 2 Yrs.	△18	B
Flight Controller without Doppler	Sperry	1781216-13 1781216-131	2500 Hrs. or 2 Yrs.	△18	B
Flight Controller with Doppler	Sperry	1781216-23 1781216-231	2500 Hrs. or 2 Yrs.	△18	B
Directional Gyro	Sperry	613260-2	2500 Hrs. or 2 Yrs.	△18	A
Servo Bracket	Sperry	615144-011	5000 Hrs.	△18	B

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Vertical Gyro	Collins	522-3985-001 332D-11	1000 Hrs.	△18	A
Yaw Rate Gyro	Collins	229-0236-000 332G-2	5000 Hrs.	△18	A
Pitch and Roll Rate Sensor	Collins	522-3851-000 345A-4B	3000 Hrs.	△18	A
Servo Drive	Collins	772-5374-001 334C-3	3000 Hrs.	△18	B
Servo Mount	Collins	522-1226-000 351B-3	5000 Hrs.	△18	B

△18 Category II Autopilot Systems.
Aircraft certified by the FAA for Category II lowered minimums must comply with the FAA regulations with respect to equipment maintenance and overhaul. These systems are not covered by the T.B.O. information contained in this HOMI.

5.6.4 ELECTRICAL

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Rotary Inverters	A. M. F.	MGE-23-34 MGE-23-340	1000 Hrs.		B
DC Starter Generator	**Lear Siegler**	**JR 1041-3 JR 1041-4 JR 1041-5**	**1000 Hrs.**		**B**
	Bendix	**JR 1041-5**			
Stabilizer Trim Contactors-Normal System	**Hartman**	A-723LD A-723N A-723N-2 A-723N-4	1000 Hrs.		A
Stabilizer Trim Contactors-Emergency System	**Hartman**	A-723P A-723P-1 A-723P-2	3000 Hrs.		A
Auxiliary Hydraulic Pump Relay	Leach	7401-22M MS24185D2	5000 Hrs.	30	A

NOTES: 30 REPLACE WITH A NEW UNIT. OVERHAUL NOT RECOMMENDED.

5.6.5 FLIGHT CONTROLS

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
L.E. Flap Hyd. Motor	Vickers	MF008B-008A MF36-3909-11Z-S595-3	5000 Hrs.		B
T.E. Flap Hyd. Motor	Vickers	MF016B-006A MF36-3909-20Z-5595-3	5000 Hrs.		B
L.E. & T.E. Flap Hyd. Stop	Bertea	53500-301 53500-303 53500-305	5000 Hrs.		B
	Sterer	56620			
L.E. & T.E. Flap Gear Box	Varo	965-001-6	5000 Hrs.	△26	B
	Calco	8201-2M1		△27	B
L.E. Flap Screw Jack Actuator	Southwestern	LA1405 LA2405 LA3405 LA410 LA410A LA4405	4000 Hrs. or 8 Years		A
T.E. Flap Screw Jack Actuator	Southwestern	LA1404 LA2404 LA3404 LA4404 LA500	4000 Hrs. or 8 Years		A
120° L.E. Flap Drive Gear Box	Southwestern	GB36-1	4000 Hrs. or 8 Years		B
Pitch Trim Actuator	Sargent	1506EA-9 -10 -12	4000 Hrs.		A
Pitch Trim Actuator Primary Motor	Sargent	1506D183	2000 Hrs.		A
Aileron Booster Assembly	Natl. Waterlift	JSM190-8	5000 Hrs.		B
Elevator Booster Assembly	Natl. Waterlift	JSM170-9	5000 Hrs.		B
L.E. Flap Drive Coupling	Fourdee	FCB50	2000 Hrs. or 4 Years		A
		JW665-1			

NOTES: △2 NOTE DELETED.

△26 LUBRICATE WITH MIL-L-6086 OIL AT OVERHAUL.

△27 LUBRICATE WITH MIL-G-21164 GREASE AT OVERHAUL.

Use and/or disclosure is governed by the statement on the title page of this document.

5.6.6 HYDRAULICS

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Engine Driven Hyd. Pump	New York Air Brake	66WBG300 66WBG300-1 66WBG300-2 66WBG300-3 66WBG300-4	At Engine T.B.O.		B
Electric Motor Driven Hydraulic Pump	Pesco	113992-010-01 113992-010-02	5 Yrs.	△3	B
	New York Air Brake	165WE01003 165WE01003-1			
Air Press. Regulator	Textron	105800-3	36 Mths.		B
Wing Flap Selector Valve	Parker-Hannifin	H61G0109 2630142	4 Yrs.		B
	Weston	18890 18890-2			B B
Main Landing Gear Door Selector Valve	Adel	51580-12 56944	4 Yrs.		B
	Parker-Hannifin	263-0065 263-0066 263-0045			B
Brake Selector Valve	Sterer	52760	4 Yrs.		B
	Adel	55204-8			
	Parker-Hannifin	1371-598850			
Nose Wheel Steering and Standby Flap Selector Valve	Parker-Hannifin	1371-579217M1 1371-579217M2	5 Yrs.		B
Steering Selector Valve	Adel	52204-81	4 Yrs.		B
	Parker-Hannifin	1371-598820			
Hyd. Systems Press. & Return Line Filter	Aircraft Porous Media	AC-2768-10		△4	A
Hyd. Systems Return Line Filter	Aircraft Porous Media	AC-2768-101P AC-2768-101PY2		△5	A
Hyd. Systems Press. Line Filter	Aircraft Porous Media	AC-2768-101NBP AC-2768-101NBPY2		△5	A
Brake System Shuttle Valve	Sterer	17610 17610-1	2500 Hrs. or 5 Yrs.		B

NOTES: △3 EXCESSIVE GROUND RUNNING WILL REDUCE THIS TIME CONSIDERABLY

△4 CLEAN OR REPLACE FILTER ELEMENT EVERY 600 HOURS

△5 CLEAN OR REPLACE FILTER ELEMENT WHEN DIFFERENTIAL PRESSURE IS INDICATED OR AT 2000 HRS. WHICHEVER COMES FIRST. PERFORM FUNCTIONAL TEST OF DIFFERENTIAL PRESSURE INDICATOR IN ACCORDANCE WITH INSTRUCTIONS IN VENDOR'S OVERHAUL MANUAL.

5.6.6 HYDRAULICS (Cont)

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Landing Gear System Isolation Valve	Parker-Hannifin	H61E0258	6 Yrs.		B
Dual Shuttle Valve	Crane	8035B	3000 Hrs. or 6 Yrs.		B
	Sterer	58740			
Manually Controllable Check Valve	Parker-Hannifin	14-354-51 14-354-251	6 Yrs.		B
	Allen Aircraft	4-C160-1 4-C160-2			
5 Cu. In. Accumulator	Parker-Hannifin	1356-593105	5 Yrs.	28	A
25 Cu. In. Accumulator	Parker-Hannifin	1356-583321	5 Yrs.	28	A
	Teledyne	60869-25			
50 Cu. In. Accumulator	Parker-Hannifin	1356-583323	5 Yrs.	28	A
	Teledyne	60868-3			
150 Cu. In. Accum.	Parker-Hannifin	1356-593769	5 Yrs.	28	A
	Teledyne	60868-6			
Emergency Extension Main L.G. Air Bottle Assy.	Tavco	23111108			
	Futurecraft	90178-2	5 Yrs.	25	A
Emergency Extension Nose L.G. Air Bottle Assy.	Tavco	23111111	5 Yrs.	25	A
	Futurecraft	90406-1			
Emergency Extension Nose L.G. Air Bottle Charging & Relief Valve	Tavco	23111119	5 Yrs.		A
	Futurecraft	90409			
Emergency Extension Nose L.G. Air Bottle Assy.	Tavco	23111188 23111188-1	5 Yrs.	25	A
	Tactair	9058-40-2			
Emergency Extension Main L.G. Air Bottle Assy.	Tavco	23111108-1 23111108-2	5 Yrs.	25	A
	Tactair	9058-40-1			
Emergency Extension Nose L.G. Air Bottle Assy.	Tavco	23111111-1	5 Yrs.	25	A

NOTES: 25 HYDROSTATIC TEST TO 4440 P.S.I.G. IS DUE EVERY 5 YEARS PER FEDERAL REGULATION, TITLE 49, PARAGRAPH 173.34 FOR 3A CYLINDER. IF CONTAINER IS CHARGED WHEN TEST IS DUE, THE TEST MAY BE DEFERRED UNTIL CONTAINER IS RECHARGED OR UNTIL THE NEXT ANNUAL INSPECTION, WHICHEVER OCCURS FIRST.

28 OVERHAUL WILL INCLUDE NDE INSPECTION PER PARAGRAPH 58.4.7.

5.6.7 FUEL SYSTEM

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Main Fuel Boost Pump	Lear Siegler	RR12040B RR12040E	300 Hrs.	24	A
		RR12040D RR12040F	2000 Hrs.		
Motor-Operated Fuel Shut-Off Valve	General Controls	AV-16B-1668C AV-16B-1669D	5000 Hrs.		B
	Whittaker	134605 134595			
External Tank Fuel Boost Pump	Lear Siegler	RR12150C RR12150D	2000 Hrs.		B
Jettison Line Flex. Hose	Lockheed	JP 806-2		25 6	A

NOTES:

6 REPLACE EVERY THREE YEARS OR WHENEVER THE AIRPLANE IS EXPOSED TO A COLD SOAK, ON THE GROUND, AT A AMBIENT TEMPERATURE OF -40°F OR LOWER FOR AT LEAST SIX HOURS.

24 THESE TIMES ARE AS SPECIFIED BY THE CERTIFICATE DATA SHEET NO. 2A15 WHICH FORMS PART OF TYPE CERTIFICATE NO. 2A15.

25 JP806-501 IS PREFERRED SPARE AND IS AN "ON CONDITION" ITEM.

5.6.8 ICE AND RAIN PROTECTION

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
De-icer Supply Press. Regulator and Safety Valve	Janitrol	E32C84	2500 Hrs.		B

5.6.9 Deleted.

5.6.10 LANDING GEAR

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Nose Landing Gear Steering Actuator	Menasco	JL1500-3 JL1500-20 JL1501-4	3 Yrs.	△20	A
Nose Wheel Steering Valve	Consolidated Controls	141V9-8	3 Yrs.		A
	Bertea	50100-303	6000 Hrs.		B
Nose Landing Gear Shock Strut	Menasco	JL 1600-11	13000 Hrs.		A
Nose Landing Gear Retract Actuator	Menasco	JL-1900-3	13000 Hrs.		A
Main Landing Gear Shock Strut	Menasco	JL 1100-5 JL 1100-7	13000 Hrs.		A
Main Landing Gear Side Actuator	Rovalco	JL 1400-13	10000 Hrs.		
	Royal Industries	JL 1400-13	13000 Hrs.	△19	A
	Menasco	JL1400-9	10000 Hrs.		A
		JL1400-11	13000 Hrs.	△19	
		JL1400-7	13000 Hrs.		
			6000 Hrs.		

NOTES: △19 IF ACTUATORS INCORPORATE STEEL BARRELS.

△20 REFER TO SERVICE BULLETIN 329-300B.

5.6.11 OXYGEN

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Pressure Reducer and Cylinder Assy	Carleton Life Support Technologies, Ltd. 1-1200 Aerowood Drive L4W 257 Mississauga, Ontario, Canada 1-905-629-3245	F5038350-7 F5038350-9 F5038350-11 F5038350-17 F5038350-19 F5038350-21	2 Yrs.	7 8	A
Portable Oxygen Cylinder	Litton, Clifton Precision	29612-A1 29612-B1 29612-C1 29612-D1	3 Yrs.	9	A
	Robertshaw	900-002-035			
	Sabre Industries	900-002-035			
Crew Regulator	Carleton Life Support Technologies, Inc.	15830-3 15830-4	2 Yrs.	11	A
	Litton, Clifton Precision	29274-13A-A1 29274-13A-A2	3 Yrs.		
Passenger Control Panel	American Safety Flight Systems	7300-3 7300-5	2 Yrs.		A
	Litton, Clifton Precision	GU-313-B1 GU-313-B3			
Door Actuator Valve Passenger Mask • Container	Carleton Life Support Technologies, Inc.	61582-10	3 Yrs.	10	A
	Puritan-Bennett Aero Systems	170110			
Lanyard Valve - Passenger Mask	Carleton Life Support Technologies, Inc.	23809-30	3 Yrs.	10	A
	Puritan-Bennett Aero Systems	170059			
Passenger Mask and Regulator	Carleton Life SupportTechnologies,	F6022200-3	3 Yrs.		A

NOTES:

7 OVERHAUL MUST INCLUDE REPLACEMENT OF HIGH PRESSURE BURST DISC.

8 HYDRO STATIC TEST TO 3085 PSIG IS DUE EVERY 3 YEARS PER FEDERAL REGULATION, TITLE 49, PARAGRAPH 173.34 FOR 3HT CYLINDERS. IF CONTAINER IS CHARGED WHEN TEST IS DUE, THE TEST MAY BE DEFERRED UNTIL CONTAINER IS RECHARGED OR UNTIL THE' NEXT ANNUAL INSPECTION, WHICHEVER OCCURS FIRST. MAXIMUM LIFE OF CYLINDER IS 24 YEARS.

9 HYDRO STATIC TEST TO 3335 PSIG IS DUE EVERY 5 YEARS PER FEDERAL REGULATION, TITLE 49, PARAGRAPH 173.34 FOR 3AA CYLINDERS. IF CONTAINER IS CHARGED WHEN TEST IS DUE, THE TEST MAY BE DEFERRED UNTIL CONTAINER IS RECHARGED OR UNTIL THE NEXT ANNUAL INSPECTION, WHICHEVER OCCURS FIRST.

10 THIS ITEM OVERHAULED BY PURITAN-BENNETT AERO SYSTEMS, 111 PENN. ST., EL SEGUNDO, CA 90245, PHONE 213-772-1421.

11 IT IS NOT RECOMMENDED THAT THIS COMPONENT BE OVERHAULED IN THE FIELD. A ZERO TIME EXCHANGE PROGRAM IS AVAILABLE FROM THE MANUFACTURER. CONTACT THE CARLETON LIFE SUPPORT TECHNOLOGIES, INC., 1-1200 AEROWOOD DR., MISSISSAUGA, ONTARIO, CANADA L4W 257, PHONE 1-905-625-3245.

5.6.12 POWER PLANT

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT
Engine	Pratt & Whitney	JT12A-6A	1800 Hrs.	△15 △12 △26 △27	A
Engine	Pratt & Whitney	JT12A-6A	3000 Hrs.	△15 △12 △27 △21 △22 △26	A
Engine	Pratt & Whitney	JT12A-8	3000 Hrs.	△15 △27 △22 △26	A
Thrust Reverser Assemblies (including Actuators) L.H. Outboard R.H. Outboard L.H. Inboard R.H. Inboard	Rohr Aircraft Corporation	140-5300-( ) All dash nos. (including Actuators)	2400 Hrs.	△13	B

NOTES: △12 T.B.O. OF 1800 HOURS APPLIES AFTER ALL THE FOLLOWING PRATT & WHITNEY REQUIREMENTS HAVE BEEN INCORPORATED:

CAMPAIGN CHANGE	PRATT & WHITNEY SERVICE BULLETIN	
63-21	478	HASTELLOY TRANSITION DUCT
65-5	921	BLEED STRAP CONTAINMENT RING
65-17	993	SECOND STAGE STATOR VANES
65-18	751	SECOND STAGE TURBINE BLADES
65-21	-	UPPER TOWER SHAFT BEARING
65-36	884	COMPRESSOR BLADE TABLOCKS

△13 OVERHAUL OF THRUST REVERSERS IS TO BE ACCOMPLISHED AS DESCRIBED IN CHAPTER 78.

△15 REFER TO PRATT & WHITNEY JT12A OVERHAUL MANUAL FOR THE CYCLE LIMITS OF COMPRESSOR DISKS. SOME AIRCRAFT ENGAGED IN NUMEROUS FLIGHTS OF SHORT DURATION MAY REQUIRE EARLY OVERHAUL BECAUSE OF THESE LIMITS. SEE PRATT & WHITNEY SERVICE BULLETIN NO. 1508 ALSO.

△21 PROVIDED THE FOLLOWING ARE ACCOMPLISHED IN ADDITION TO △12 . PRATT & WHITNEY SERVICE BULLETIN NO. 1490 (REPLACEMENT OF COMBUSTION CHAMBER AND DUCT) AND PRATT & WHITNEY SERVICE BULLETIN NO. 1584 (REMOVAL OR COMBUSTION CHAMBER AIR SCOOP).

△22 PRATT & WHITNEY SERVICE BULLETIN 2005 (TOWER SHAFT BEARING REPLACEMENT) MUST BE INCORPORATED NO LATER THAN 2200 HOURS.

△26 PRATT & WHITNEY SERVICE BULLETIN 3409 (FUEL MANIFOLD, NOZZLE, AND COMBUSTION CHAMBER REWORK) SHOULD BE INCORPORATED ANYTIME SIGNIFICANT FIRST STAGE TURBINE BLADE EROSION IS OBSERVED. INCORPORATION OF COMBUSTION CHAMBER REWORK IN PRATT & WHITNEY SERVICE BULLETINS 4414 AND 4418 IS ALSO RECOMMENDED.

△27 CALENDAR TBO IS SEVEN YEARS (NOT INCLUDING TIME UNDER PROPER PRESERVATION) IN ACCORDANCE WITH PRATT AND WHITNEY SERVICE BULLETIN 1508.

5.6.13 FIRE DETECTION AND EXTINGUISHING

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Fire Extinguishing Container Assy	Accessory Products	802900-1		△10	A
	HTL Advanced Tech	2241-005			
	Walter Kidde	891471-01 891471-02			
Explosive Actuator	Accessory Products	800545-2		△14	A
	HTL Advanced Tech	30903835			
	Walter Kidde	872627-01			

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Fire Extinguisher, Carbon Dioxide, Portable	Various	AN6045-147	—	△20	A

NOTES: △10 HYDRO STATIC TEST TO 1800 PSIG IS DUE EVERY 5 YEARS PER FEDERAL REGULATION, TITLE 49, PARAGRAPH 173.34 FOR 4DA CYLINDERS. IF CONTAINER IS CHARGED WHEN TEST IS DUE, THE TEST MAY BE DEFERRED UNTIL CONTAINER IS RECHARGED OR UNTIL THE NEXT ANNUAL INSPECTION, WHICHEVER OCCURS FIRST.

△11 NOTE DELETED.

△14 REPLACE EVERY 6 YEARS AFTER MANUFACTURE OR 4 YEARS AFTER INSTALLATION, WHICHEVER COMES FIRST.

△20 HYDRO STATIC TEST TO 3500 PSIG IS DUE EVERY 5 YEARS PER FEDERAL REGULATION, TITLE 49, PARAGRAPH 173.34 FOR 3AA CYLINDER. IF CONTAINER IS CHARGED WHEN TEST IS DUE, IT SHOULD BE DISCHARGED AND TESTED.

5.6.14 CABIN WINDOWS.

PART NAME	MANUFACTURER	PART NUMBER	T.B.O.	NOTES	CAT.
Cabin Window	PPG Ind.	JF442-1 -7 -8 JF764-1 -501 -7 △29		△23	A

NOTES: △23 INSPECT EVERY 5 YEARS, REFER TO PARAGRAPH 5.5.20.

△29 JF764-7 IS THE PREFERRED REPLACEMENT PART FOR ALL AIRCRAFT.

SECTION 5.7
CONTINUED AIRWORTHINESS

Contents Page

GENERAL 5-37

100/300-HOUR AND ANNUAL INSPECTIONS 5-37

SPECIAL INSPECTIONS 5-37

AIRWORTHINESS DIRECTIVES 5-37

INFORMATION 5-37

LIFE LIMITED COMPONENTS 5-38

5.7.1 GENERAL. Service experience has shown that aging airplanes need more care and special attention during the maintenance processes. At times, more frequent inspection of structural components is required. Inspection programs in this publication have been prepared to meet the recommendations of FAA Advisory Circular 91-60.

5.7.2 100/300 HOUR and ANNUAL INSPECTIONS. The inspections in Section 5.3 and Section 5.4 have been prepared to provide safety of operation. These inspections will be revised as new situations arise. Overhaul and replacement periods in Section 5.6 and the lubrication/servicing recommendations in Chapter 12 should be maintained and insofar as possible should be incorporated in the normal scheduled inspections.

5.7.3 SPECIAL INSPECTIONS. Some of the inspections in Section 5.5 are provided for special situations and conditions. The remaining inspections are for the purpose of providing continued airworthiness for aging airplanes and generally meet requirements of the Supplemental Inspection Document (SID) described in FAA Advisory Circular 91-56. AC91-56 is not presently a requirement for JetStar airplanes; however, the concept of a SID is useful to operators and therefore, the information has been provided by the Lockheed Martin Aircraft & Logistics Centers.

5.7.4 AIRWORTHINESS DIRECTIVES. FAA Airworthiness Directives (AD) should be complied with promptly. Structural improvements are usually directed to lowering working stresses, and the sooner they are made, the better they are to slow the onset of cumulative damage to the original structure. In some instances, this publication may require more frequent inspections than are specified in a particular AD. In such cases, the more frequent inspections are necessary for continued airworthiness.

5.7.5 INFORMATION. A free exchange of data and sharing of operational experience is required for continuing success of the continued airworthiness program for JetStar airplanes. To report peculiar operational conditions, see Technical Information, page Introduction-3.

Lockheed Martin publications will be updated when the need is indicated by experience and reports from operators.

5.7.6 LIFE LIMITED COMPONENTS

FAA Airworthiness Limitations Instructions

Life Limited Components

Part Number	Nomenclature	Qty	(Discard)	HOMI Sec Ref
	Chapter 32 – Landing Gear			
JL1955-3	Cylinder, Nose Landing Gear Steering	1	Discard Use JL1955-7	32.4
JL1955-7	Cylinder, Nose Landing Gear Steering	1	3600 Flights	32.4
	Chapter 55 – Stabilizers			
JE24-2	Fitting, Empennage Pivot	1	Discard Use JE24-6	55.1
JE24-4	Fitting, Empennage Pivot	1	Discard Use JE24-6	55.1
JE24-6	Fitting, Empennage Pivot	1	No Life Limit	55.1
	Chapter 57 – Wings			
MS20008-20	Bolt, Wing Attach, Upper	4	1900 Hrs	57-1
MS20010-22	Bolt, Wing Attach, Upper			57-1
	(AC w/o SB329-157)	6	1900 Hrs	
	(AC w/SB329-157)	4	1900 Hrs	
MS20012-28	Bolt, Wing Attach, Lower	4	1900 Hrs	57-1
MS20010-22	Bolt, Wing Attach, Lower	4	1900 Hrs	57-1
MS21250-100022	Bolt, Wing Attach, Lower	2	1900 Hrs	57-1
	(5056-5200 and AC w SB329-157)			
2752-080	Nut, Barrel - Upper	4	1600 Hrs	57-1
2752-108	Nut, Barrel - Upper	6	1600 Hrs	57-1
2752-108	Nut, Barrel - Lower	4	1600 Hrs	57-1
2752-126	Nut, Barrel - Lower	4	1600 Hrs	57-1
NAS577-12AX	Nut, Barrel - Lower	2	1600 Hrs	57-1
JW121-6L	Fitting – Actuator, T.E. Inboard Flap	1	4024 Flights	57-1
JW121-6R	Fitting – Actuator, T.E. Inboard Flap	1	4024 Flights	57-1

5.7.6 LIFE LIMITED COMPONENTS (CONT)

FAA Airworthiness Limitations Instructions

Life Limited Components

Part Number	Nomenclature	Qty	(Discard)	HOMI Sec Ref
Chapter 57 – Wings (Cont)				
JW121-8L	Fitting – Actuator, T.E. Inboard Flap	1	4024 Flights	57-1
JW121-8R	Fitting – Actuator, T.E. Inboard Flap	1	4024 Flights	57-1
JW123-6L	Fitting – Actuator, T.E. Outboard Flap	1	4024 Flights	57-1
JW123-6R	Fitting – Actuator, T.E. Outboard Flap	1	4024 Flights	57-1
JW123-9L	Fitting – Actuator, T.E. Outboard Flap	1	4024 Flights	57-1
JW123-9R	Fitting – Actuator, T.E. Outboard Flap	1	4024 Flights	57-1
NAS630-24	Bolt, Pylon Attach	8	3125 Flights	57.7

CHAPTER 6
DIMENSIONS AND AREAS

TABLE OF CONTENTS

Contents Page

GENERAL 6-1

FIGURES
6-1. Principal Dimensions 6-2
6-2. Airplane-Three Views 6-4
6-3. Fuselage Station Diagram 6-5
6-4. Wing and Tank Station Diagram 6-6
6-5. Horizontal Tail Station Diagram 6-7
6-6. Vertical Tail Station Diagram 6-8
6-7. Nacelle Station Diagram 6-9

6.1 GENERAL. This chapter includes charts and tables which show areas, dimensions and locations of the major structural members of the aircraft. All linear dimensions are in inches unless otherwise specified. Dimensions given are with the airplane in the normal ground attitude.

GENERAL

Span, Maximum	54 feet 5 inches
Length, Overall	60 feet 5 inches
Height	20 feet 5 inches
Minimum Ground Clearance (Approximate)	
Speed Brake Retracted	2 feet 6 inches
Speed Brake Extended	6 inches
Wing Tip	3 feet 6 inches

WINGS

Type	Low
Airfoil Section and Thickness (Percent of Chord)	
At Root (W.S.41.5) NACA63A112	12
At Construction Tip NACA63A309 (MOD)	8.34
Chord	
At Root (W.S. 42.5)	163.8 inches
At Construction Tip (W.S. 318)	60.9 inches
Mean Aerodynamic	131.2 inches
Incidence	
At Root (W.S. 41.5)	+1 degree
At Construction Tip	-1 degree
Dihedral (At Wing Reference Plane)	2 degrees
Sweep at 25 percent Chord	30 degrees
Aspect Ratio	5.27

HORIZONTAL TAIL

Span	24 feet 9 inches
Chord	
At Root	110 inches
At Construction Tip (B.L. 147)	36 inches
Mean Aerodynamic	79.3 inches
Incidence (From 0 degrees Incidence)	1 degree up, 9 degrees down
Dihedral	0 degrees
Sweep at 25 percent Chord	30 degrees
Aspect Ratio	4.03

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Figure 6-1. Principal Dimensions (Sheet 1 of 2)

HORIZONTAL TAIL (Cont)

Airfoil Section and Thickness (Percent of Chord)	
At Root NACA 63A010	10
At Construction Tip NACA 63A008	8

VERTICAL TAIL

Span	149 inches
Chord	
At Root (W.L. 118)	155 inches
At Construction Tip (W.L. 267)	58 inches
Mean Aerodynamic	113.8 inches
Airfoil Section and Thickness (Percent of Chord)	
At Root NACA 63A009	9
At Construction Tip NACA 63A005	5
Sweep at 25 Percent Chord (Horizontal Stabilizer Incidence -1 degree)	30 degrees
Aspect Radio	1.4

FUSELAGE

Length	58 feet 9.6 inches
Diameter (Maximum)	85 inches
Maximum Frontal Area	40.4 square feet

AREAS

Wings, Total (Including Ailerons, Leading Edge Flaps, Trailing Edge Flaps, Tips, and 99 square feet of Fuselage)	542.5 square feet
Ailerons, aft of Hinge Line	24.38 square feet
Flap (Extended)	62.6 square feet
Leading Edge Flap (Forward of Hinge Line)	34.0 square feet
Horizontal Tail, Total	149.04 square feet
Horizontal Stabilizer	117.8 square feet
Elevator (Aft of Hinge Line, Including 0.6 square feet of External Balance Forward of Hinge)	31.2 square feet
Vertical Tail, Total	110.2 square feet
Vertical Stabilizer	94.0 square feet
Rudder (Aft of Hinge Line, Including 0.3 square feet of External Balance Forward of Hinge and 1.93 square feet of Tab)	16.2 square feet
Speed Brake	9.17 square feet

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Figure 6-1. Principal Dimensions (Sheet 2 of 2)

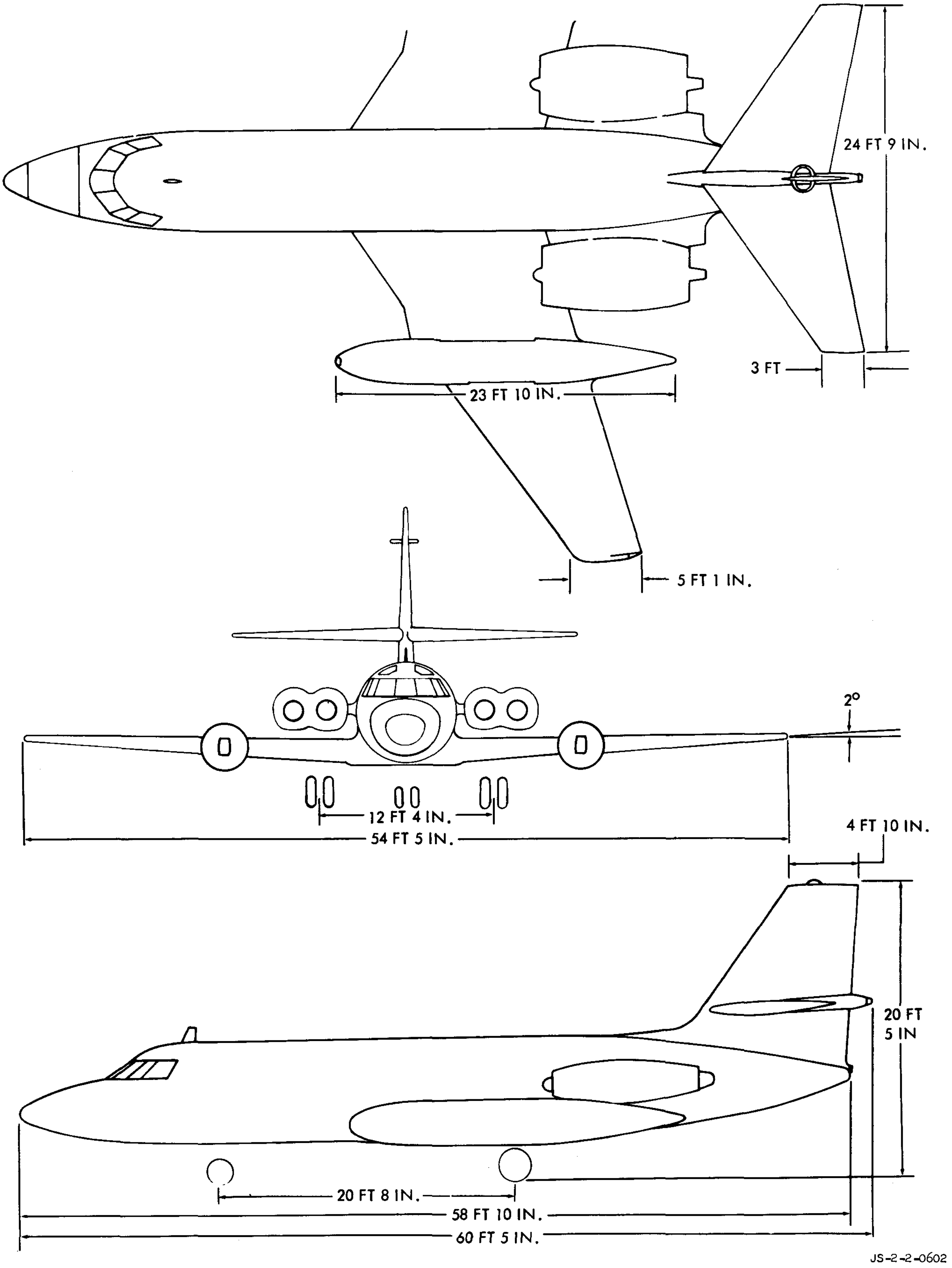


Figure 6-2. Airplane-Three Views

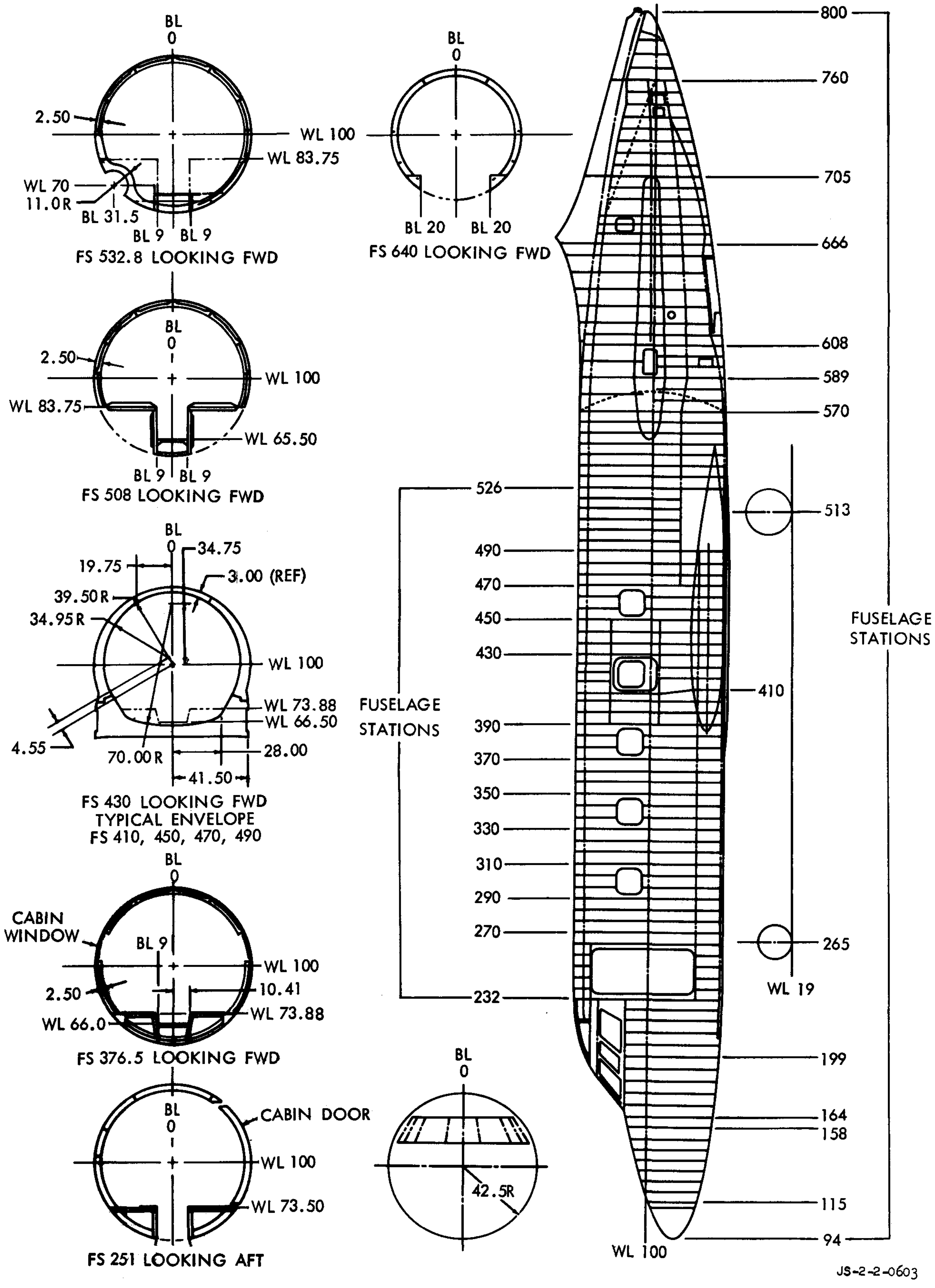
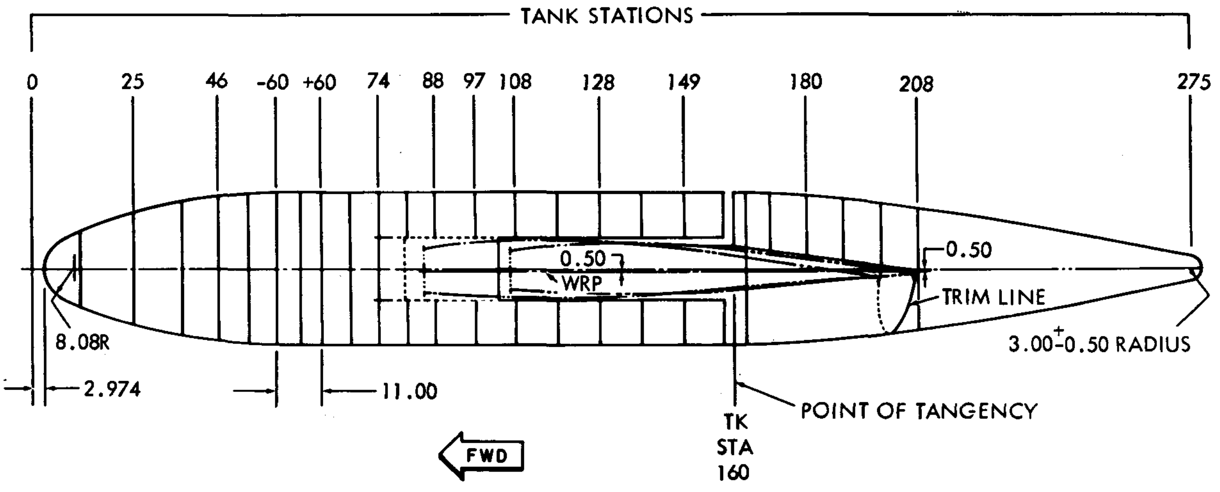
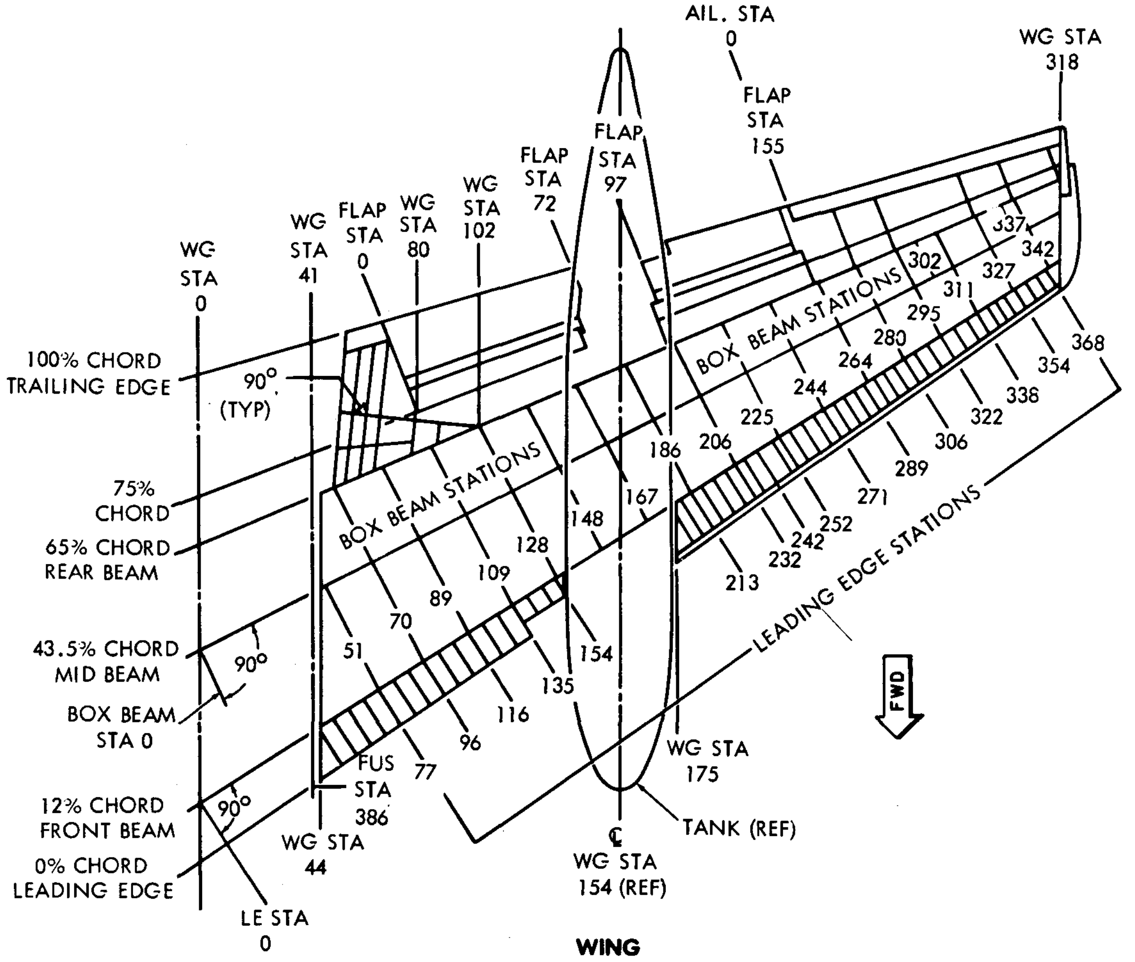


Figure 6-3. Fuselage Station Diagram



WING TANK



WING

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Figure 6-4. Wing and Tank Station Diagram

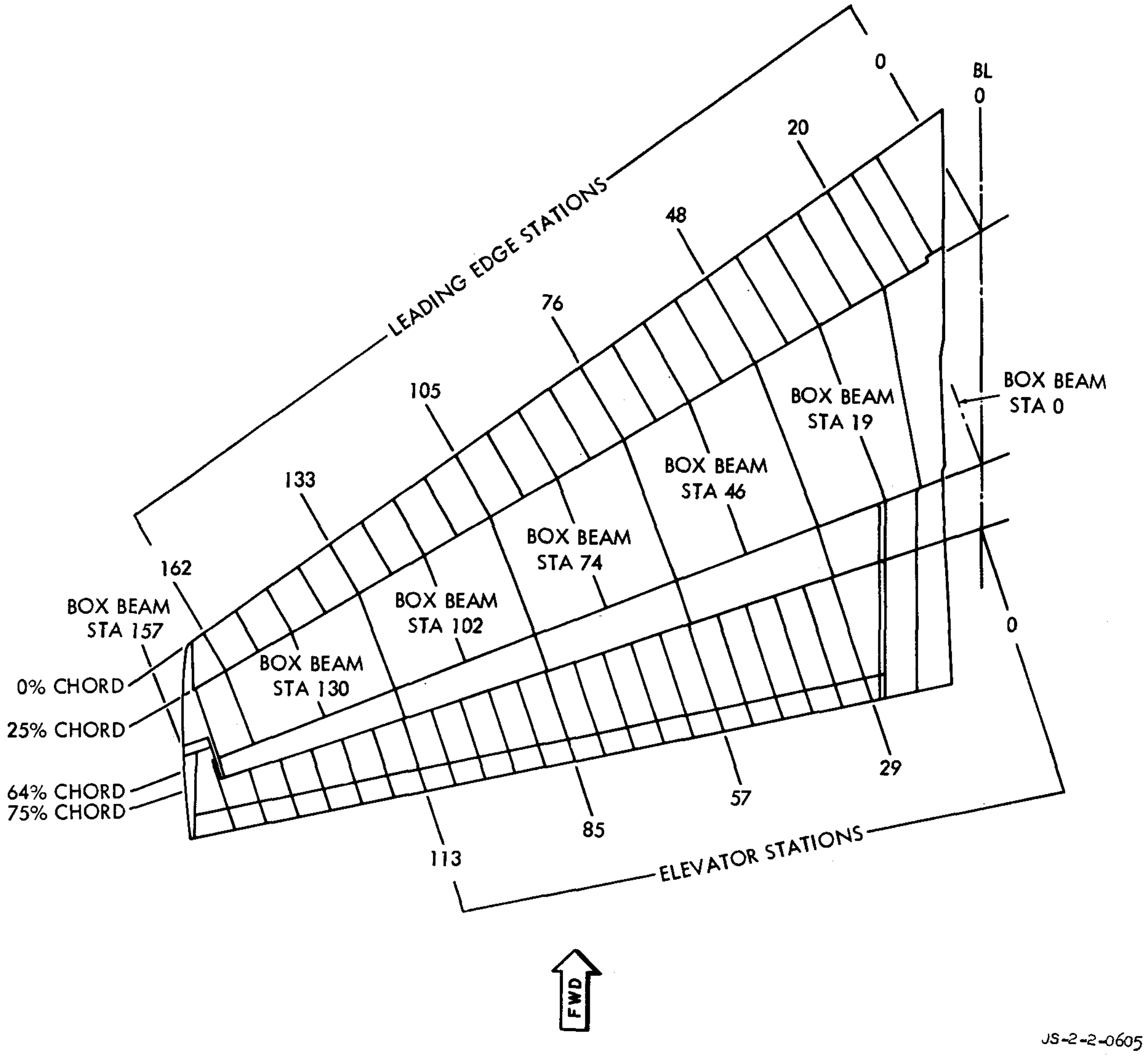

Figure 6-5. Horizontal Tail Station Diagram

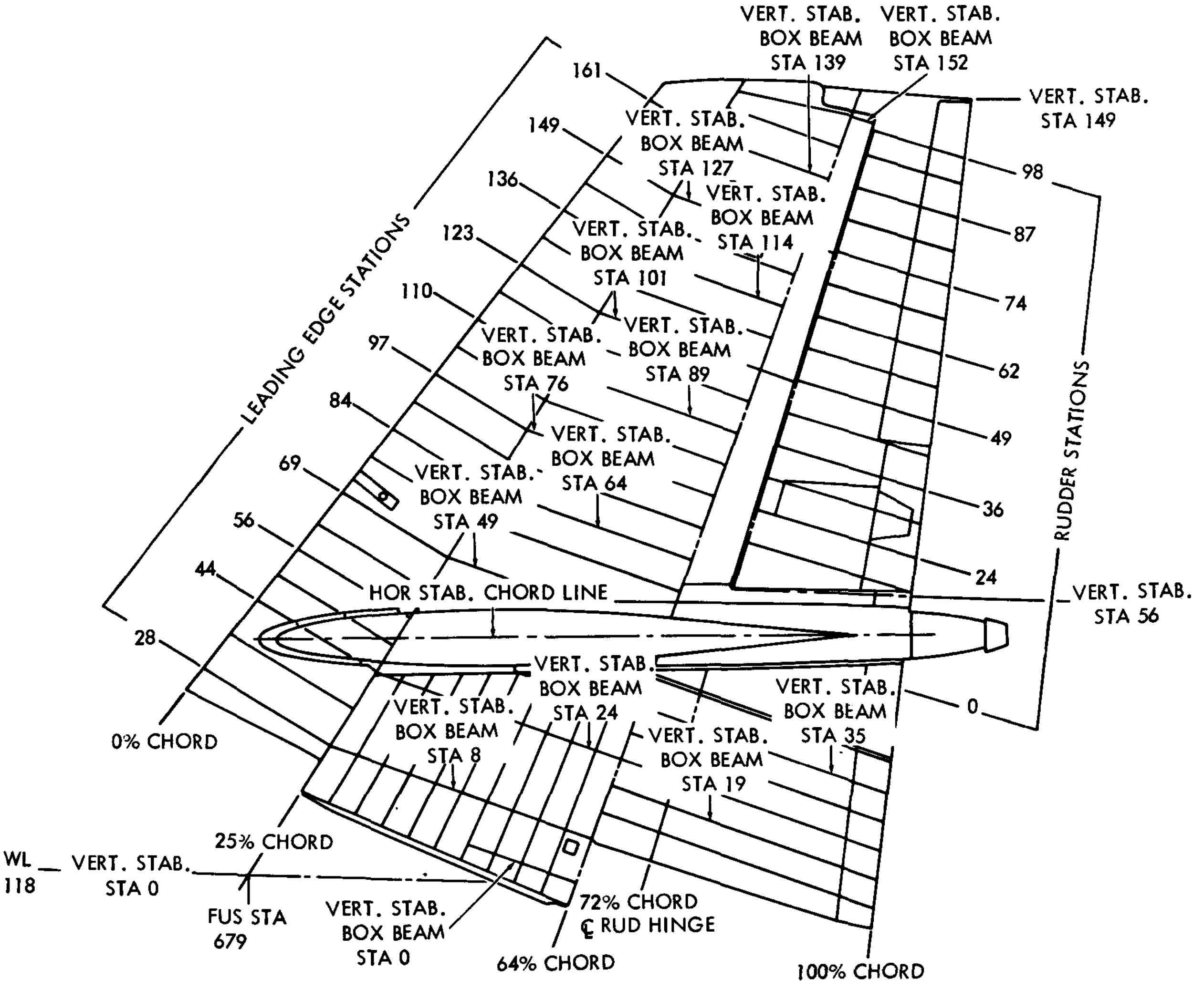
VERT. STAB. BOX BEAM STA 139
VERT. STAB. BOX BEAM STA 152
VERT. STAB. STA 149
161
149
136
123
110
97
84
69
56
44
28
LEADING EDGE STATIONS
VERT. STAB. BOX BEAM STA 127
VERT. STAB. BOX BEAM STA 114
VERT. STAB. BOX BEAM STA 101
VERT. STAB. BOX BEAM STA 89
VERT. STAB. BOX BEAM STA 76
VERT. STAB. BOX BEAM STA 64
VERT. STAB. BOX BEAM STA 49
HOR STAB. CHORD LINE
98
87
74
62
49
36
24
0
RUDDER STATIONS
VERT. STAB. STA 56
VERT. STAB. BOX BEAM STA 24
VERT. STAB. BOX BEAM STA 8
VERT. STAB. BOX BEAM STA 19
VERT. STAB. BOX BEAM STA 35
0% CHORD
25% CHORD
WL 118
VERT. STAB. STA 0
FUS STA 679
VERT. STAB. BOX BEAM STA 0
64% CHORD
72% CHORD
℄ RUD HINGE
100% CHORD
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Figure 6-6. Vertical Tail Station Diagram

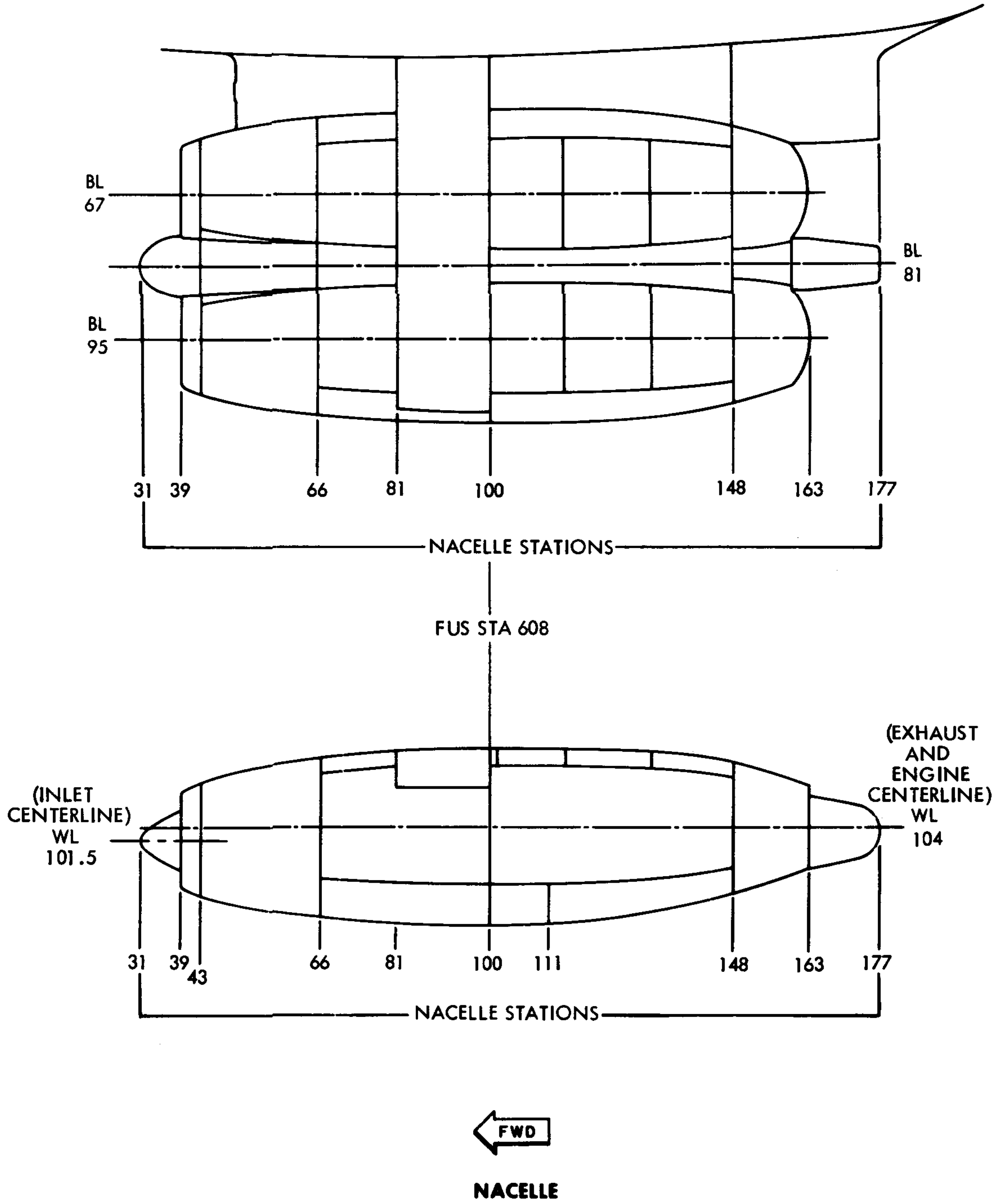

Figure 6-7. Nacelle Station Diagram

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CHAPTER 7
LIFTING AND SHORING

TABLE OF CONTENTS

Section		Page
7.0	LIFTING	7-3
7.1	SHORING	7-11
7.2	METHODS OF HANDLING A DAMAGED AIRPLANE	7-13

SECTION 7.0
LIFTING

Contents Page

JACKING 7-3
Jacking Precautions 7-3
Wheel Change Jacking 7-3
Strut Jacking 7-4
Wing and Fuselage Nose Jacking 7-4
Lowering the Airplane 7-4

HOISTING 7-4

FIGURES
7-1. Landing Gear Jacking Provisions 7-6
7-2. Airplane Center of Gravity Chart 7-7
7-3. Wing and Fuselage Jacking Provisions 7-8
7-4. Hoisting Provisions 7-9
7-5. Major Component Weights 7-10

7.0.1 Several means for jacking the airplane are available. The method used will depend on the maintenance operation to be performed. Most operations can be performed with only one strut jacked, and this procedure should be used whenever possible. Some extensive maintenance procedures, as well as operational checks of the landing gear, will require jacking of the entire airplane. Jacking more than one gear at a time is not recommended because of jacking instability. If the entire airplane must be raised, use the procedures given for wing-and-fuselage-nose jacking instead of trying to jack the individual struts.

7.0.1.1 JACKING PRECAUTIONS.

WARNING

Failure to observe any of these safety precautions may result in damage to the airplane or injury to personnel.

a. Do not jack the airplane outside the hangar if the wind exceeds 10 miles per hour. Head airplane into the wind if possible.

b. Always choose a level place with solid footing to jack.

c. Do not jack the airplane when wing panels are removed or when there is damage between the wing jack points and the fuselage.

d. It is recommended that the airplane not be jacked at the wing jack points if fuel other than residual is in internal tanks 1 and 4. If a jack should slip from the wing jack point and pierce a tank, a dangerous fuel spill could result. This is not a weight limitation.

e. Do not attempt to raise the airplane by inflating the landing gear shock struts. Overinflation of the struts can result in seal failures. Use caution when lowering the airplane. Rapid lowering will endanger the airplane, ground equipment and personnel.

f. Check to see that landing gear downlocks have been installed. (See figures 12-19 and 32-7.)

g. Observe all other precautions relative to ground handling of the airplane as circumstances may require.

h. Install main or nose gear strut locks if strut extension is not desired.

i. Operate the jacks at an even rate to avoid any attitude change which might excessively sideload the jack pads. While jacking, keep the jack locknut screwed down to within one inch of the jack case. When the airplane is jacked to the desired height, screw the locknut down until it seats against the case.

j. If the airplane has been jacked outside the hangar, moor the airplane in the normal manner so that wind gusts will not shift the airplane.

7.0.1.2 WHEEL CHANGE JACKING. The dual wheels of the airplane permit wheel change without jacking the airplane. When only one wheel of the dual set needs to be changed, run the other wheel up an inclined block similar to the one shown in figure 7-1. When both wheels on one gear require replacement, jack the entire strut in accordance with the strut jacking procedures.

7.0.1.3 STRUT JACKING.

WARNING

Do not attempt to raise the airplane by inflating the landing gear shock struts. Overinflation of the struts can result in seal failures. Rapid lowering of the airplane will endanger the airplane, ground equipment and personnel.

Jack points are on the bottom of each gear strut. Any one gear may be safely jacked up to an airplane gross weight of 40,860 pounds, providing the airplane's center of gravity is within permissible flight limits. Strut-to-ground clearances are shown on figure 7-1. Use an axle jack with a minimum capacity of 12 tons for the main gear and a 5-ton jack for the nose gear.

7.0.1.4 WING AND FUSELAGE NOSE JACKING. A jack point is provided on each wing and on the fuselage nose centerline for jacking the entire airplane. Wing and nose jacking is similar to gear jacking except the airplane's weight and center of gravity must be considered before jacking. To be safe for jacking, the center of gravity location must fall within the limits of figure 7-2. With the weight and center of gravity location in this area, the maximum safe load on any jack pad will not be exceeded. Normally, the center of gravity will fall between fuselage stations 470 and 488, since this represents the permissible center-of-gravity range for flight. To be sure the center of gravity is permissible, always compute it for the particular load condition existing at the time of jacking. Refer to the Weight and Balance Manual for instructions on weighing the airplane and computing the center of gravity. Use the following procedure for jacking:

a. Plot the airplane center-of-gravity and weight on the chart in figure 7-2 to be sure it is safe to jack.

b. Position the airplane in the desired jacking area. If outside, head the airplane into the wind.

c. Release the parking brakes.

d. Remove wheel chocks.

e. Disconnect all external power.

f. Remove the cover plugs from the airplane jacking points and install 11011-MEU jack pads at each of the jack points shown on figure 7-3. The pads slip into sockets and are secured in place with one screw for each pad.

g. Install the jacks (10 ton capacity) in position beneath the jacking pads.

Note

The installed pads tilt about 7 degrees from the vertical (the nose pad tilts forward and the wing pads tilt aft). This normal condition does not cause any side-loading but does make it necessary to take care when positioning the jacks, to be sure they are not placed directly under the pads and perfect mating exists between the jack recess and the ball on the jack pad.

h. Install main or nose gear strut locks if strut extension is not desired.

i. Operate the jacks at an even rate to avoid any attitude change which might excessively sideload the jack pads. While jacking, keep the jack locknut screwed down to within one inch of the jack case. When the airplane is jacked to the desired height, screw the locknut down until it seats against the case.

j. If the airplane has been jacked outside the hanger, moor the airplane in the normal manner so that wind gusts will not shift the airplane.

7.0.1.5 LOWERING THE AIRPLANE.

a. Clean the polished surfaces of the strut pistons and apply a light coat of MIL-L-7870 oil to them.

b. Check the area beneath the airplane to be sure it is clear of all obstructions.

c. Release the jack locknuts and lower the jacks at an even rate to keep the airplane level.

7.0.2 HOISTING. Hoisting provisions are available for the empennage assembly and the engines. See figure 7-4 for the means of attachment and part numbers on the hoists and slings. See figure 7-5 for approximate weight of major components.

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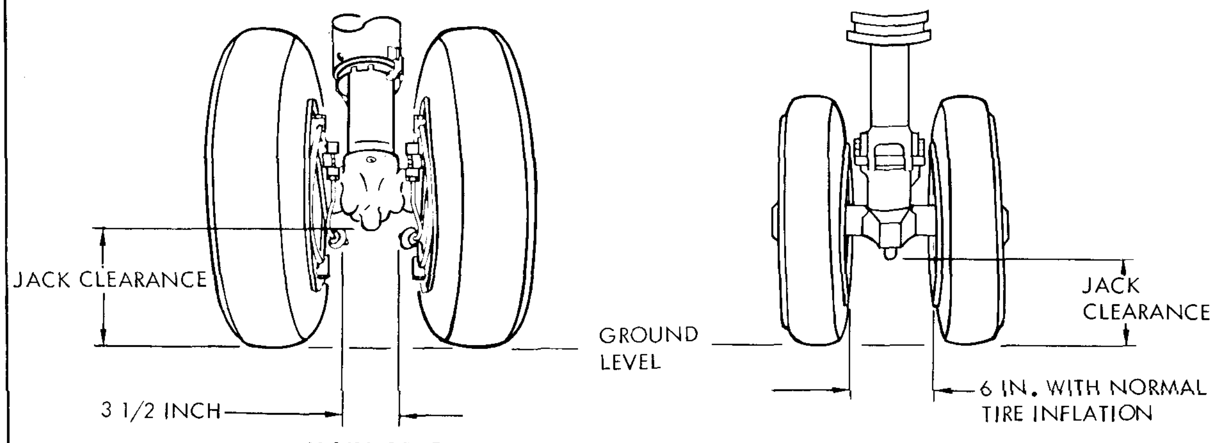


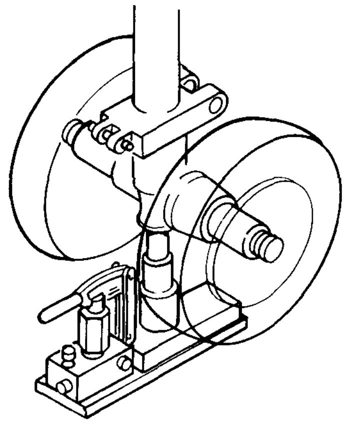
MAIN GEAR

NOSE GEAR

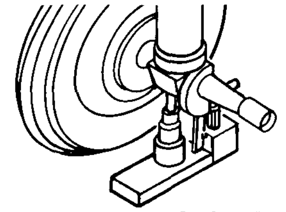
LANDING GEAR GROUND CLEARANCES

CONDITION	JACK CLEARANCE (APPROXIMATE)
BOTH TIRES, NORMAL INFLATION	8 1/2 IN.
ONE TIRE FLAT	8 IN.
BOTH TIRES FLAT	5 3/4 IN.

CONDITION	JACK CLEARANCE (APPROXIMATE)
BOTH TIRES, NORMAL INFLATION	5 3/4 IN.
ONE TIRE FLAT	5 IN.
BOTH TIRES FLAT	4 IN.



NOSE LANDING GEAR AXLE JACK



MAIN LANDING GEAR AXLE JACK

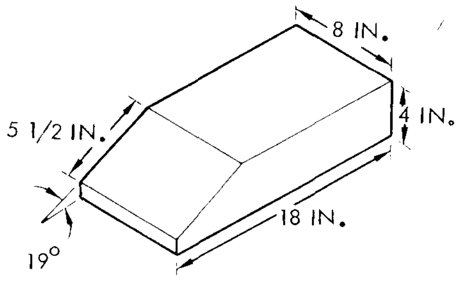


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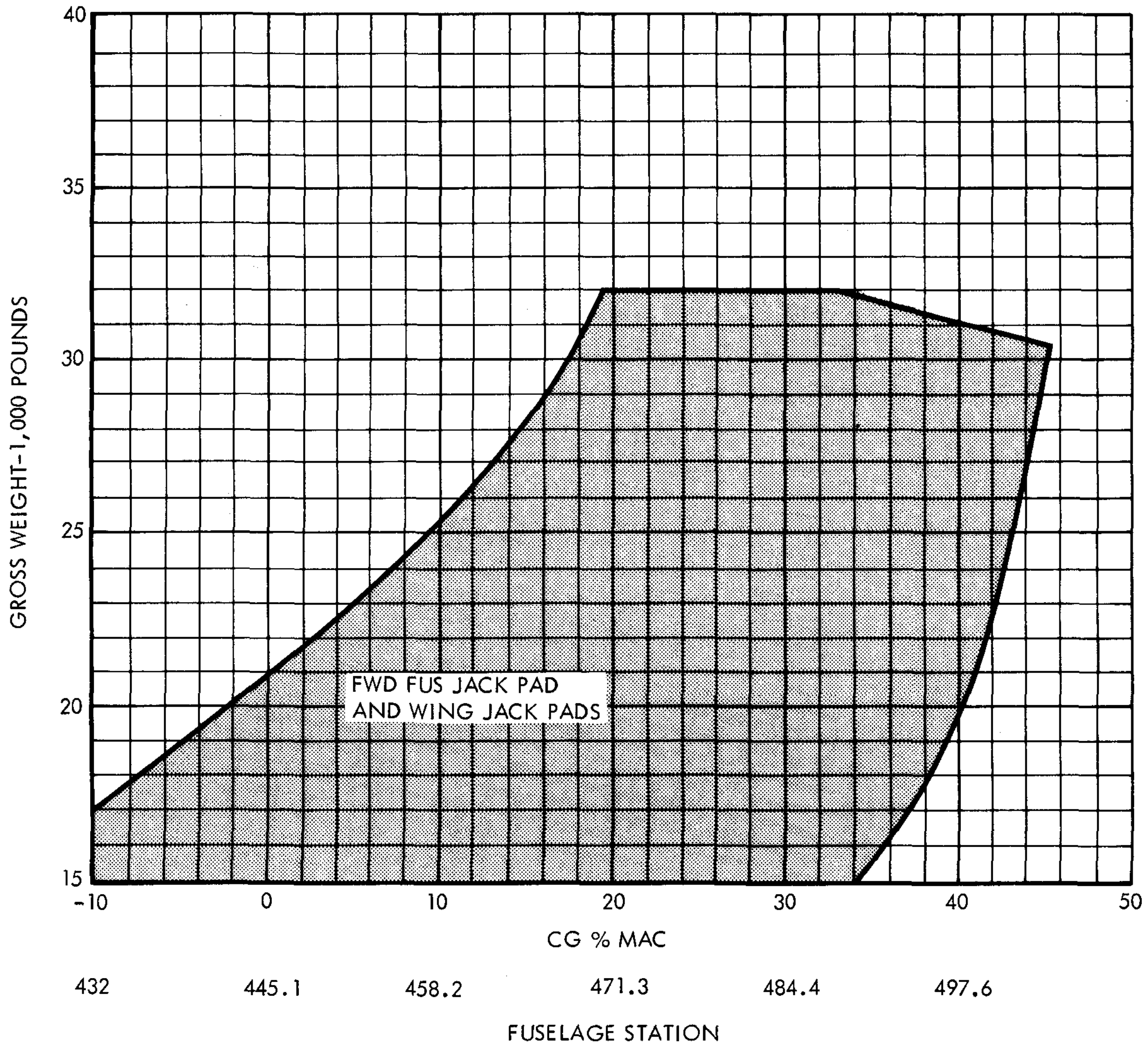
Figure 7-1. Landing Gear Jacking Provisions



AIRPLANE CENTER OF GRAVITY (CG) LOCATION

NOTE

1. JACK THE AIRPLANE WHEN THE WEIGHT AND CG FALL INSIDE THE SHADED AREA ON THE CHART.

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Figure 7-2. Airplane Center of Gravity Chart

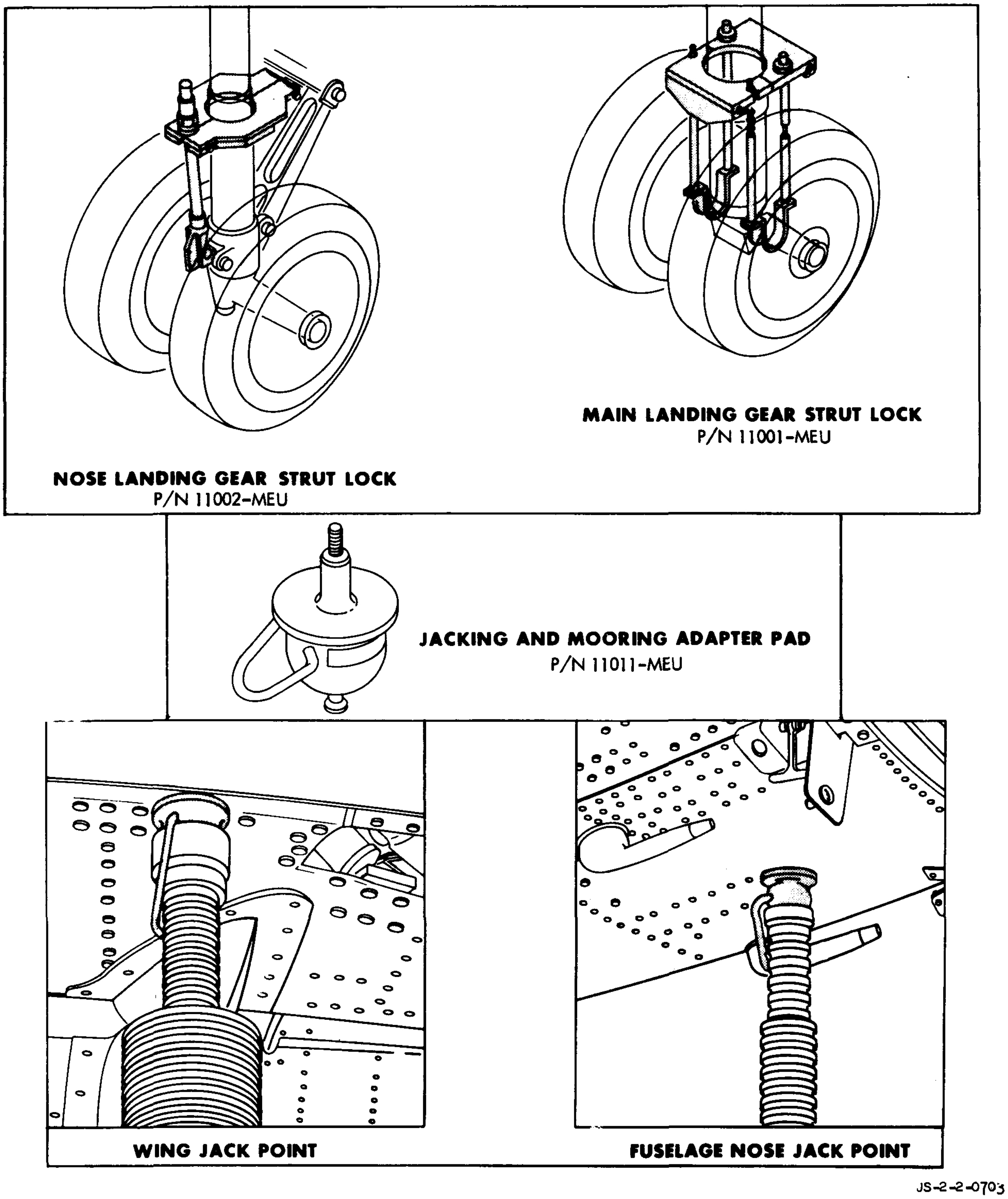
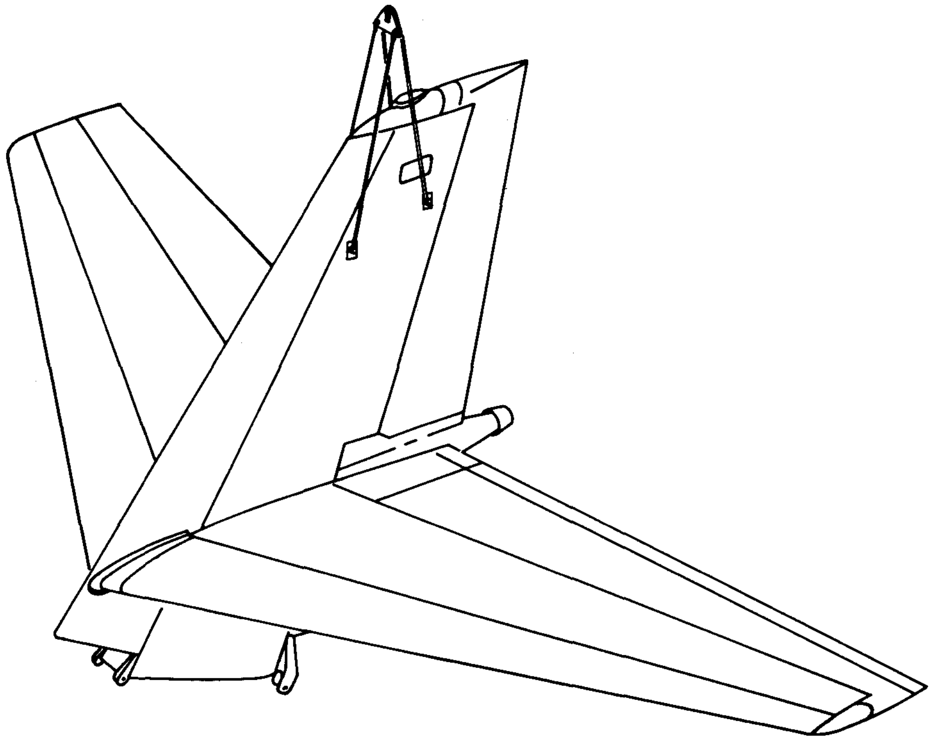
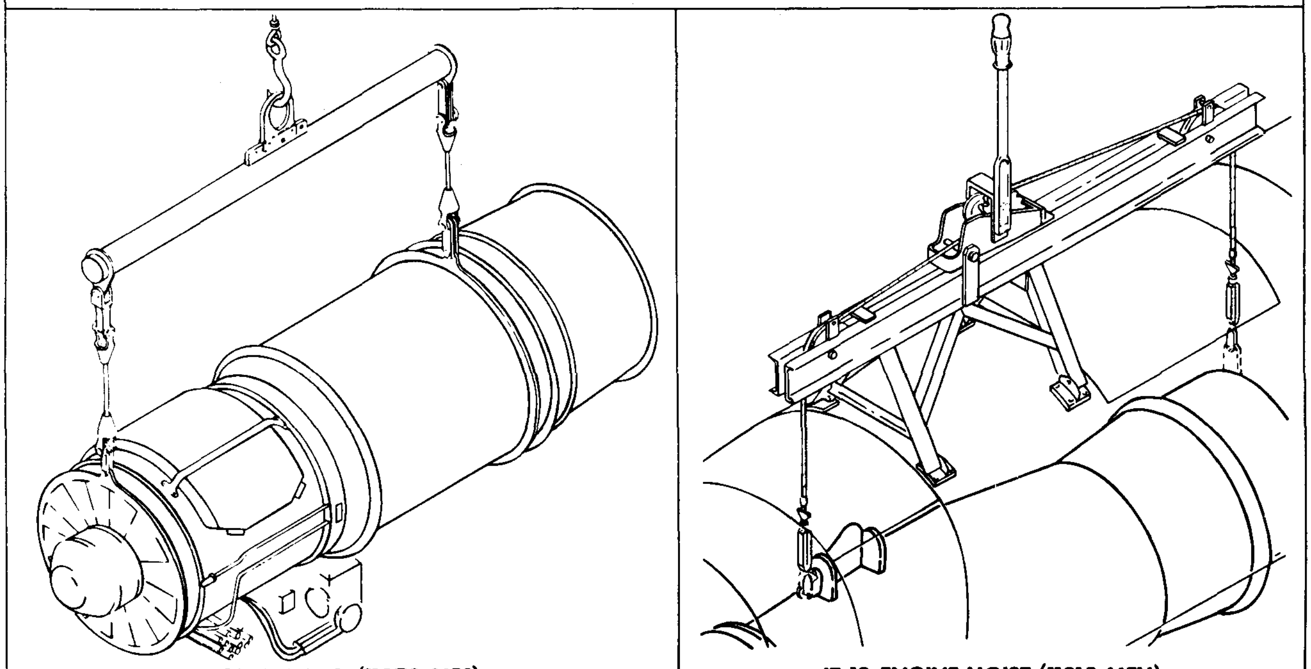


Figure 7-3. Wing and Fuselage Jacking Provisions



AIRPLANE GENERAL PURPOSE SLING (11007-301-MES OR 7000-101-MES)



ENGINE SLING (11073-MES)

JT-12 ENGINE HOIST (11015-MEH)

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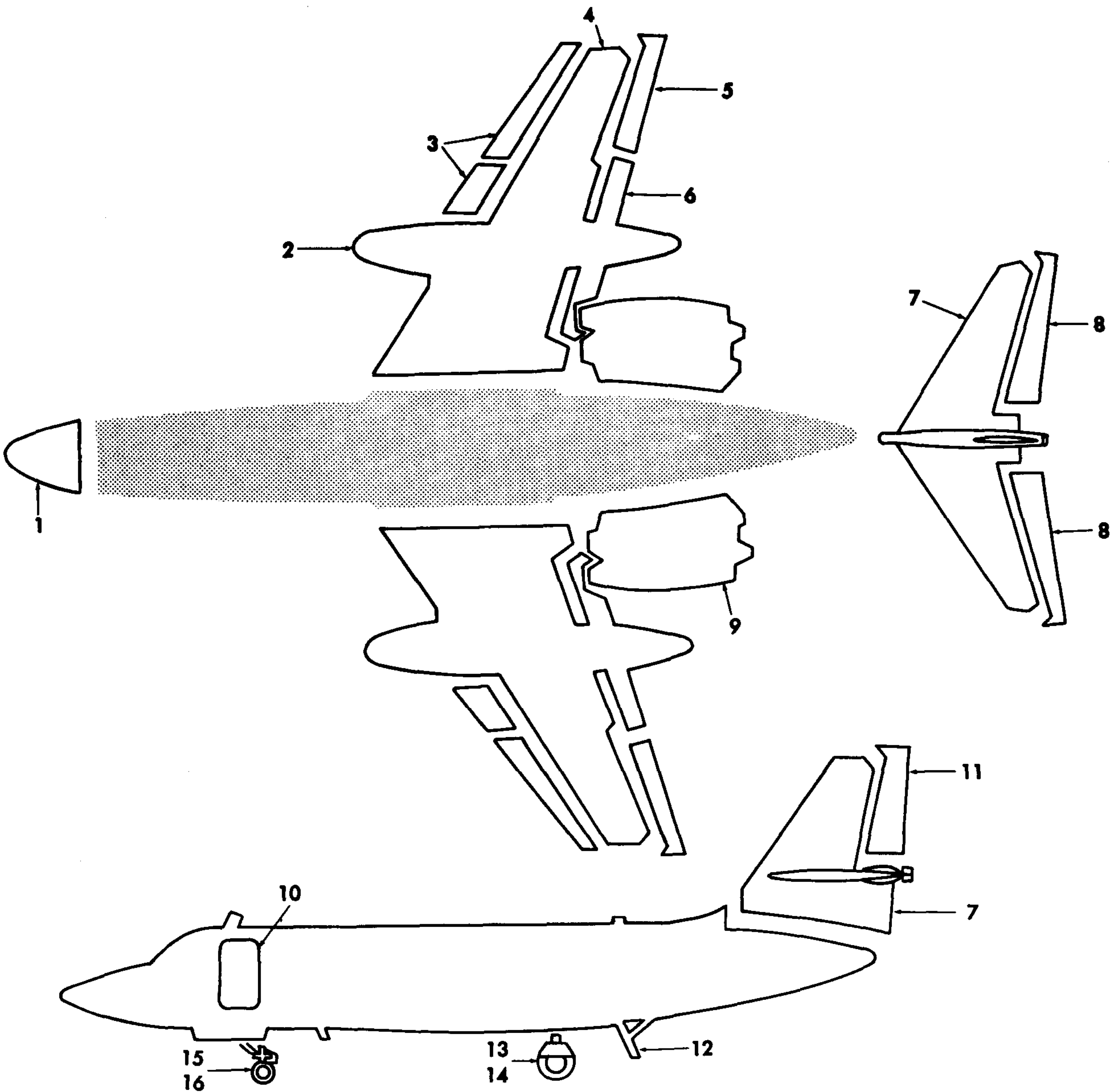
Figure 7-4. Hoisting Provisions



APPROXIMATE WEIGHT OF MAJOR COMPONENTS

1. SLIDING NOSE ASSEMBLY	82 LBS.
2. EXTERNAL FUEL TANK ASSEMBLY (EACH)	333 LBS.
3. LEADING EDGE FLAP (BOTH SECTIONS, EACH WING)	101 LBS.
4. WING ASSEMBLY (LESS FLAPS, AILERON, AND EXTERNAL TANK)	1516 LBS.
5. AILERON (EACH)	52 LBS.
6. TRAILING EDGE FLAP (BOTH SECTIONS, EACH WING)	79 LBS.
7. EMPENNAGE ASSEMBLY (LESS THE ELEVATORS AND RUDDER)	639 LBS.
8. ELEVATOR (ONE SIDE)	56 LBS.
9. ENGINE NACELLE ASSEMBLY (EACH SIDE, LESS ENGINES)	760 LBS.
10. MAIN ENTRANCE DOOR	47 LBS.
11. RUDDER	73 LBS.
12. SPEED BRAKE	88 LBS.
13. MAIN LANDING GEAR STRUT ASSEMBLY (LESS WHEELS AND TIRES)	128 LBS.
14. MAIN GEAR WHEEL AND TIRE (EACH ASSEMBLY)	110 LBS.
15. NOSE LANDING GEAR STRUT ASSEMBLY (LESS WHEELS AND TIRES)	146 LBS.
16. NOSE WHEEL AND TIRE (EACH ASSEMBLY)	22 LBS.

JS-2-2-0705

Figure 7-5. Major Component Weights

SECTION 7.1
SHORING

Contents Page

GENERAL 7-11

WING ACCESS PANELS 7-11

7.1.1 GENERAL. When the removal of a major structural item is required, it may be necessary to shore portions of the aircraft to avoid excessive deformation or failure. Shoring required to put the wing in neutral position is defined in paragraph 28.2.2.2. For specific shoring instruction, see Technical Information, page Introduction-3.

7.1.2 WING ACCESS PANELS. It is necessary to shore the wing when the upper wing covers are removed. Detailed instructions for access cover removal are given in Section 28.2.

SECTION 7.2

METHODS OF HANDLING A DAMAGED AIRPLANE

Contents Page

GENERAL 7-13
EQUIPMENT AND MATERIALS 7-13
MOORING THE AIRPLANE FOR PNEUMATIC BAG LIFTING 7-14
METHOD OF USING PNEUMATIC BAGS 7-14
RAISING THE AIRCRAFT ON ONE WING USING PNEUMATIC BAGS 7-15
RAISING AIRPLANE WITH NOSE GEAR COLLAPSED USING A CRANE 7-15
MOVING AIRPLANE WITH BOTH MLG AND NLG STRUCTURALLY INOPERATIVE 7-15
MOVING THE AIRCRAFT WITH THE NOSE GEAR INOPERATIVE 7-15
MOVING THE AIRCRAFT WITH EITHER MLG OR NLG OPERABLE 7-16

FIGURES

7-6. Pneumatic Bag Placement Under Wing and Fuselage 7-17
7-7. Airplane Mooring for Pneumatic Bag Raising. 7-18

7.2.1 GENERAL. Once an airplane is damaged on a runway, quick and effective means must be employed to clear the runway. During any removal, two major considerations should be kept in mind.

a. Safe methods for lifting and removal will be utilized to protect personnel.

b. Approved procedures for handling the airplane will prevent further damage to the structure or components.

First steps in lifting the airplane in preparation for removing it from the runway is to reduce the weight to a minimum.

As parts and equipment are removed, the effect on gross weight and CG can be computed from data in the handbook of Weight and Balance data and supplement applicable to the damaged airplane.

Pneumatic Bags provide an alternative means of raising an airplane when the use of mechanical lifting devices is impracticable or undesirable. The bags are 6-1/2 feet wide by 7-1/2 feet long (inflated) and have a collapsed height of 6 inches, expanding to six feet when inflated. The bags are made of a nylon fabric, similar to fuel bladder cells, and have a lifting capacity of 24,000 pounds. Each bag weighs 135 pounds and is inflated using a 3.5 psig at 40 cfm. blower, through a hose supplied with the bags. The bags can be used in temperature of -55°C to 70°C. The tarpaulin cover serves as a protective pad on which to place the bag during lifting.

The manner in which the bags are used will vary considerably with many different factors such as airplane weight, position of C.G., terrain, structural condition, physical position and availability of equipment. In general however, the bags should be placed so that the maximum possible lifting area is in contact with the airplane structure and the airplane is kept in a stable condition. Use as many bags as possible. Careful consideration should be given to the position of the airplane C.G. for the particular case. If possible, the C.G. position should be improved and the bags placed and inflated according to the findings. In this manner it should be possible to prevent the airplane tilting or shifting position suddenly. An additional safety precaution is that of mooring the airplane during the lifting operations. This is explained more fully below. The best place to apply lift is between the spars as near to the wing root rib as possible. Care should be taken to place the bags to avoid obstructing the MLG door and jack pad areas. This way either the gear could be extended or the jacks could be installed to replace the bags. Another bag could be placed under the nose just aft of the nose jack pad area in case of nose gear collapse. Two other bags should be placed, one on the outboard of each wing for stabilizing the airplane only. These bags should be inflated gradually only after the other bags have absorbed the lifting load. It should be understood that the pneumatic bags are a means for lifting only and are not intended as a semi-permanent means of support.

7.2.2 EQUIPMENT AND MATERIALS.

a. Pneumatic Bags - 24,000 pound lifting capacity, MIL-P-6640. Available from the following sources:

U. S. Rubber Co., Woonsocket, R. I.

General Tire & Rubber Co., Akron, Ohio

Goodyear Tire & Rubber Co., Los Angeles, Calif.

Firestone Tire & Rubber Co., Los Angeles, Calif.

b. Air Compressor, or equivalent - capable of maintaining 3.5 psig minimum at a volume of 40 cfm.

7.2.2 (Cont).

Note

For convenience in rough terrain, air-nitrogen bottles, used with a regulator, would be the best means of inflation.

c. Wood timbers suitable for cribbing wing and fuselage.

d. Emergency Mooring Equipment

1. Cables or ropes of sufficient strength to adequately moor the airplane during lifting operations with pneumatic bags.

2. Block and tackle or equivalent means of keeping cable taut or paying cable out while lifting airplane.

3. Padding for insertion between cables, cribbing and airplane structure.

7.2.3 MOORING THE AIRPLANE FOR PNEUMATIC BAG RAISING. The airplane is equipped with mooring and tie down provisions on the MLG and NLG and all three jack points (Reference figure 7-3, page 7-8). For mooring when using pneumatic bags while the gears are retracted, see figure 7-7 and the following instructions.

a. Install jacking and mooring adapter pads (P/N 11011-MEU) to wing and fuselage jack points.

b. Attach cables and/or block and tackle assemblies to these three points to support the airplane in both forward and aft inboard and outboard axis.

c. Anchor the ends of these cables or ropes to stakes driven into the ground or tied to heavy equipment placed in the proper position.

•7.2.4 METHOD OF USING PNEUMATIC BAG.

a. Air bags may be placed under fuselage near nose or tail in order to keep airplane level while it is being lifted.

b. Ordinarily, lifting load on fuselage structure is not enough to develop local damage, but some areas should not be lifted with air bags. The following fuselage areas are capable of support when using air bags.

1. Directly aft of fuselage station 232.00 under cockpit area.

2. Forward of fuselage station 490.00 between wing root ribs.

c. The following underbody areas must not be used with air bags for lifting.

1. Under the main landing gear wheel well door areas.

2. Wherever antennas or pitot systems are installed.

d. Moor airplane to restrict its movement. See paragraph 7.2.3 "MOORING AIRPLANE FOR PNEUMATIC BAG RAISING."

e. Clear ground, on which bag is to be placed, of all sharp or rough objects and check that underside of structure to be lifted has no sharp protrusions or other possible sources of damage to the bag. Any such sources should be removed or covered with a thick padding.

f. Position a bag near each location shown on figure 7-6, undo tarpaulin cover straps and remove inflation hose.

g. Spread the tarpaulin cover flat, unroll the bag and arrange the bottom of the bag so that it is flat and properly centered on the cover.

h. Unfasten the outlet covers and check that each outlet sleeve is properly tied off to prevent air escaping.

j. Replace sleeve inside its base reinforcement, and close and secure outlet covers.

k. Maneuver tarpaulin cover into position, attempting to get the inlet valve as near the air supply source as possible. The cover should be positioned so that the bag will have maximum contact with the structure.

l. Remove valve cap and attach inflation hose to bag and to air supply source.

m. Start the blowers and inflate the bags. Progressively crib the inspar area of wing and under bulkheads at fuselage stations 232.00 and 490.00 to provide secondary support in the event of a bag collapse.

CAUTION

THE PRIME REQUISITE IS TO KEEP THE AIRPLANE LEVEL THUS PREVENTING SLIPPING OR TILTING.

n. Support airplane on its landing gear or, if this is impossible, on cribbing, crash dollies, or jacks.

CAUTION

JACKS MUST BE SET ON LEVEL GROUND WITH THE RAM VERTICAL TO AVOID IMPOSING SIDE LOADS ON JACK OR JACK PADS. USE STEEL OR WOODEN SUPPORTS OF SUFFICIENT THICKNESS AND BEARING AREA, AS NECESSARY, TO DISTRIBUTE LOADS AND PREVENT JACKS FROM TIPPING OR SINKING INTO SOFT SURFACE.

o. Deflate bags by opening one or both outlets.

7.2.4 (Cont).

Note

Using both outlets release pressure quickly in case of an emergency.

p. Remove inflation hose, replace valve cap, and move tarpaulin cover, with bag still on it, away to an uncongested area for packing.

q. Examine all parts and surfaces for damage. Clean as required.

r. Deflate the bag by compressing and rolling it towards the outlets.

s. Place rolled bag centrally on tarpaulin, place coiled inflation hose toward one end of rolled bag, fold over tarpaulin ends and secure strap.

7.2.5 RAISING THE AIRCRAFT ON ONE WING USING PNEUMATIC BAGS. **Air bags are advantageous for lifting one side of airplane only. When one main gear collapses and other main gear and nose gear support the airplane in a one-wing-down position, low wing must be rotated until it is about level with opposite wing. As low wing comes up, jack point moves as it rotates about opposite main gear. If a jack were used for such a lifting job, severe side loads would be induced and jack would require shifting in increments to keep up with shift of the jack point.**

a. Moor aft fuselage section of airplane as outlined in paragraph 7.2.3.

b. Defuel down wing and inboard tanks of opposite jack point.

c. Allow fuel to remain in up wing outboard and external tanks. This will provide aid in rotating down wing up again and reduce lifting loads under down wing.

d. Raise down wing to height so that MLG can be extended.

e. Large sheets of steel or timbers must be used under jacks to distribute load over a wide ground area.

f. Remove all remaining fuel to reduce overall weight for towing.

g. If airplane is on soft ground, planking will provide a temporary roadway to main runway surface.

7.2.6 RAISING AIRPLANE WITH NOSE GEAR COLLAPSED USING JACKS.

a. Raise airplane by using a jack at each wing jack pad. Do not exceed maximum load on each wing jack. Figure 7-2, Page 7-7.

b. Crib forward fuselage at fuselage station 232.00 as airplane is raised by wing jacks.

c. Follow forward fuselage up with a restraining block and tackle to prevent aft fuselage from rotating around wing jack points to ground.

7.2.7 RAISING AIRPLANE WITH NOSE GEAR COLLAPSED USING A CRANE.

a. A crane may be used in combination with a sling to lift the nose.

b. The sling should be tied around nose at fuselage station 232.00; this is the strongest bulkhead in the nose for such a lifting operation. Thick padding or tires must be used between the fuselage and the sling to distribute the load and protect the airplane from damage.

c. The load at the nose should be reduced to a minimum, even if a crane of larger capacity is available.

d. A load in tail either in the aft equipment area or spread evenly on floor will move the CG aft.

7.2.8 MOVING THE AIRPLANE WITH BOTH MAIN GEAR AND NOSE GEAR STRUCTURALLY INOPERATIVE.

a. The area of wing most capable of supporting weight of airplane on trailer is also location of main jack pads. Therefore, airplane must be supported at rear spar while trailer is moved into position.

b. A platform of railroad ties or equivalent means of cribbing should be built up to an angle matching the dihedral of the wing and the contour of the fuselage to more equally distribute the load.

c. A covering of tires or equivalent should be used over the cribbing material so as to prevent damage to the airplane.

7.2.9 MOVING THE AIRCRAFT WITH THE NOSE GEAR INOPERATIVE.

a. If airplane is to be moved without using nose gear, use flat bed truck (capacity 10,000 pounds) under fuselage forward section. Tow airplane using two tractors attached by cables to MLG towing lugs. Coordinate movement of tractors and truck.

b. The load at the nose should be reduced to a minimum. Refer to paragraph 7.2.7 d. for information on ballasting the rear of the airplane to reduce the weight at the station 232.00 bulkhead.

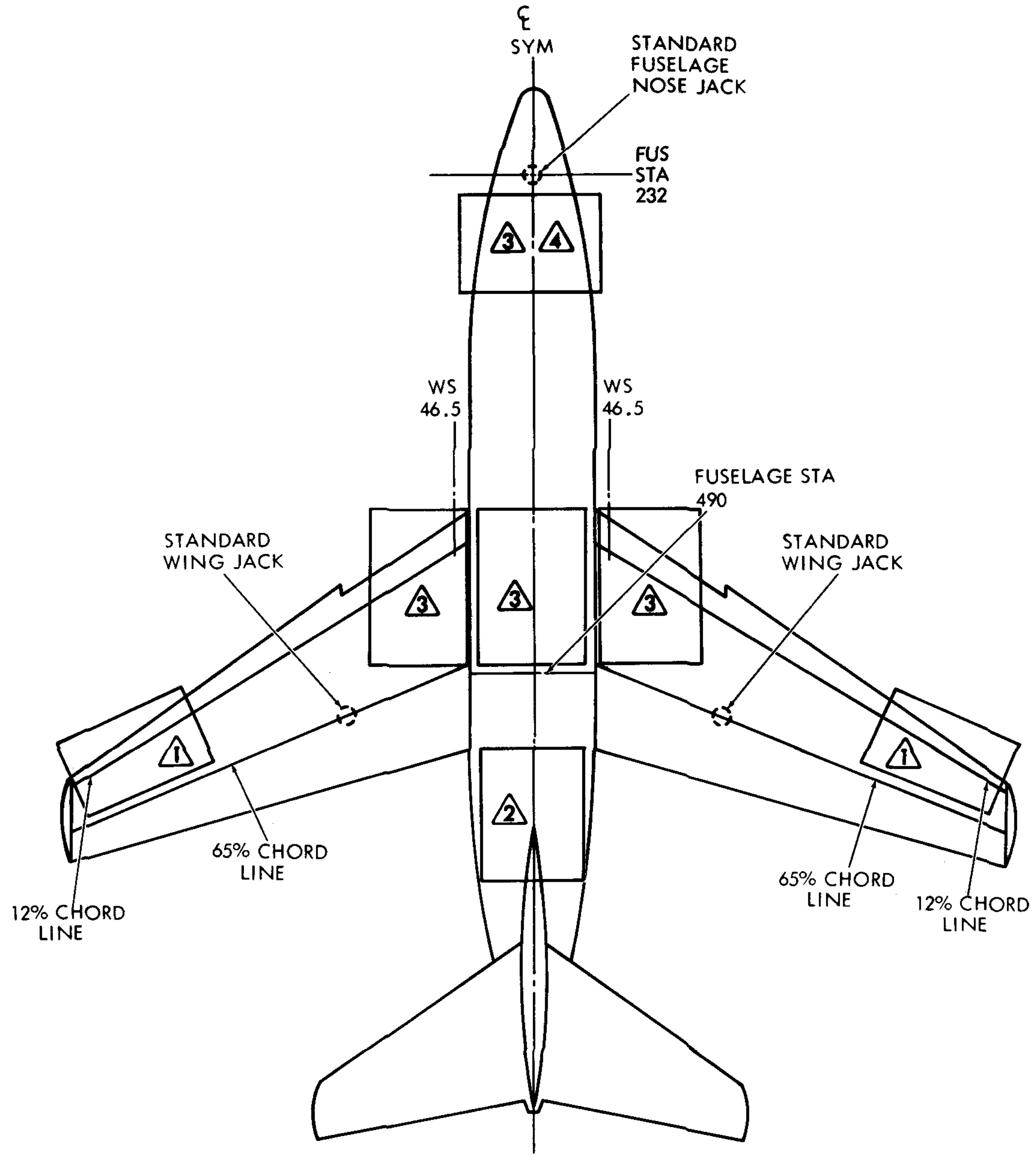
c. A cradle, padded and contoured to fit the underside of the nose at the station 232.00 bulkhead is used to support the nose on truck.

7.2.9 (Cont).

d. Some kind of moveable load table is needed between cradle and truck bed to permit turning the nose and allow guiding airplane while turning.

7.2.10 MOVING AIRCRAFT WITH EITHER THE NOSE GEAR OR MAIN GEAR OPERABLE. When an airplane is damaged during a wheels up landing, landing gear may be released after the airplane is on jacks. This will simplify moving airplane to a maintenance facility.

Even though retracting mechanism for nose gear is damaged and gear is not vertical, temporary bracing may be used for towing.



NOTE

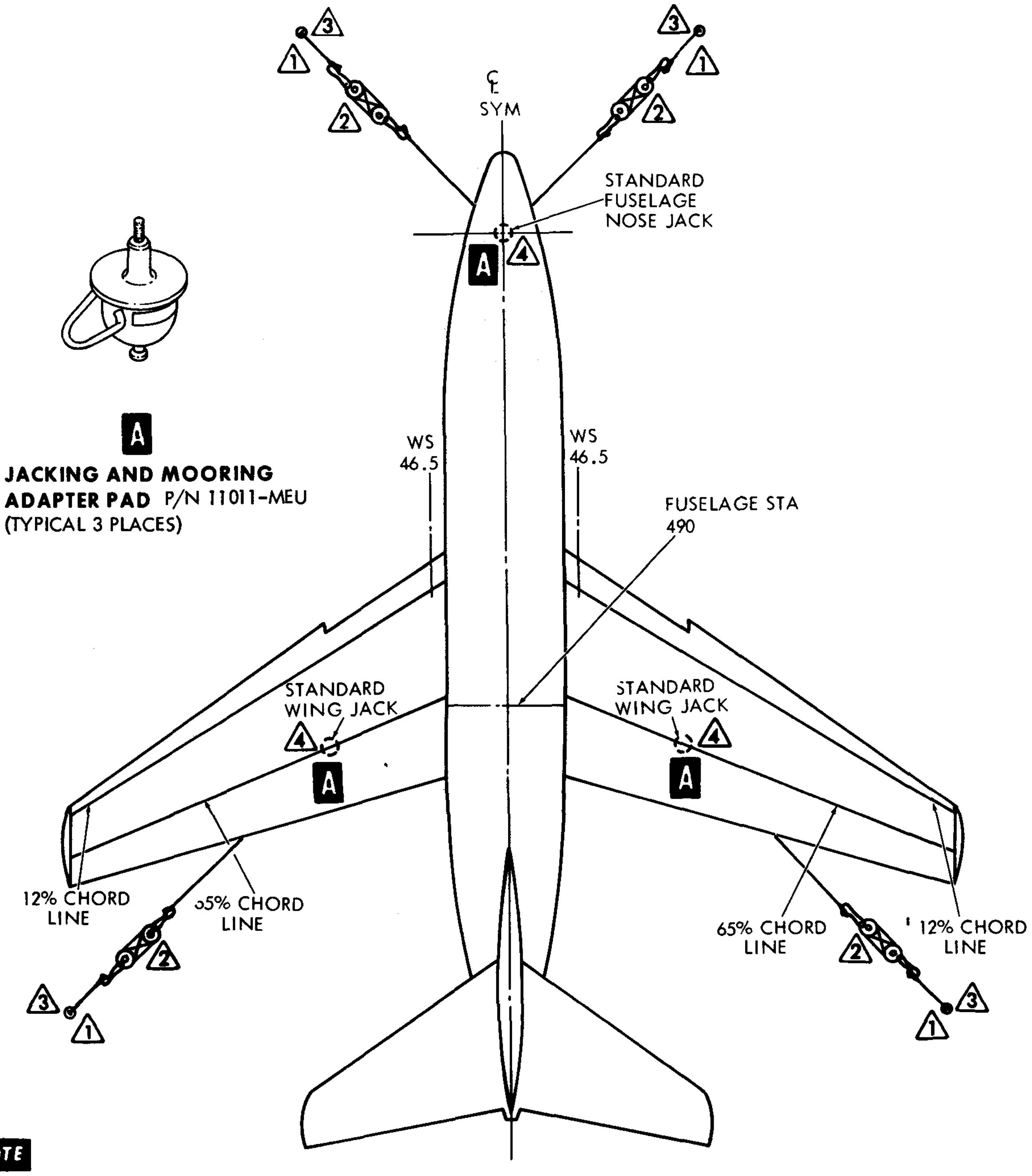
△1 WING STABILIZING ONLY

△2 FUSELAGE STABILIZING ONLY

△3 AIRPLANE LIFTING POSITION

△4 NOSE LIFTING POSITION

Figure 7-6. Pneumatic Bag Placement Under Wing and Fuselage



NOTE

1 PLACE MOORING PROVISIONS AT BEST POSSIBLE LOCATION TO SUPPORT AIRPLANE ON PNEUMATIC BAGS

2 BLOCK AND TACKLE OR EQUAL ARE TO BE PLAYED OUT AS AIRPLANE IS LIFTED BY PNEUMATIC BAGS

3 ANCHOR TO GROUND STAKES OR HEAVY EQUIPMENT

4 PAD BETWEEN CABLES AND/OR ROPES AS REQD TO PROTECT AIRPLANE SURFACES

Figure 7-7. Airplane Mooring for Pneumatic Bag Raising

CHAPTER 8
LEVELING AND WEIGHING

TABLE OF CONTENTS

Section	Page
8.0 LEVELING	8-3
8.1 WEIGHING	8-7

SECTION 8.0
LEVELING

Contents Page

LEVELING THE AIRPLANE 8-3
External Leveling Procedure 8-3
Internal Leveling Procedure. 8-3

FIGURES
8-1. External Leveling Provisions 8-5
8-2. Internal Leveling Provisions. 8-6

8.0.1 LEVELING THE AIRPLANE.

8.0.1.1 EXTERNAL LEVELING PROCEDURE. This method of leveling the airplane makes use of the alignment and symmetry points shown on figure 8-1. A surveyor's transit, a steel scale or equivalent, and jacks are required to level the airplane by the following procedure:

a. Jack the airplane slightly to establish a near level position of the airplane.

Note

To level the airplane, jack at the wing jack points and nose jack points only. Use no other means of support.

b. Set up a surveyor's transit slightly aft of the airplane at a point from which both wing tips can be seen.

c. Level the transit.

d. With one man holding the scale vertically so the top of the scale is touching point 3R (see figure 8-1) take a reading from the scale through the transit.

e. Repeat step (d) for point 3L.

f. Any difference in the readings taken at points 3R and 3L indicates a difference in elevation between these two points. Adjust the wing jacks until the readings at points 3R and 3L are the same. The airplane is now laterally level.

g. Set up the transit outboard of the wingtips at a location from which points 1 and 2 (see figure 8-1) can be seen.

h. Level the transit.

i. With a man holding the scale vertically with the zero end against point 1, take a reading from the scale through the transit. This reading represents the height of point 1 above the transit line of sight.

j. Repeat step (i) for point 2.

k. Subtract the reading taken at point 2 from the reading taken at point 1. The difference should be 3.70 inches, with point 1 higher than point 2.

l. Adjust the nose jack until point 1 is 3.70 inches above point 2. The airplane is now level longitudinally.

m. Recheck the lateral level of the airplane by repeating steps (b) through (f).

8.0.1.2 LEVELING THE AIRPLANE USING INTERNAL LEVELING PROVISIONS. The two sets of internal leveling provisions in the airplane are shown in figure 8-2. Use the following procedure to level the airplane with the provisions at the entrance door:

a. Gain access through the interior headliner above the entrance door, and thread a string through the hollow rivet in the door header.

b. Attach a plumb bob to the lower end of the string, and allow the plumb bob to swing freely above the ladder track.

c. Jack the airplane until the plumb bob comes to rest directly above the 1/16-inch diameter hole in the upper face of the ladder track. The airplane is now level.

CAUTION

For leveling of the airplane, jack through the wing jack points and the nose jack point only. Use no other medium of support.

8.0.1.2 (Cont).

Use the following procedure to level the airplane with the provisions at fuselage station 449.59:

a. Remove the overhead liner and the floor panel covering the leveling provisions at fuselage station 449.59.

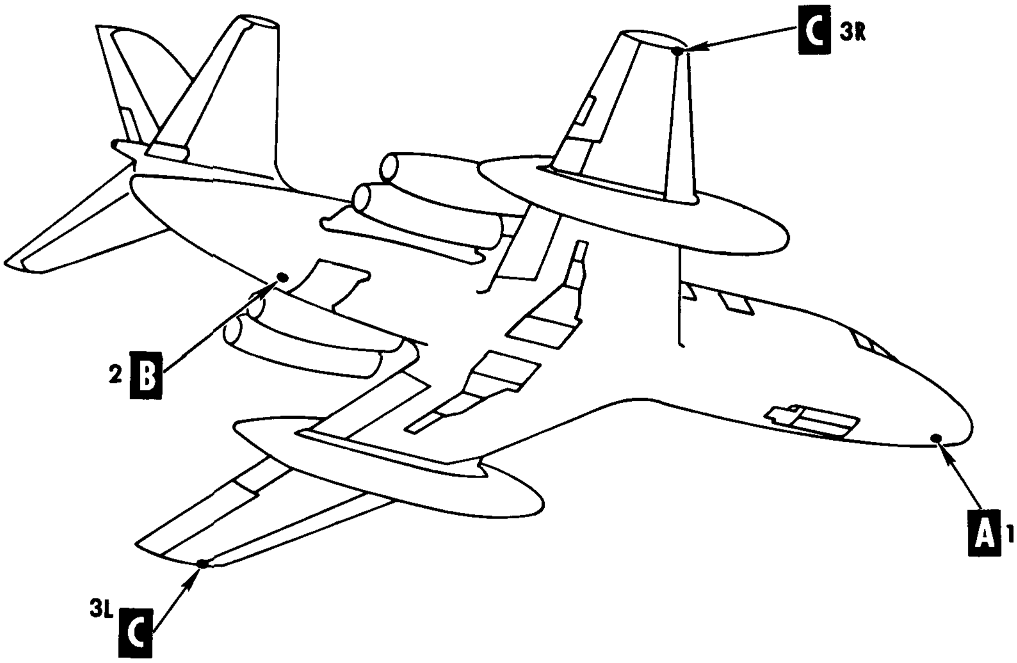
b. Thread a string through the hollow rivet in the overhead frame at fuselage station 449.59.

c. Attach a plumb bob to the lower end of the string, and position the bob so it will swing freely above the leveling plate on the lower frame.

d. Jack the airplane until the plumb bob is centered over the cross hairs on the leveling plate. The airplane is now level

CAUTION

For leveling the airplane, jack through wing jack points and nose jack point only. Use no other medium of support.



NOTE

1. ENTIRE AIRPLANE ALIGNED BY MEASURED HEIGHT OF POINT 1 ABOVE POINT 2. HOLD TO CLOSE LIMITS.
2. ALL POINTS ARE CIRCLED BY A 0.75 INCH DIAMETER RING PAINTED GLOSSY BLACK.

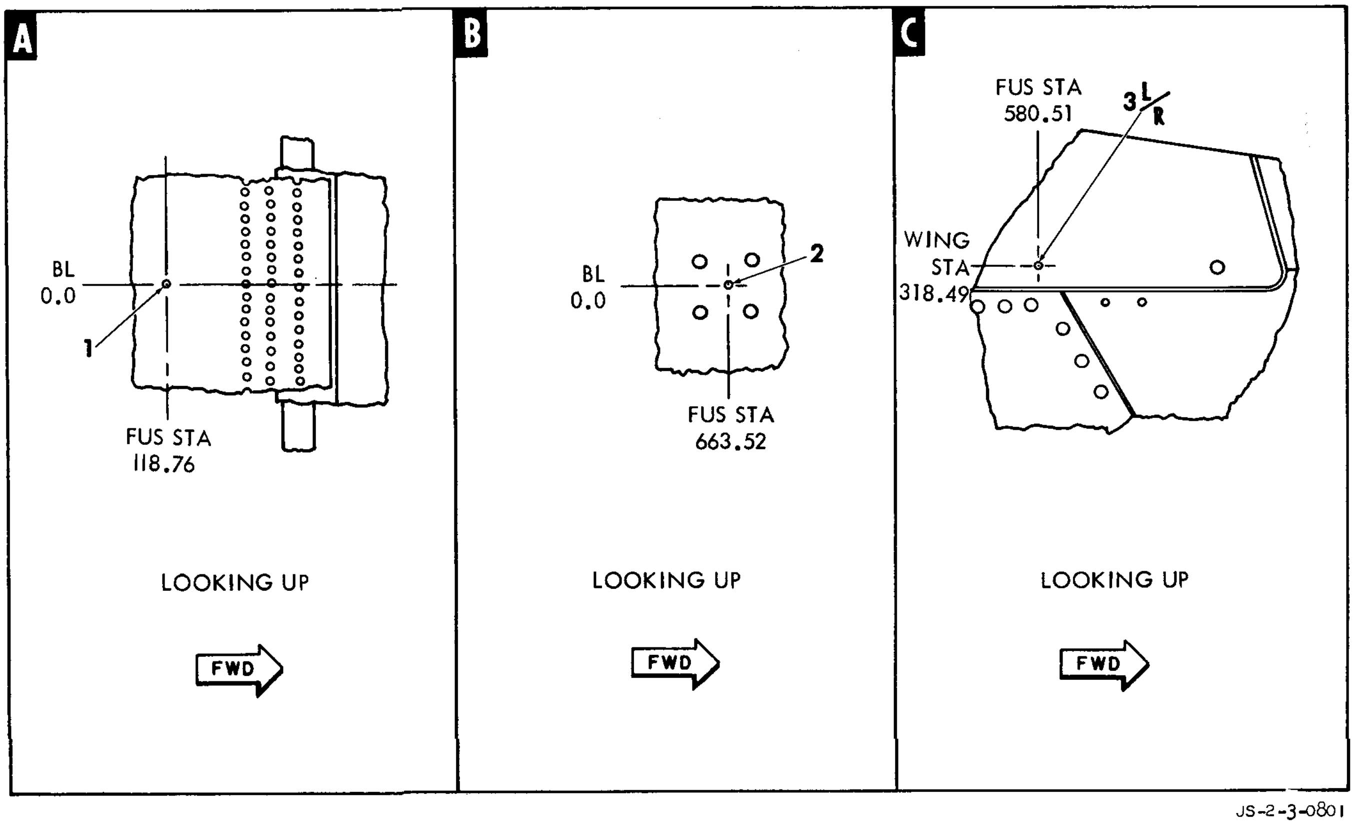


Figure 8-1. External Leveling Provisions

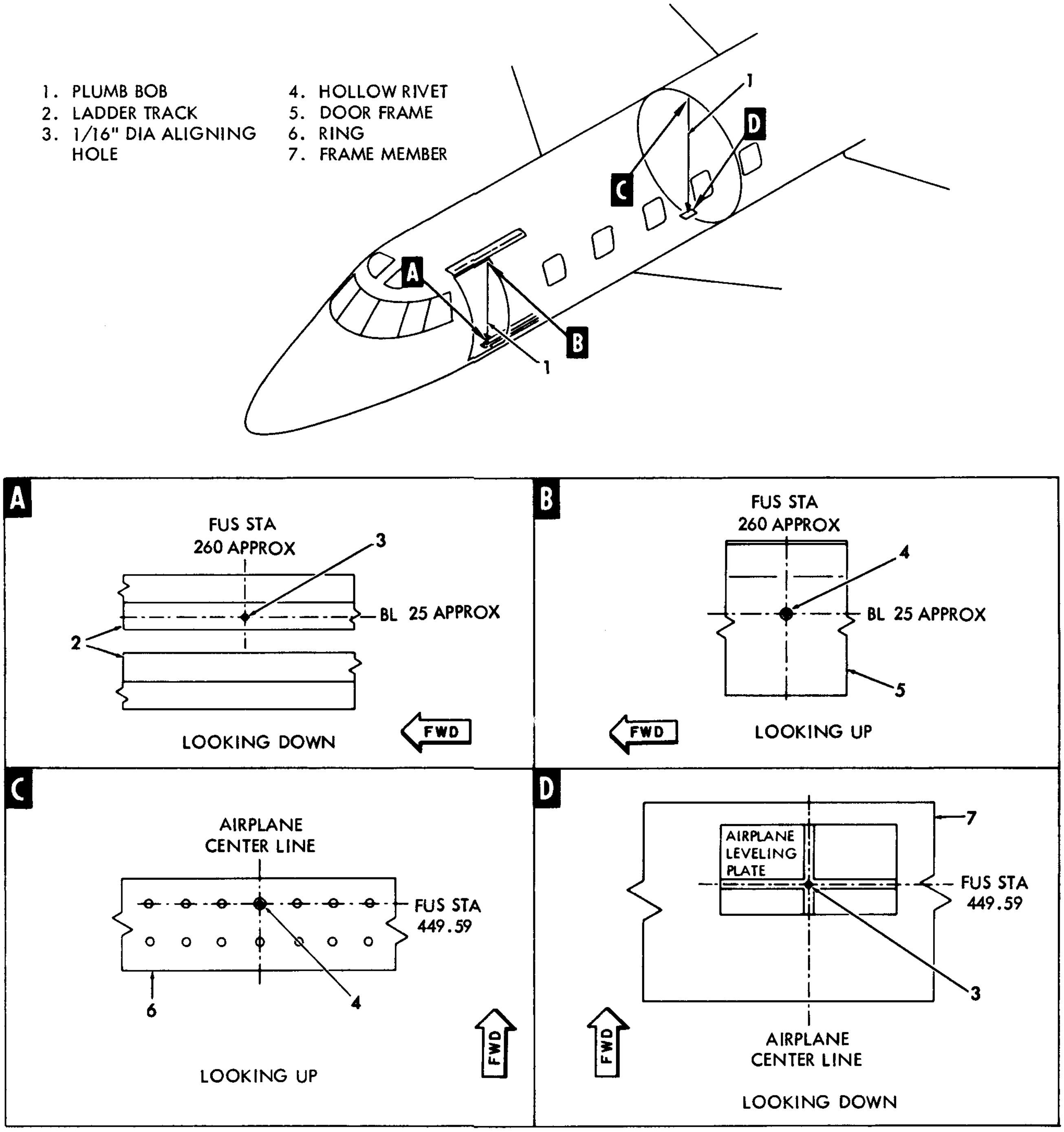


Figure 8-2. Internal Leveling Provisions

SECTION 8.1
WEIGHING

Contents Page

WEIGHING THE AIRPLANE 8-7

WEIGHING THE AIRPLANE. For weighing instructions refer to Section II of the JetStar Airplane Weight and Balance Handbook.

CHAPTER 9
TOWING AND TAXIING

TABLE OF CONTENTS

Section Page

9.0 TOWING 9-3

9.1 TAXING 9-7

SECTION 9.0
TOWING

Contents Page

GENERAL 9-3
NOSE GEAR TOWING 9-3
MAIN GEAR TOWING 9-3
TOWING PRECAUTIONS 9-3

FIGURES

9-1. Towing Provisions 9-5

9.0.1 GENERAL. Towing on a hard level surface is best accomplished with a nose gear towbar. When the airplane is on soft ground, deep snow, or when towing up a steep grade, tow with cables at the main landing gear instead of the towbar.

9.0.2 NOSE GEAR TOWING. Two steel lugs on the front of the nose landing gear (figure 9-1A) provide attachment for a Lockheed 1000-MEU-2 towbar or a Lockheed 11013-MEU towbar adapter kit. Always disconnect nose gear scissors when towing from the nose gear. To disconnect the scissors, simply remove the connection pin (figure 9-1B), separate the two scissor arms, and reinsert the connection pin in the upper arm as shown in figure 9-1C to keep the arm out of the way. Safe towing limit is 10,000 pounds maximum straight pull and 5,000 pounds at 45 degrees. Allowable tow loads with one flat tire are 6,000 pounds fore and aft and 3,000 pounds with nose gear swiveled 45 degrees to airplane center-line.

9.0.3 MAIN GEAR TOWING. Towing with cables at the main landing gear lugs (figure 9-ID) is accomplished with stranded steel tow cables, 3/8-inch in diameter or larger, about 50 feet long. This puts the tug at a minimum safe distance of about ten feet ahead of the nose. To avoid overloading the main gear structure, safe towing limits should be observed. Do not exceed a 7,500-pound straight pull or a ±30-degree tow cable angle from the fore and aft direction. Allowable tow load with one flat tire is 4,500 pounds fore and aft, at a 30-degree tow cable angle.

9.0.4 TOWING PRECAUTIONS.

WARNING

Failure to observe any of these safety precautions may result in damage to the airplane or injury to personnel.

a. Never tow the airplane while any of the engines are operating.

b. Never tow the airplane without having someone in the pilots' compartment to operate the brakes. Hydraulic power must be available to operate the brakes. The battery switch must be ON, the landing gear selector handle in the LG DOWN position, and the normal (No. 1) hydraulic system selected on the Hydraulic Power Selector Switch. Make sure the brakes are in good operating condition, the pressure gage on the accumulator in the nose wheel well indicates a charged accumulator, and brake system pressure is built up by operating the electrically driven No. 3 hydraulic pump. During towing, operate the No. 3 hydraulic pump as necessary to keep hydraulic pressure above 1,250 PSI (operate whenever the brake low pressure light is on).

CAUTION

On airplanes 5001 through 5045, not having detented landing gear selector valves, electrical power MUST be applied to the down side of the landing gear selector valve for the brakes to be operated with the electric pump.

9.0.4 (Cont).

c. When towing, wait until the airplane has started moving before turning the nose gear. A large force is required to overcome the inertia of the airplane when it is standing still. This force might place dangerous side loads on the landing gear structure if a start is attempted while the nose wheel is at an angle.

d. Never tow the airplane faster than a slow walk. Avoid quick stops and starts.

e. Never tow the airplane near obstacles without having someone walking at each wing tip and at the empennage to guard against collision. Make sure an adequate system for relaying signals is understood by everyone involved in the operation. Use candle type signal lights at night.

f. Never allow anyone to enter or leave the airplane or to ride on the outside of the airplane while it is moving.

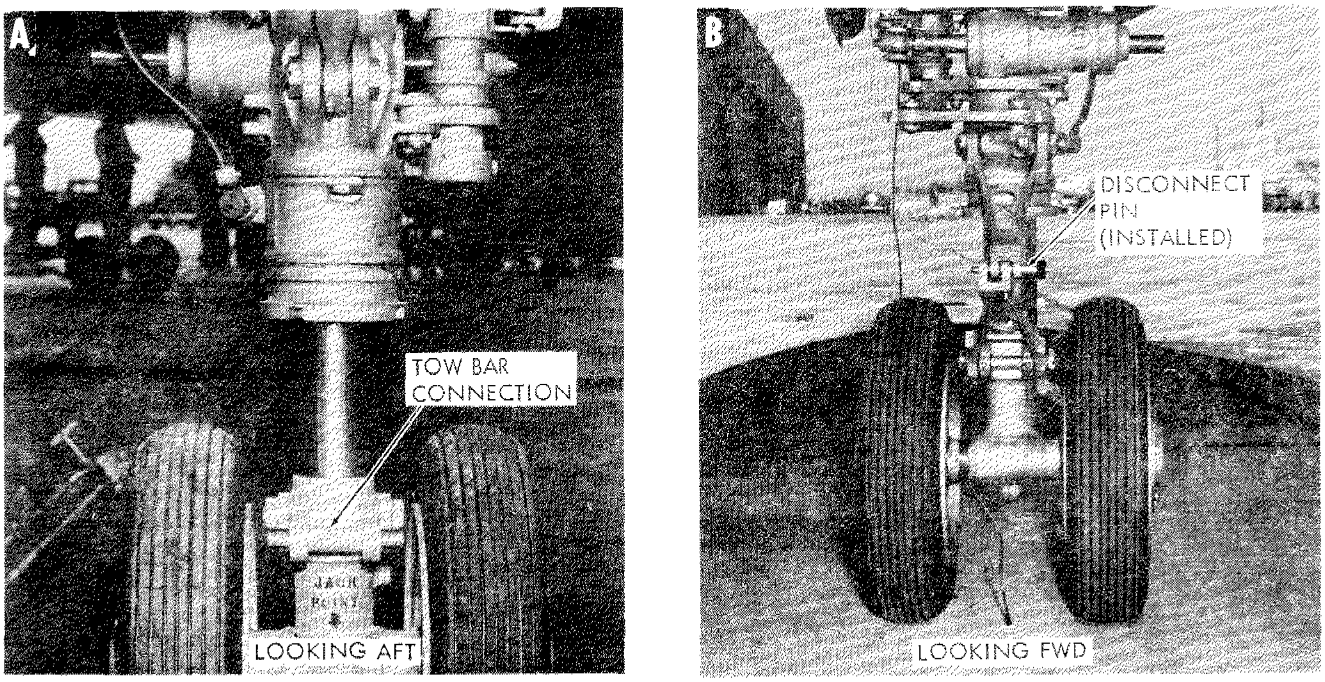
g. Never remove the chocks until ready to tow the airplane. Make sure the person in the flight station is ready to operate the brakes and that brake pressure is available. The normal (No. 1) hydraulic system must be selected by the switch on the instrument panel.

h. When towing from the nose gear, disconnect the nose gear scissors. Steering actuator damage can occur if the nose gear scissors are left connected and turning angle of 53 degrees is exceeded.

i. Check to see that landing gear downlocks have been installed. (See Chapter 32.)

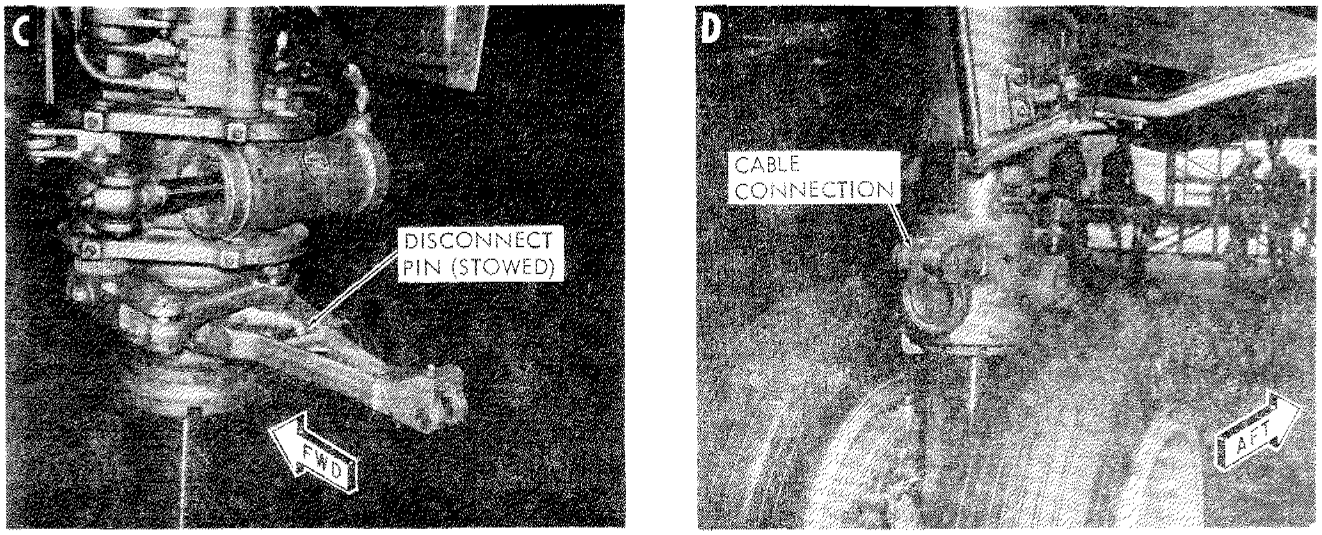
Note

Scissors disconnect pin should be installed on RH side of nose gear as shown in View B, Figure 9-1.



NOSE GEAR

NOSE GEAR
SCISSORS CONNECTED



NOSE GEAR
SCISSORS DISCONNECTED

MAIN GEAR

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Figure 9-1. Towing Provisions

SECTION 9.1
TAXIING

Contents Page

GENERAL 9-7

TURNING RADII 9-7

TAXIING PRECAUTIONS 9-7

PARKING BRAKE OPERATION 9-7

FIGURES
9-2. Turning Radius 9-8

9.1.1 GENERAL. When moving the airplane, employ all special procedures and precautions to insure the safety of personnel and to prevent damage to equipment.

9.1.2 TURNING RADII. Airplane turning radii requirements are shown in figure 9-2.

9.1.3 TAXIING PRECAUTIONS. Make sure the brakes are in good operating condition, the pressure gage on the accumulator in the nose wheel well indicates a charged accumulator, and brake system pressure has been built up. Hydraulic power is required for brakes and steering during taxi. One of the following is required to provide this power:

a. The No. 2 engine/pump operating with the normal (No. 1) hydraulic system selected.

b. The No. 3 engine/pump operating with the standby (No. 2) hydraulic system selected (airplanes 5046 and up). This option is not available on airplanes 5001 through 5045 not modified by Service Bulletin 329-214.

c. The electric motor driven hydraulic pump operating to keep the BRAKE LOW PRESS warning light OFF with the normal (No. 1) hydraulic system selected.

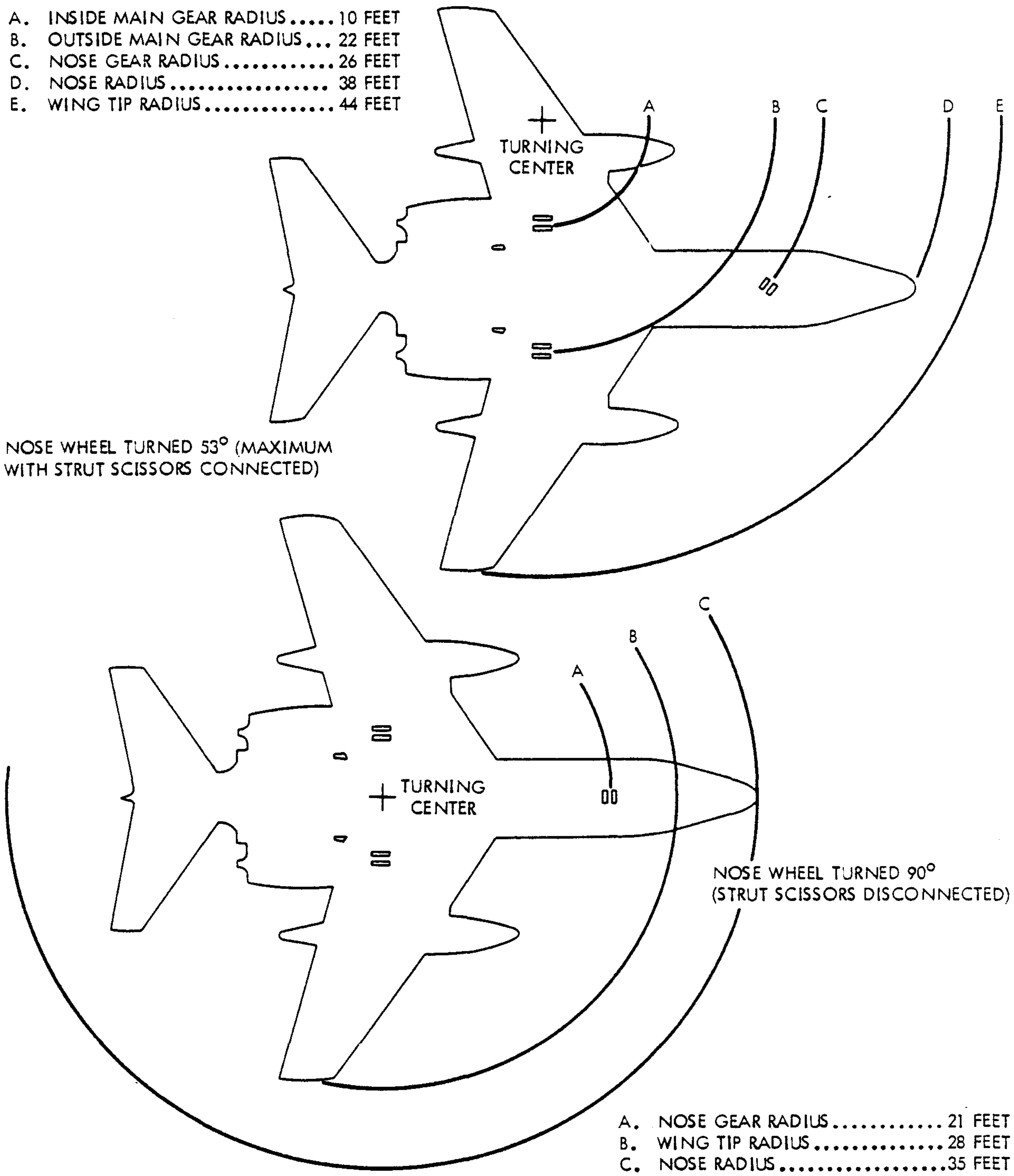
Note

When operating on the STANDBY-2 system, seven to ten full consecutive actuations of the nose gear steering cylinder will cause the hydraulic fuse to close. Equalizing the hydraulic pressure for a brief period of time will allow the fuse to reset.

CAUTION

On airplanes 5001 through 5040 (with Adel valves), electrical power must be applied to the down side of the landing gear selector valve to allow the brakes to be operated from the electric or the engine-driven pumps.

9.1.4. PARKING BRAKE OPERATION. Refer to Section 10.0.



A.	INSIDE MAIN GEAR RADIUS	10 FEET
B.	OUTSIDE MAIN GEAR RADIUS	22 FEET
C.	NOSE GEAR RADIUS	26 FEET
D.	NOSE RADIUS	38 FEET
E.	WING TIP RADIUS	44 FEET

NOSE WHEEL TURNED 53° (MAXIMUM WITH STRUT SCISSORS CONNECTED)

NOSE WHEEL TURNED 90° (STRUT SCISSORS DISCONNECTED)

A.	NOSE GEAR RADIUS	21 FEET
B.	WING TIP RADIUS	28 FEET
C.	NOSE RADIUS	35 FEET

Figure 9-2. Turning Radius

CHAPTER 10
PARKING AND MOORING

TABLE OF CONTENTS

Contents **Page**

10.0 PARKING **10-3**

10.1 MOORING **10-5**

10.2 STORAGE DURING PERIODS OF INACTIVITY 10-9

SECTION 10.0
PARKING

Contents

Page

PARKING 10-3
PARKING PRECAUTIONS 10-3
PARKING BRAKE OPERATION 10-3

10.0.1 PARKING. The main gear brakes may be set for temporary parking. Pressure for actuating these brakes is obtained from the normal hydraulic system accumulator or from the electrically driven No. 3 hydraulic system pump. Once the brakes are set, they will hold without additional pressure. They should not be set for extended periods, especially if they are hot from recent use. Use the following procedure to park the airplane:

a. Move the airplane into the parking area and to the desired spot. The exact position of the airplane may be dictated by anticipated use of maintenance equipment or engine operation. If a choice is available, head the airplane into the wind for parking.

b. Hold the airplane with its brakes. If desired to set the parking brakes temporarily, refer to paragraphs 10.0.2 and 10.0.3.

c. Have an assistant place chocks forward and aft of main gear wheels.

d. Release the brakes. If the parking brakes have been set, release them by depressing the rudder pedals as indicated in paragraph 10.0.3.

e. Connect a static ground wire to the grounding receptacle on the right side of the nose.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

f. If the airplane is to be parked for an extended period, install protective covers over the engine intake and exhausts and install plugs in the air conditioning system ram air inlets.

g. If the airplane is to be parked outside for extended periods, the flaps should be retracted to prevent water from collecting in them.

10.0.2 PARKING PRECAUTIONS. Parking brakes are for temporary use only. Hydraulic pressure from an engine driven pump or from the electric motor driven pump is required to set the parking brake. The anti-skid switch must be OFF and the battery switch ON when the parking brake is set. The normal (No. 1) hydraulic system must be selected, and the No. 2 engine pump or the electric motor driven pump must maintain pressure to keep the BRAKE PRESS LOW warning light off.

After proper installation of the main landing gear chocks, the parking brakes should be released, the operating machinery shut down, and the battery switch set to OFF.

10.0.3 PARKING BRAKE OPERATION. The JetStar parking brake is conventional in that either the right or left brake can be set or released independently. For parking, it is recommended that both be set. The following paragraphs should be reviewed by JetStar operators to ensure familiarity with parking brake operation:

a. A parking brake tee handle on the pilot's side of the main instrument panel controls catches on the brake linkages from the pilot's rudder pedals. Applying the brakes and pulling the tee handle positions the catches to engage parking brake pawls attached to the airplane structure. (Do not push in handle. This movement is automatic upon release.) This holds the rudder pedals, and thus the brakes, in the brakes-on position.

b. Applying additional pressure to the brakes releases the catches from the pawls, and a spring returns the PARK BRAKE handle to the brakes-off position. As the tee handle operates two catches, one on each of the pilot's brake pedal linkages, it is possible to set or release either the left or the right brakes independently. An asymmetrical setting can be detected by noting the relative positions of the right and left rudder pedals.

CAUTION

When releasing the parking brakes prior to taxiing or towing, ensure that both right and left brakes are released. The normal release procedure is to pivot both rudder pedals hard forward until the parking brake handle springs back to the released position. Do not attempt to push the parking brake handle in until it is certain that the parking brake catches have released.

SECTION 10.1

MOORING

Contents Page

MOORING 10-5

PERMISSIBLE WIND VELOCITIES 10-5

MOORING PROVISIONS 10-5

MOORING DURING COLD WEATHER 10-5

FIGURES

10-1. Mooring Provisions 10-7
10-2. Mooring and Evacuation Data 10-8

10.1.1 MOORING. (See figure 10-1.) When calm weather is predicted with little likelihood of change, and the airplane is not to be left unattended for long periods, mooring is not necessary. If strong winds are expected, however, the airplane should be moored. The airplane is not provided with fittings specifically for attaching mooring lines. The main landing gear towing lugs and the nose landing gear structure are used for mooring.

10.1.2 PERMISSIBLE WIND VELOCITIES. Maximum velocities of surface winds which the airplane can withstand while parked or moored depend upon the gross weight of the airplane and field conditions. Figure 10-2 shows a graph of wind velocity versus airplane gross weight for different field conditions. This graph gives limits for wind velocities requiring mooring or evacuating the airplane, as well as data on damage which may be expected if these limits are exceeded.

10.1.3 MOORING PROVISIONS. See figure 10-1 for the recommended methods of mooring the airplane. Observe the following precautions:

a. Head the airplane in the direction from which the highest wind or gusts will come.

b. Moor the airplane away from other airplanes so it will not be damaged due to shifts in mooring.

c. Place the nose wheel in a straight forward-and-aft position.

d. Fill fuel tanks to capacity if time permits.

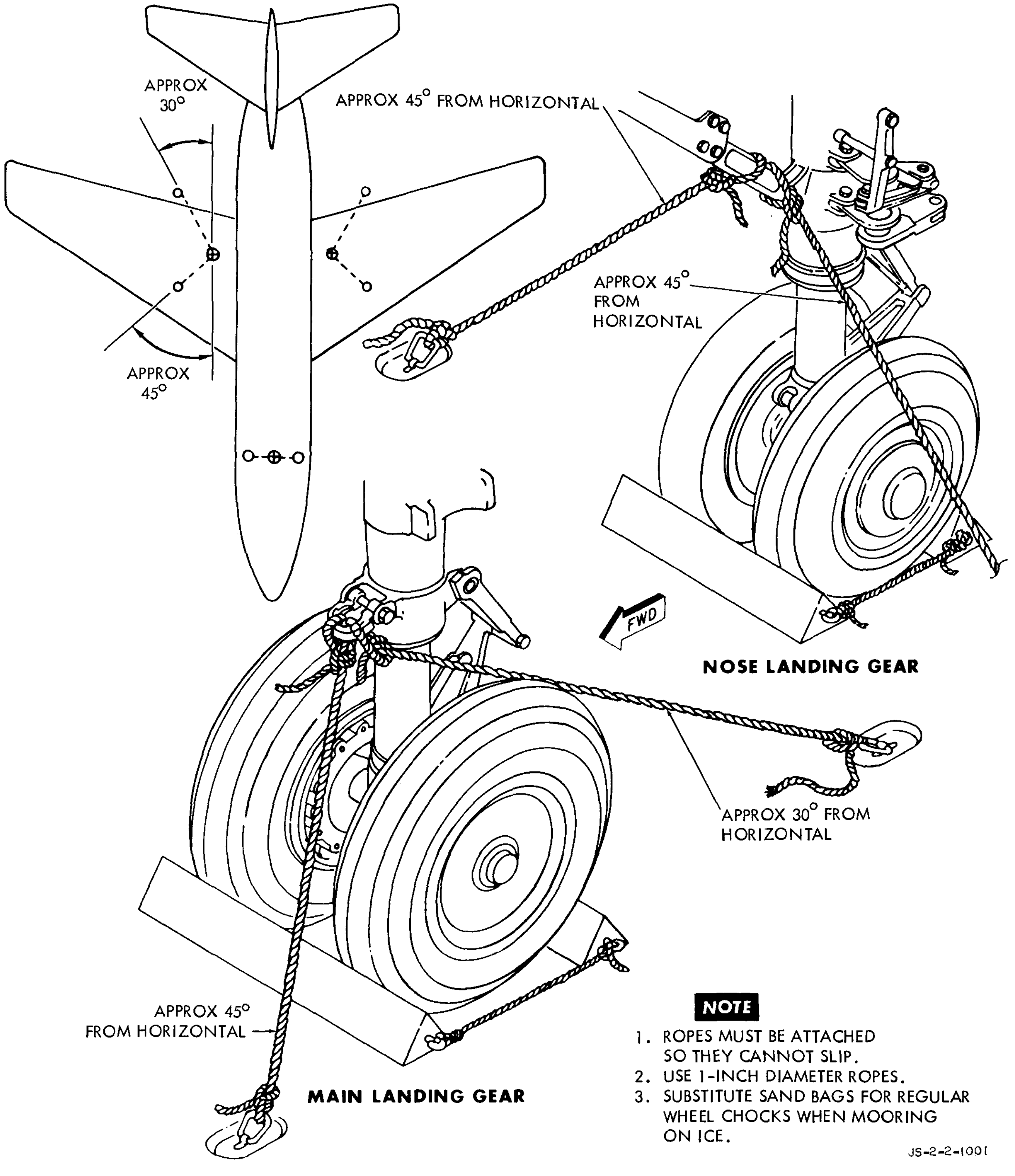
e. Make sure the rudder and the ailerons are in neutral position. Set the trim tabs to zero and raise the leading and trailing edge flaps.

f. Install protective plugs and covers as necessary.

g. If the airplane is going to be moored for an extended period of time, disconnect the battery. Reconnect the battery before moving the airplane.

h. Close all access panels, the entrance door and the speed brake.

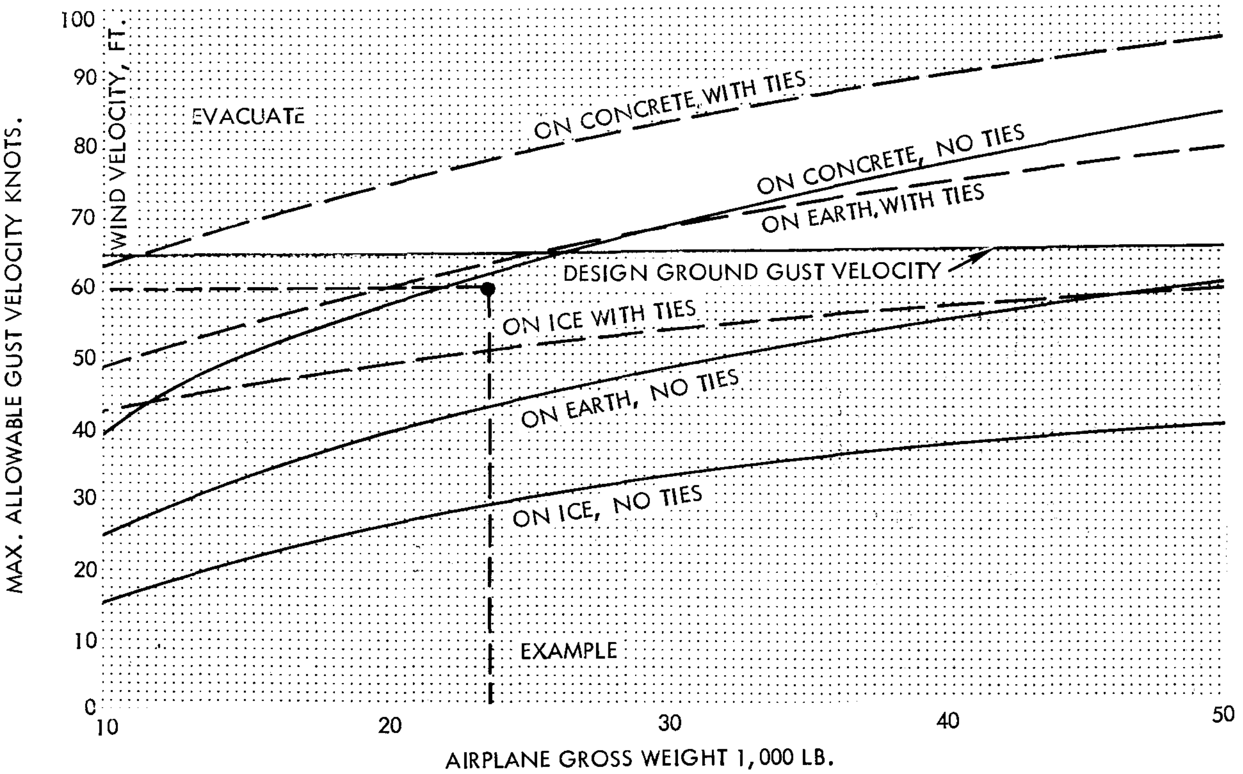
10.1.4 MOORING DURING COLD WEATHER. When mooring the airplane in an extremely cold area, clear the parking area of snow and ice. Place an insulator, such as canvas or heavy building felt, on the ground and roll the tires onto it. This will keep the tires from freezing to the ground.


APPROX 30°
APPROX 45°
APPROX 45° FROM HORIZONTAL
APPROX 45° FROM HORIZONTAL
FWD
NOSE LANDING GEAR
APPROX 30° FROM HORIZONTAL
APPROX 45° FROM HORIZONTAL
MAIN LANDING GEAR
JS-2-2-1001

NOTE

1. ROPES MUST BE ATTACHED SO THEY CANNOT SLIP.
2. USE 1-INCH DIAMETER ROPES.
3. SUBSTITUTE SAND BAGS FOR REGULAR WHEEL CHOCKS WHEN MOORING ON ICE.

Figure 10-1. Mooring Provisions



1. CURVES WITH "TIES" BASED ON ONE INCH MANILA LINE, INCLUDING KNOT FACTOR AND TWO ROPES PER GEAR.

2. USE THIS GRAPH TO DETERMINE THE VELOCITY OF SURFACE WIND GUSTS AT WHICH THE AIRPLANE MUST BE MOORED OR EVACUATED TO A SAFE WEATHER AREA.

3. GUST VELOCITIES WITHIN THE SHADED "EVACUATE" AREA ARE DANGEROUS TO THE AIRPLANE STRUCTURE. IF THESE VELOCITIES ARE EXPECTED, FLY THE AIRPLANE TO A SAFE WEATHER AREA.

4. GUST VELOCITIES BETWEEN THE SHADED "EVACUATE" AREA AND THE LINE MARKED "DESIGN GROUND GUST VELOCITY", MAY CAUSE DAMAGE TO THE CONTROL SURFACES. IF THE AIRPLANE IS SUBJECTED TO GUST VELOCITIES OF THIS VALUE, THOROUGHLY CHECK SURFACE CONTROLS AND POINTS OF ATTACHMENTS BEFORE FLIGHT.

5. GUST VELOCITIES BELOW THE LINE MARKED "DESIGN GROUND GUST VELOCITY" ARE SAFE PROVIDED A PLOT OF THE PARKING AND MOORING CONDITION OF THE AIRPLANE WILL BE BELOW THE LINE DESCRIBING THIS CONDITION ON THE GRAPH.

6. EXAMPLE: YOUR AIRPLANE IS SITTING ON A CONCRETE STRIP UNMOORED. HIGH WINDS ARE EXPECTED WITH GUSTS UP TO 60 KNOTS. YOU ESTIMATE THAT WITH THE PRESENT FUEL AND EQUIPMENT ON BOARD, YOUR AIRPLANE HAS A GROSS WEIGHT OF 24,000 POUNDS. TO FIND THE TYPE OF MOORING REQUIRED, PROCEED AS FOLLOWS:
   - A. START AT 24 ON THE GROSS WEIGHT SCALE ALONG THE BOTTOM OF THE GRAPH AND READ UP UNTIL YOU COME TO THE HORIZONTAL LINE INDICATING 60 KNOTS.
   - B. THIS POINT IS ABOVE THE "ON ICE-NO TIES" "ON ICE-WITH TIES", AND THE "EARTH-NO TIES" LINES INDICATING THAT NONE OF THESE CONDITIONS IS SATISFACTORY. THE POINT IS SLIGHTLY BELOW THE "CONCRETE-NO TIES" LINE INDICATING THAT IF THE ESTIMATED GROSS WEIGHT AND EXPECTED GUST VELOCITY ARE CORRECT, THE AIRPLANE WILL BE SAFE AS IT IS ON THE CONCRETE STRIP WITHOUT TIES. SINCE THE POINT YOU HAVE FOUND IS ONLY SLIGHTLY BELOW THE SAFE LINE, IT WILL BE SAFER TO ADD ANOTHER FOUR OR FIVE THOUSAND POUNDS OF FUEL OR TO TIE THE AIRPLANE DOWN USING THE RECOMMENDED MOORING PROCEDURES.

JS-2-2-1002

Figure 10-2. Mooring and Evacuation Data

SECTION 10.2

STORAGE DURING PERIODS OF INACTIVITY

The requirement for placing the aircraft in long term storage during periods of inactivity and the maintenance of preservation and depreservation are found in various sections of this manual. Listed below are the locations of this information:

SUBJECT	PARAGRAPH
Lubrication During Periods of Inactivity	12.12.2
Air Conditioning System	21.0.6.4
Pressurization System	21.1.7.7
Electronics Cooling System	21.2.6.2
Toilet Ventilation	21.3.4
Electrical Equipment	24.0.5
Fire Protection System	26.4
Control Cables	27.8.2
Fuel Tanks	28.2.4
Hydraulic System	29.0.2.6
De-Icing Boots	30.1.7
Landing Gear	32.0.6
Oxygen	35.5
Airframe	51.10

For engine preservation requirements, see Pratt and Whitney JT12 Maintenance Manual 435107.

Should an aircraft have to be moved in state of preservation, anything disturbed should be represerved as may be required.

CHAPTER 12

SERVICING

TABLE OF CONTENTS

SECTION		PAGE
12.0	GENERAL	12-3
12.1	SERVICING THE FUEL SYSTEM	12-5
12.2	SERVICING THE ENGINE OIL SYSTEM	12-13
12.3	SERVICING THE ENGINE FIRE EXTINGUISHER SYSTEM	12-17
12.4	SERVICING THE HYDRAULIC SYSTEM	12-19
12.5	SERVICING THE BATTERIES	12-27
12.6	SERVICING THE LANDING GEAR	12-31
12.7	SERVICING THE PITOT STATIC SYSTEM	12-35
12.8	SERVICING THE ENVIRONMENTAL SYSTEMS	12-39
12.9	SERVICING THE PITCH TRIM ACTUATOR	12-41
12.10	SERVICING THE OXYGEN SYSTEM	12-43
12.11	SERVICING THE LAVATORY WATER SYSTEM	12-45
12.12	LUBRICATION	12-47
12.13	DELETED	
12.14	EXTERIOR CLEANING	12-153

SECTION 12.0

GENERAL

Contents Page
SERVICING POINTS 12-3
SERVICING HAZARDS 12-3
ADVERSE CLIMATIC CONDITIONS 12-3
EXTERIOR SURFACE CLEANSING 12-3
FIGURES
12-1 Servicing Points 12-4

12.0.1 SERVICING POINTS. Figure 12-1 shows the location of all airplane servicing points. Details of the servicing points are given, when needed, with the servicing procedure for the affected system.

12.0.2 SERVICING HAZARDS. Many hazards, both to the airplane and to maintenance personnel, arise during servicing. To assure safe performance of servicing operations, all of the precautions given with the servicing procedures must be observed, and personnel must be on guard against additional hazards and acts of carelessness which may occur during servicing. Basically, safe operation involves keeping all ground servicing equipment in good repair, good housekeeping, adherence to prescribed servicing procedures, and the use of equipment in accordance with established operating procedures.

12.0.3 ADVERSE CLIMATIC CONDITIONS. Servicing is more difficult in extreme weather. Low temperatures cause metals to contract and rubber to harden. Fluid line joints may develop leaks and require torque readjustments. In hot weather, air pressure in tires and accumulators rises and fluids, especially fuel, evaporate much more rapidly; therefore the danger of explosion is greater. Dry weather and high winds increase the chance of dust and other foreign matter contaminating the systems during servicing. When servicing with fuel in rainy weather, use a canopy to shelter the immediate area around the filler port to keep water out of the fuel system.

12.0.4 EXTERIOR SURFACE CLEANSING. The following procedures can be used for cleaning the airplane:

a. Prior to cleaning:

   1. Static ground the aircraft.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

   2. Insure that all openings are closed and secure.

   3. Mask all openings such as doors, vents, air intakes, exhaust outlets, landing gear areas, and flap track areas.

   4. Mask all materials such as landing gears, tires, windows, radome, antennas (fiberglass), wing tips, and plastics.

b. Cleaning painted or unpainted areas:

   1. Spray or brush apply cleaner solution on surfaces.

Note

Cleaner solution shall be one (1) part detergent concentrate with 8 to 20 (depending upon condition of the surface) parts of water.

   2. Allow solution to soak a minimum of 2 to 3 minutes.

   3. Lightly scrub with brush.

   4. Flush thoroughly with water.

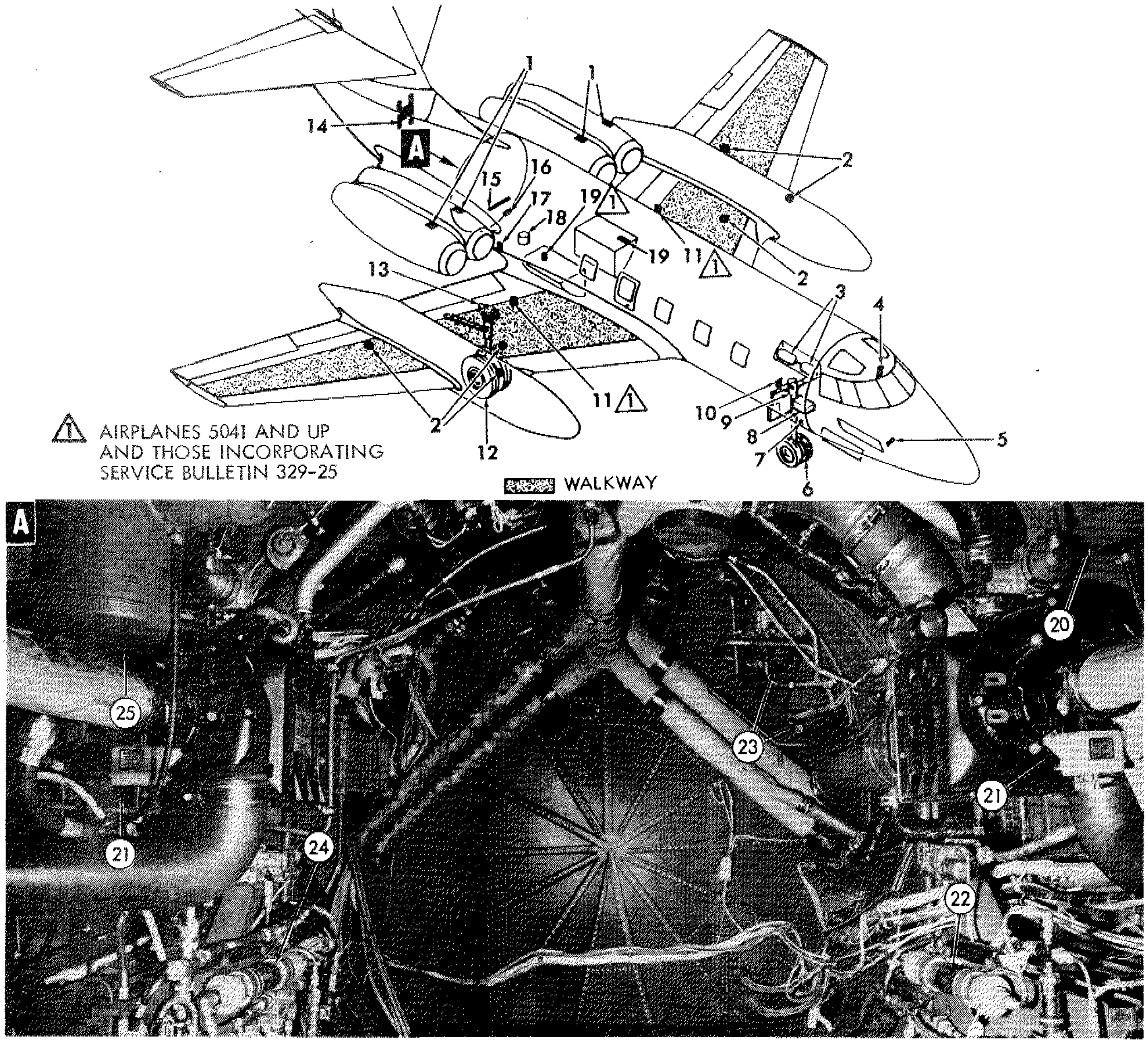
   5. Re-apply solution, as necessary, to obtain a clean surface, and re-rinse.

   6. Direct a stream of dry compressed air to insure removal of all entrapped water and contaminates.

c. Demask and hand clean. Hand clean areas protected in step a.

d. Re-lubricate as necessary.

e. Restore the airplane to normal.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21
24
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1. ENGINE OIL TANK FILLER CAP
2. FUEL FILLER CAP
3. STATIONARY OXYGEN CYLINDERS
4. PORTABLE OXYGEN CYLINDER
5. BRAKE ACCUMULATOR
6. NOSE LANDING GEAR TIRES
7. EXTERNAL ELECTRICAL POWER RECEPTACLE
8. BATTERY
9. NOSE LANDING GEAR STRUT
10. NOSE LANDING GEAR EMERGENCY EXTENSION AIR BOTTLE
11. MAIN LANDING GEAR UPLOCK ACCUMULATOR
12. MAIN LANDING GEAR TIRES
13. MAIN LANDING GEAR STRUT
14. PITCH TRIM ACTUATOR
15. THRUST REVERSER ACCUMULATOR
16. SINGLE POINT DEFUELING VALVE AND FITTING
17. LAVATORY WATER SERVICING PANEL
18. TOILET FACILITIES
19. MAIN LANDING GEAR EMERGENCY EXTENSION AIR BOTTLE
20. STANDBY HYDRAULIC SYSTEM RESERVOIR
21. COOLING TURBINE OIL RESERVOIR
22. STANDBY HYDRAULIC SYSTEM ACCUMULATOR
23. ENGINE FIRE EXTINGUISHER BOTTLES
24. NORMAL HYDRAULIC SYSTEM ACCUMULATOR
25. NORMAL HYDRAULIC SYSTEM RESERVOIR

JS-2-2-1201

Figure 12-1. Servicing Points

SECTION 12.1

SERVICING THE FUEL SYSTEM

Contents Page

GENERAL 12-5
FUEL SERVICING PRECAUTIONS 12-5
STATIC GROUNDING FOR FUELING AND DEFUELING 12-7
REFUELING THE AIRPLANE 12-7
TYPES OF FUEL 12-7
FUEL ADDITIVES 12-8
DEFUELING THE AIRPLANE 12-8
QUICK DRAIN CONTAMINATION CHECK 12-8
ENGINE FUEL FILTER DRAIN 12-8

FIGURES

12-2 **Fuel Servicing** 12-9
12-3 **Fuel Tank Servicing Capacities** 12-10
12-4 Fuel Brand Names 12-11

12.1.1 GENERAL. Fuel system servicing includes all ground operations pertinent to the fueling, defueling, maintenance, and draining of the fuel system.

WARNING

No fuel servicing should be attempted by any individual not thoroughly trained in all applicable established policies and procedures of airplane ground servicing. No servicing should be started before authoritative responsibility has been delegated and realized by all members of the servicing crew. Any carelessness will place life and equipment in jeopardy. Know and observe all safety precautions. Failure to do so could cause explosion and/or fire.

WARNING

Observe the following LEL (lower explosive limit) vapor concentration levels during fuel tank maintenance work. Above 20 percent LEL is not considered "firesafe." Purge tanks with clean air down to 20 percent LEL before removing equipment and opening tank access panels. Between 20 percent and 5 percent LEL enter tanks only with full face, air supplied respirators. Between 5 percent and 1.5 percent LEL use half face, vapor chemical cartridge respirators. Below 1.5 percent LEL plus 19.5 percent oxygen is considered "healthsafe".

12.1.2 FUEL SERVICING PRECAUTIONS. The safety precautions listed below do not include items of preventive maintenance on servicing equipment or the techniques of operating such equipment. All servicing tools and equipment must be in good repair and must be operated only by authorized personnel.

12.1.2 (Cont).

a. Cease all fueling operations whenever a fire hazard appears, no matter how minor the hazard may seem.

b. Locate the airplane outdoors, at least 100 feet from any hangar or building, measured from the fueling point and/or fuel vents.

c. Do not taxi, tow, or operate any airplane within 50 feet of any fuel servicing operation. Do not perform fuel servicing operations within 50 feet of any fuel spill until the spill has been removed and the area rendered safe.

d. Fueling and maintenance of adjacent airplanes must be accomplished more than 50 feet from the fueling point and/or fuel vents on the airplane being serviced.

e. Make sure all airplane static grounding procedures are accomplished before beginning fuel servicing.

f. Do not perform fueling operations during an electrical storm or during servicing of the airplane oxygen system.

g. Permit no unauthorized personnel aboard the airplane or in the immediate vicinity during fueling operations.

h. Any equipment not used in the specific fueling operations must be placed at least 50 feet from the airplane and the fueling source.

i. Fuel servicing hose reels must contain the maximum capacity of hose at all times. Mobile servicing units must be parked at the maximum hose length (never closer than 20 feet) from any part of the airplane being serviced or from any other aircraft or structure.

j. "NO SMOKING" precautions must be observed during all fuel servicing operations. "NO SMOKING AREA" signs must be placed in the outlying limits (normally 50 feet) of the fueling/defueling areas to warn personnel not concerned with fueling operations to stay clear of the area while smoking.

k. A fire extinguisher with a 50-pound minimum capacity of carbon dioxide, or equivalent, must be located strategically within 50 feet of all fueling operations. In addition, a wheeled extinguisher of the type specified must be adjacent to the mobile servicing units. Access to fire extinguishers must not be obstructed.

l. Make sure the airplane radar or any other nearby radar or radio (excluding interphone system) is not operating during fuel servicing operations.

WARNING

There must be no operation of airborne transmitting equipment within a radius of 100 feet, or ground transmitting equipment within 300 feet. This equipment may be placed in STANDBY.

m. During fueling/defueling operations, constantly watch for any signs of fuel leakage, overflow, or seepage. If any fuel escape is noticed, shut down all powered equipment immediately.

n. If a major fuel spill occurs during servicing operations, shut down all powered equipment and cease operation. All personnel must leave the spill area except those stationed around the contaminated area to prevent ignition of the spilled fuel. The installation fire department should be notified immediately and should take charge of rendering the area safe before any further operations, including the movement of the airplane or any ground equipment. If a fire starts, immediately move all nearby airplanes and equipment to a safe area.

o. Perform fuel servicing as soon as possible after each flight or series of flights; this will tend to reduce the amount of moisture condensate from air inside the tanks.

p. To prevent damage to the wing observe all no step areas and use a wing mat or wing socks while walking on the wing.

q. Do not drag the fueling hose over the wing leading edge. The hose can damage the deicing boot.

WARNING

To preclude the possibility of fire or explosion resulting from sparks, make sure all persons removing or installing fuel tank filler caps statically ground their bodies by touching the ground terminal near the tank filler neck.

r. Make sure all snow, ice, frost or water is carefully removed from the areas surrounding each wing filler port. Before inserting the supply hose nozzle, make sure no frost or condensate exists on the interior surfaces of the filler port, and the nozzle is clean and free of snow, frost or water.

s. Fuel servicing should be accomplished at the time of day when the temperature of the fuel is lowest (except in extreme arctic conditions).

t. Thirty minutes after each fuel servicing operation, or 30 minutes after removal of an airplane from a heated shelter into an ambient temperature of 33°F, or lower, drain approximately a quart of fuel from each of the quick drain points on the lower surface of the wing to remove any accumulated moisture condensate.

u. Never add alcohol or ethelyne-glycol to the fuel used in this airplane. (See paragraph 12.1.6 for information on fuel anti-icing additive.)

v. An obstructed fuel vent outlet can cause serious structural damage to the airplane.

12.1.3 STATIC GROUNDING FOR FUELING AND DEFUELING.

a. Attach static ground lines at the forward fuselage ground point located on the lower fuselage center line just aft of the nose cone bulkhead, and at the aft fuselage ground point located on the left side of the fuselage aft of and above the speed brake opening.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

b. Ground the fuel supply unit securely to the same terminal point as the static ground lines from the airplane.

c. In refueling and defueling, before attaching or inserting the nozzle make absolutely certain that the ground wire of the fuel nozzle is securely attached to the ground receptacle. A receptacle is located near each filler opening and near the defueling connection point.

Note

All of the ground receptacles, with the exception of the ones on the external tanks, are the standard jack type receptacles. On the external tanks, use the life line attach bracket as the ground point.

WARNING

Extreme caution should be used when switch fueling (mixing Jet A, Jet A-1 or JP-5 fuel with Jet B or JP-4). Different fuels have different flash and electrostatic characteristics and can produce fuel/air vapors with broader flammability characteristics, thus increasing the chance for ignition.

Switch fueling should always be done at reduced flow rates whenever the fuel being added does not contain the Shell ASA-3 anti-static additive or equivalent. Fueling should be accomplished at approximately half the normal rate. Splashing of fuel within the tank should also be avoided.

12.1.4 REFUELING THE AIRPLANE. Each of the fuel tanks has a separate filler opening and must be filled individually (see figures 12-1 and 12-2). See figure 12-3 for the fuel tank capacities. Refuel the airplane according to the following procedure:

a. Observe all precautions given in paragraph 12.1.2. For partial fuel loading, observe the weight and balance requirements of the Airplane Flight Manual or the Handbook of Weight and Balance Data.

b. Ground the refueling nozzle to the airplane. Use the grounding provisions located near the filler opening.

Note

If a single point refueling truck is used, the flow must be reduced. An adapter is available which can be quickly installed on the standard single point refueling nozzle. It can be ordered from Parker Aircraft Company by the name Fuel Shut-Off Adapter, Part No. F-559. (See figure 12-2.)

c. Remove the filler cap by lifting up the recessed handle and twisting to "OPEN."

Note

The internal fuel tanks should be filled equally prior to filling the external tanks. However, external fuel may be added (not to exceed 4500 pounds total external fuel) with the internal tanks less than full provided the airplane center of gravity is maintained. All fuel should be balanced span-wise upon completion of servicing.

d. Fill each tank through the appropriate filler opening, observing through the opening to determine when the tank is full.

Note

The fuel tank is full when the fuel level reaches the step on the inside diameter of the filler cap adapter.

e. Replace the filler cap and make sure it is securely locked.

Note

On airplanes 5001, 5002, & 5004 thru 5040 not modified by Service Bulletin 329-70, overfill warning installation is provided which prevents reinstalling the filler cap if the tank has been overfilled. If the cap cannot be replaced, drain fuel from the tank until the cap can be installed and locked.

12.1.5 TYPES OF FUEL. The fuels that are acceptable for use in the JetStar are those which conform to Pratt & Whitney Specification PWA522. Fuel cannot be purchased to this specification: however, all ASTM D1655 Jet A, A-1, and

B fuels meet the requirements of PWA522, and fuel can be purchased to ASTM Specifications. As a further aid, figure 12-4 is provided to give the commercial designations and producing companies of fuels which are known to conform to PWA522, or ASTM D1655 Jet A, A-1, or B. Jet A fuel is also identified as NATO F-30 or United Kingdom AVTUR 40. Jet A-1 is NATO F-35 or United Kingdom AVTUR 50. Jet B is MIL-T-5624. Grade JP-4, NATO F-40, or United Kingdom AVTAG.

Note

Because of the low flash and high volatility characteristics of Jet B and JP-4 fuels, it is recommended that preference be given to Jet A, Jet A-1 and JP-5 fuels. Aviation gasoline has not been tested in this airplane and is not to be used.

CAUTION

The presence of even relatively small quantities of TCP results in severe erosion, scaling, and pitting of the first-stage turbine nozzle vanes. Similarly, automobile gasoline is not acceptable because of the common use of TCP and a variety of other undesirable additives.

12.1.6 FUEL ADDITIVES. The use of Phillips PFA-55MB or equivalent (Military Specification MIL-I-27686) anti-icing additive in the fuel is approved and recommended. The quantity of anti-icing additive must not exceed 0.15 percent by volume. The additive must be distributed in proportion to the quantity of fuel in each tank. It must be added to the fuel by agitation or metered into the tank during refueling operations in order to allow the additive to thoroughly mix and blend with fuel. The additive should not be dumped into the tank because it would not mix with the fuel and would cause the tank sealant to deteriorate. The additives for oxidation inhibiting, anti-corrosion, thermal stability, metal deactivation, anti-static, lubricity improvement, biocide, and anti-smoke listed in Pratt and Whitney Service Bulletin No. 2016 may be beneficial if used in the concentrations indicated and properly mixed with the fuel.

12.1.7 DEFUELING THE AIRPLANE. A defueling valve and fitting is provided for single-point defueling of all fuel tanks (see figures 12-1 and 12-2). Defuel the airplane according to the following procedure:

a. Open the access door to the single-point defueling valve and fitting.

b. Remove the cap from the defueling fitting, and attach the defueling hose to the fitting.

c. Manually operate the defueling valve to "OPEN."

d. Open the crossfeed valves on the fuel manifold.

e. Pump the fuel from the tanks by use of either the tank boost pumps (with external electrical power attached to the airplane) or an external suction pump.

f. When fuel flow ceases, either shut down the boost pumps and remove external power from the airplane, or if an external pump was used, shut down the pump.

g. Manually operate the defueling valve to "CLOSED."

h. Remove the defueling hose from the defueling fitting, and reinstall the defueling fitting cap and access cover.

Note

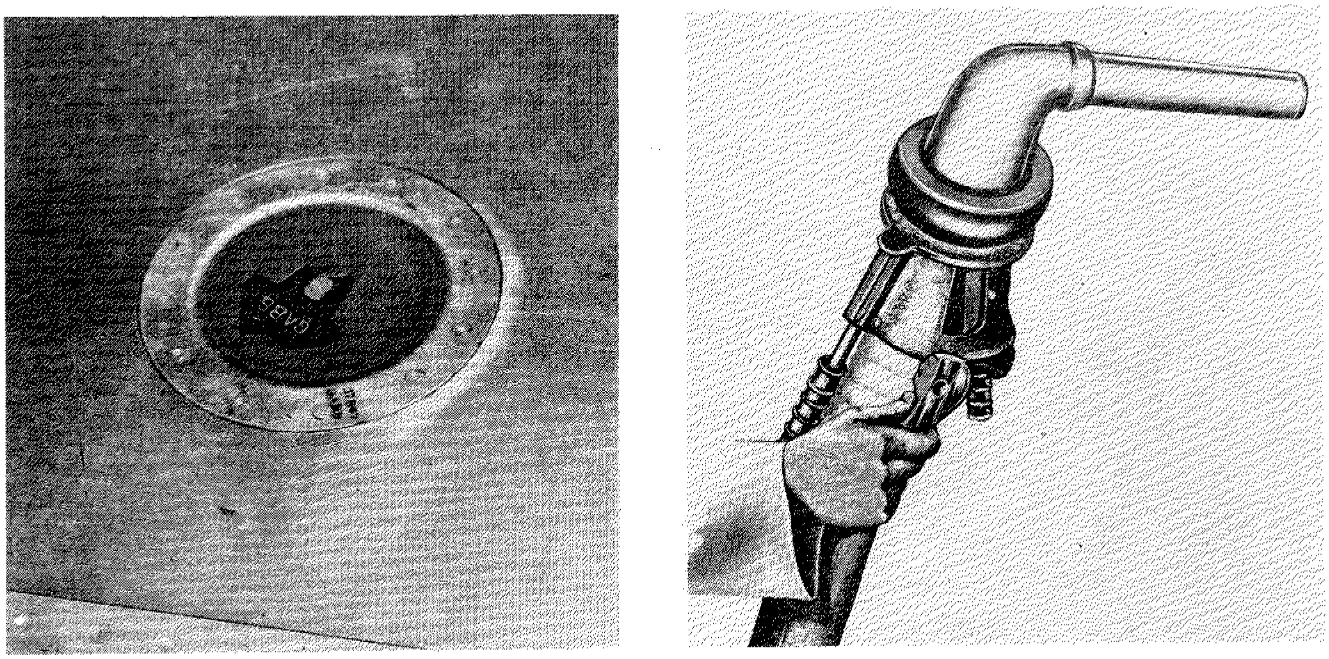
Approximately 10 gallons of unusable fuel remain in each wing tank and 3 gallons in each external tank. This fuel can be removed through the quick drain poppett valves. Approximately 25 gallons of undrainable fuel will remain in the tanks and lines.

12.1.8 QUICK DRAIN CONTAMINATION CHECK. A poppet drain valve is in each fuel tank sump and at the low point of each tank for the purpose of removing moisture accumulations from the tanks. (See figures 12-2 and 28-17.) These flushmounted valves are on the lower surfaces of the wings and external tanks. They are opened by pushing upward with a screwdriver, and are locked by turning a quarter turn. A sump drain container (Lockheed Part No. 11012-MEU) can be used instead of a screwdriver to prevent fuel spillage.

12.1.9 ENGINE FUEL FILTER DRAIN. (See figure 12-2.) The drain valve on the bottom of the engine fuel filter is located at the low point in the engine fuel system. Therefore, any moisture that is in the system will accumulate at this valve where it can be drained. Access to this valve is by the lower nacelle access door. To drain, depress the valve and allow the drainage to collect in a pan.

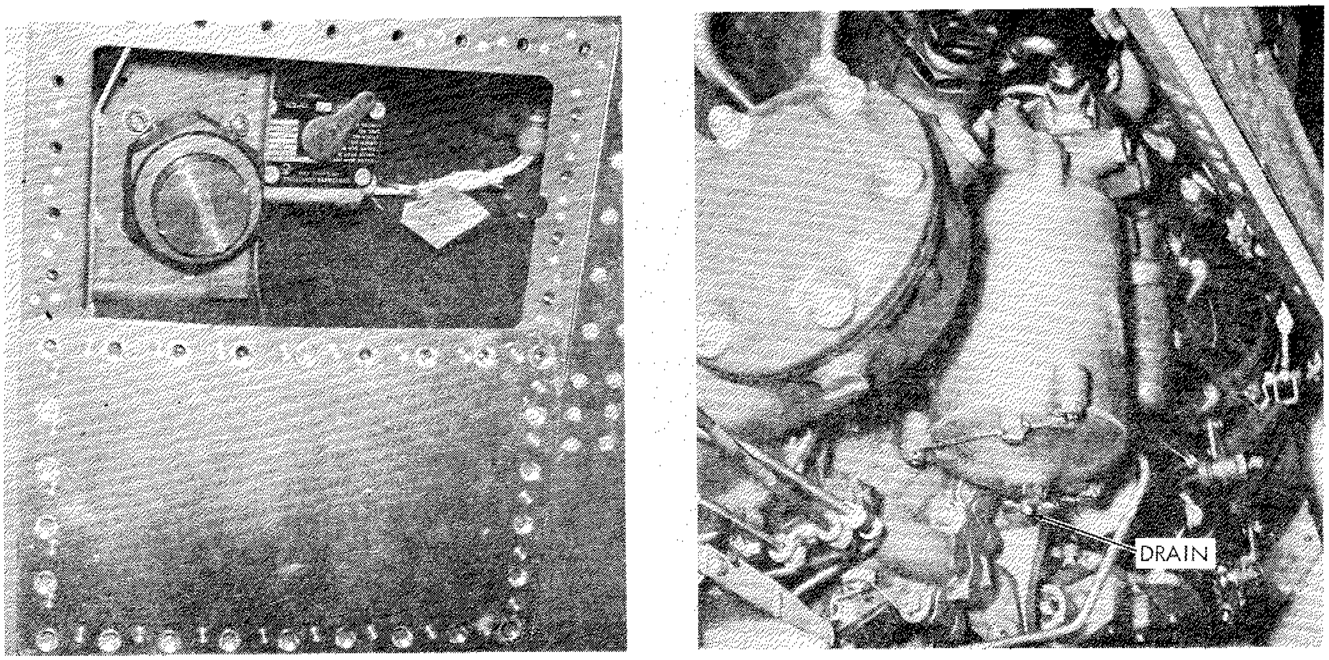
Note

Depress the valve to the full open position and rotate it to lock it open. To close the valve, rotate it in the opposite direction and allow it to close by spring action.



FUEL FILLER CAP

ADAPTER FOR SINGLE POINT REFUELING HOSE



SINGLE POINT DEFUELING FITTING

FUEL LOW PRESSURE FILTER

JS-2-2-1202

Figure 12-2. Fuel Servicing

AIRPLANES 5001, 5002 AND 5004 THROUGH 5040, SERVICE BULLETIN 329-70 NOT INCORPORATED												
TOTAL FUEL (GROUND FUELING)							USABLE FUEL					
Tank	U.S. Gal.	Kero. Lbs. *	JP4 Lb. **	Imp. Gal.	Liters	Expan. Volume U.S. Gal	U.S. Gal.	Kero. Lbs. *	JP4 Lb. **	Imp. Gal.	Liters	Expan. Volume U.S. Gal.
1	371	2486	2412	309	1404	7.4	350	2345	2275	291	1325	7.4
2	388	2600	2522	323	1469	7.7	375	2513	2438	312	1419	7.7
3	388	2600	2522	323	1469	7.7	375	2513	2438	312	1419	7.7
4	371	2486	2412	309	1404	7.4	350	2345	2275	291	1325	7.4
L. Ext.	568	3806	3692	473	2150	11.3	565	3786	3673	470	2139	11.3
R. Ext.	568	3806	3692	473	2150	11.3	565	3786	3673	470	2139	11.3
Total	2654	17784	17252	2210	10046	52.8	2580	17288	16772	2148	9765	52.8

AIRPLANES 5003, 5041 & UP AND THOSE INCORPORATING SERVICE BULLETIN 329-70												
TOTAL FUEL (GROUND FUELING)							USABLE FUEL					
Tank	U.S. Gal.	Kero. Lbs. *	JP4 Lb. **	Imp. Gal.	Liters	Expan. Volume U.S. Gal	U.S. Gal.	Kero. Lbs. *	JP4 Lb. **	Imp. Gal.	Liters	Expan. Volume U.S. Gal.
1	385	2580	2503	321	1457	11.9	375	2513	2437	312	1419	11.9
2	399	2673	2593	332	1510	8.06	390	2613	2535	325	1476	8.06
3	399	2673	2593	332	1510	8.06	390	2613	2535	325	1476	8.06
4	385	2580	2503	321	1457	11.9	375	2513	2437	312	1419	11.9
L. Ext.	568	3806	3692	473	2150	11.3	565	3786	3673	470	2139	11.3
R. Ext.	568	3806	3692	473	2150	11.3	565	3786	3673	470	2139	11.3
Total	2704	18118	17576	2252	10235	62.52	2660	17824	17290	2215	10068	62.52

NOTES:
* Based on 6.7 Lb. per U.S. Gal. kerosene, for ICAO standard day.
** Based on 6.5 Lb. per U.S. Gal. JP4, for ICAO standard day.

Figure 12-3. Fuel Tank Servicing Capacities

OIL COMPANY	PRODUCT NAMES	API AND ASTM DESIGNATION
American Oil	American Jet Fuel Type A American Turbine Fuel American JP-4	JetA Jet A Jet B
Atlantic Richfield	Arcojet A Arcojet A-1 Arcojet B	Jet A Jet A-1 Jet B
BP Trading	BP A.T.K. BP A.T.G.	Jet A-1 Jet B
Caltex Petroleum Corp.	Caltex Jet A-1 Caltex Jet B	Jet A-1 Jet B
Chevron Oil Co. (Standard Oil Co. of Texas Division)	Chevron Jet Fuel A-1 Chevron Jet Fuel A-50 Jet Engine Fuel B	Jet A-1 Jet A Jet B
Chevron Oil Co. (Salt Lake)	Chevron Jet Fuel A-50 Chevron Jet Fuel B	Jet A Jet B
Cities Service Oil	Citgo Turbine A	Jet A
Continental Oil	Conoco Jet-50 Conoco Jet-60 Conoco JP-4	Jet A Jet A-1 Jet B
Gulf Oil	Gulf Jet A Gulf Jet A-1 Gulf Jet B	Jet A Jet A-1 Jet B
Humble Oil - U.S. (affiliate of SONJ)	Esso/Enco Turbo Fuel A Esso/Enco Turbo Fuel A-1 (Bonded only) Esso/Enco Turbo Fuel B	Jet A Jet A-1 Jet B
Esso International - all countries other than U.S. (affiliate of SONJ)	Esso Turbine Fuel A Esso Turbine Fuel A-1 Esso Turbine Fuel B	Jet A Jet A-1 Jet B
Mobil Oil	Mobil Jet A Mobil Jet A-1 Mobil Jet B	Jet A Jet A-1 Jet B
Phillips Petroleum	Philjet A-50 Philjet JP-4	Jet A Jet B
Richfield Division, Atlantic Richfield	Richfield Turbine Fuel A Richfield Turbine Fuel A-1	Jet A Jet A-1
Shell Oil	AEROSHELL Turbine Fuel 640 AEROSHELL Turbine Fuel 650 AEROSHELL Turbine Fuel JP-4	Jet A Jet A-1 Jet B
Sinclair Division, Atlantic Richfield	Sinclair Superjet Fuel Sinclair Superjet Fuel	Jet A Jet A-1
Standard Oil of British Columbia, Ltd.	Chevron Jet Fuel A-50 Chevron Jet Fuel B	Jet A Jet B
Standard Oil - California Standard Oil - Kentucky	Chevron Jet Fuel A-1 Chevron Jet Fuel A-50	Jet A-1 Jet A
Standard Oil (Ohio)	Jet A Kerosene Jet A-1 Kerosene	Jet A Jet A-1
Texaco	Texaco Avjet A Texaco Avjet A-1 Texaco Avjet B	Jet A Jet A-1 Jet B
Union Oil	76 Turbine Fuel Union JP-4	Jet A Jet B

NOTE

BECAUSE OF THE LOW FLASH AND HIGH VOLATILITY CHARACTERISTICS OF TYPE B FUEL, IT IS RECOMMENDED THAT PREFERENCE BE GIVEN TO TYPE A AND A-1 FUELS.

JS-12-0-004

Figure 12-4. Fuel Brand Names

SECTION 12.2

SERVICING THE ENGINE OIL SYSTEM

Contents Page

GENERAL 12-13
OIL TANK CAPACITY 12-13
RECOMMENDED OILS 12-13
SERVICING THE OIL TANK 12-13
DRAINING THE ENGINE OIL SYSTEM 12-14

FIGURES

12-5 Engine Oil Servicing 12-15

12.2.1 GENERAL. Each of the four engines has a separate oil system and must be serviced individually. Check the oil system on each engine for need of servicing after each flight.

12.2.2 OIL TANK CAPACITY. Each of the engines has an oil tank with a useable capacity as shown below.

ENGINE OIL				
Each Engine Oil System	U.S. Gal.	Qts.	Imp Gal.	Liters
Tank Servicing Capacity	1.0	4.0	0.83	3.79
Unusable Oil	0.5	2.0	0.42	1.89
Total	1.5	6.0	1.25	5.68

12.2.3 RECOMMENDED OILS. Service the JT-12 engines with oil which meets Pratt & Whitney Specification PWA-521 (See Pratt and Whitney Service Bulletin No. 238 for additional information). Type I commercial turbine oils which have been approved for use in the JT-12 are listed below.

(a) BP Aero Turbine Oil 15
(b) Castrol 3C Gas Turbine Lubricant

Type II commercial turbine oils which have been approved for use in the JT-12 are listed below:

(a) Esso Turbo Oil 2380/Exxon Turbo Oil 2380
(b) Esso Turbo Oil 25/Exxon Turbo Oil 25
(c) Caltex Starjet 5
(d) Texaco Starjet 5
(e) BP Enerjet 51
(f) Aero Shell Turbine Oil 500
(g) Aero Shell Turbine Oil 555
(h) Castrol 205/Stauffer Jet II
(i) Mobil Jet Oil II
(j) Chevron Jet Engine Oil 5/
Caltex RPM Jet Engine Oil 5
(k) Castrol 5000

WARNING

Type II oil may contain Tricresyl Phosphate (TCP) which may be absorbed through the skin and produces paralysis if taken internally. Appropriate protective measures should be taken to avoid such exposures.

Note

It is not recommended that one brand of oil be mixed with another. If it becomes necessary to use a new brand, drain the system completely and refill with the new brand. The new oil will mix with the small amount of residual oil in the engine without creating any problem.

12.2.4 SERVICING THE OIL TANK.
(See Figure 12-5.)

Note

Prior to all engine shutdowns, run the engine at IDLE for two to five minutes (depending on intensity of prior running), increase to 75 percent rpm for 15 to 30 seconds, then retard the power level to "OFF." This procedure ensures adequate internal cooling and proper scavenging of residual oil.

a. Within one hour of engine shutdown, check the oil level at the tank. If the oil level is low, add oil to bring level up to "FULL" mark.

12.2.4 (Cont).

b. Should a check of the oil show an above normal level, a check for fuel content in the oil should be made. Refer to the Pratt & Whitney Maintenance Manual, paragraph 2-1278-1. The maintenance manual specifies the time limitations for a specific percent of dilution for each action required.

CAUTION

An excessive oil supply may result in fluctuating oil pressure, loss of oil out of breather and buildup of oil tank internal pressure.

Note

Once the engine is serviced following shutdown, it will not be necessary to recheck the oil level until termination of the next flight provided the oil strainer has not been removed.

c. If the engine was not serviced within one hour of shutdown, start the engine and run through the shutdown procedure (two minutes at IDLE and 15 to 30 seconds at 75 percent rpm). Shutdown the engine and check the oil level. If necessary, fill to the "FULL" level.

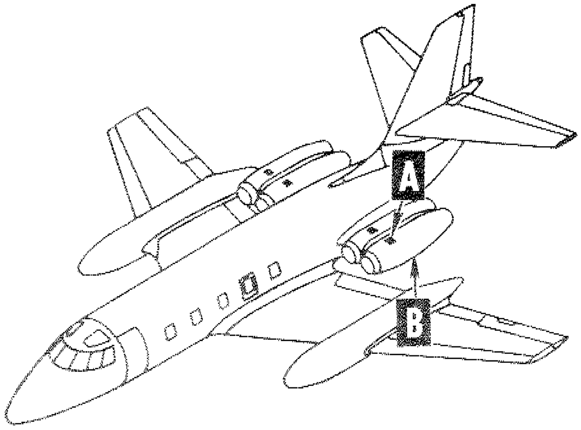
CAUTION

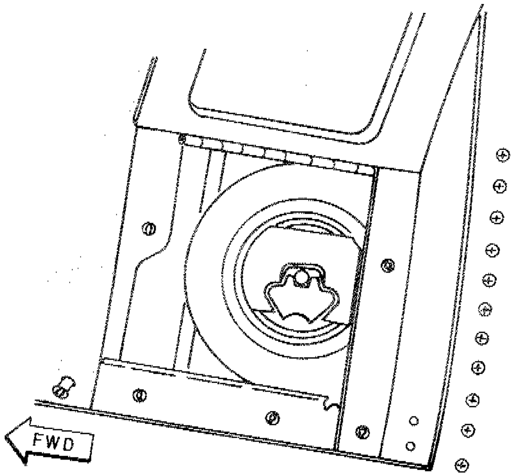
Motoring-over the engine tends to load the bearing compartments with oil. If the engine is motored-over at any time prior to servicing the oil tank, the engine must be started and the shutdown procedure followed before servicing the tank.

Note

The above information applies to engines after incorporation of Pratt & Whitney Campaign Change 63-15. *Prior to this, the* "FULL" mark on the oil tank dip stick is one quart too high. On such engines, the "FULL" level should be considered to be one quart below the position marked.

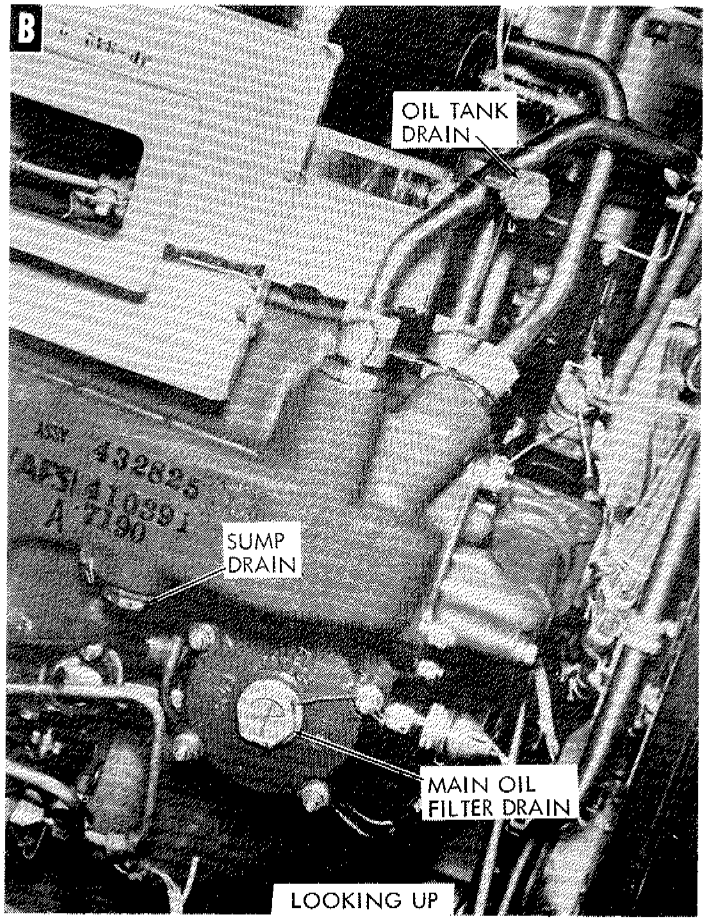
12.2.5 DRAINING THE ENGINE OIL SYSTEM. (See Figure 12-5.) Engine oil may be drained from three points on each engine. The drain plug in the line between the oil tank and the oil pump drains the tank; the drain plug on the gear box drains the sump; and the drain plug on the oil strainer drains the strainer. To drain the system, remove these plugs and catch the oil in a suitable container. When draining is complete, replace the plugs and secure them with safety wire.





FILLING PROVISIONS (TYPICAL ALL ENGINES)

A



DRAIN PROVISIONS (TYPICAL ALL ENGINES)

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Figure 12-5. Engine Oil Servicing

SECTION 12.3

SERVICING THE ENGINE FIRE EXTINGUISHER SYSTEM

Contents Page

BOTTLES 12-17

SQUIBS 12-17

FIGURES

12-6 Engine Fire Extinguisher Servicing Chart 12-18

12.3.1 BOTTLES. There is a pressure gage on each of the two engine fire extinguisher bottles (see figure 12-1). Check the bottle pressure and ambient temperature and plot the resulting point on the graph in figure 12-6. If the point lies between the two curves the bottle is acceptable; if the point lies outside the area bounded by the two curves, remove the bottle for recharging. Send the discharged bottles to a base facility or depot for recharging by qualified personnel. The bottles are to be charged with 6.5 pounds of bromotrifluoromethane (BT) and pressurized with 620 (±40) PSI of dry nitrogen at 70°F. Replace the discharged bottle with a fully charged one.

CAUTION

When replacing a bottle, be sure to disconnect the electrical leads from the terminals at the base of the discharge coupling. Also, before reconnecting the leads to complete the installation of the new sphere, double check the guarded firing switch on the main instrument panel to be sure it is in the "OFF" position.

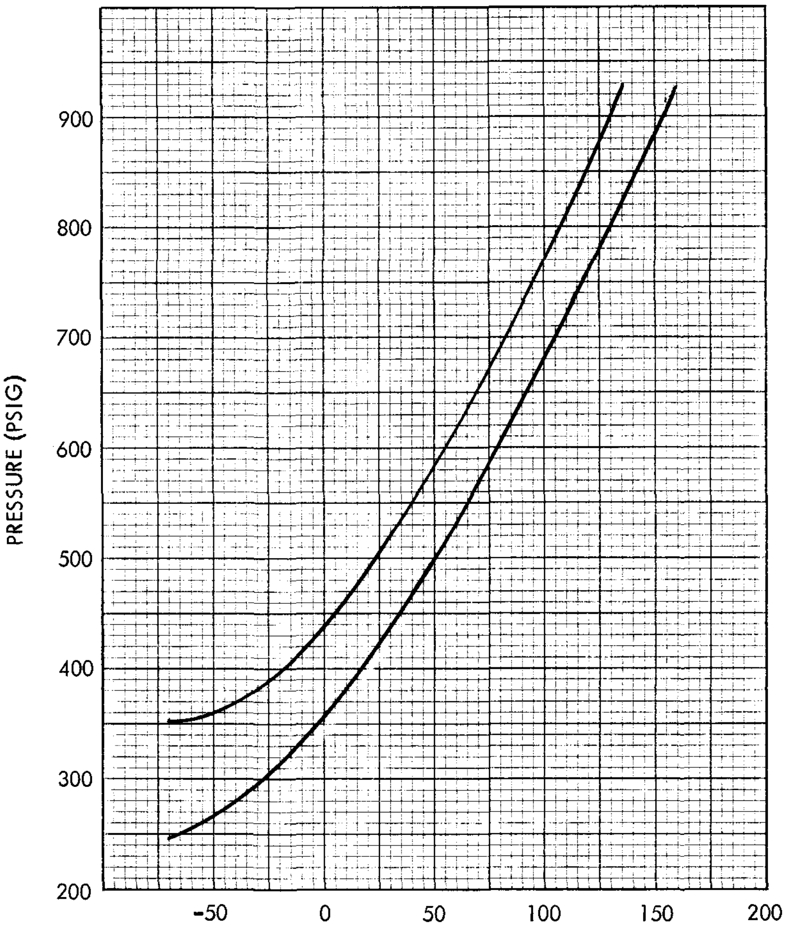
Note

Fire extinguisher bottles are made by several different vendors. When component parts or recharge kits are ordered, the manufacturer's name should be stated or reference should be made to the vendor parts list.

12.3.2 SQUIBS. Check the squibs for continuity. If this check indicates an open circuit, replace the squib with a new one.

WARNING

Do not exceed 35 milliamps for the continuity check on the squibs. Handle the squib with extreme care. Do not hit or drop it or expose it to heat. The squib contains an explosive charge and can be set off by improper handling.



NOTE

TEMPERATURE - PRESSURE CURVES GIVE ACCEPTABLE PRESSURE RANGE (BETWEEN CURVES) FOR CF_3Br SPHERES CHARGED TO 600^{+40}_{-00} PSIG AT 70°F.

VENDOR	ACCESSORY PRODUCTS	HTL INDUSTRIES	WALTER KIDDE
V-BAND COUPLING TORQUE	75 IN - LBS	△1	45 IN - LBS
SQUIB INSTALLATION TORQUE	75 IN - LBS	50 IN-LBS	150 - 200 IN - LBS
RECHARGE KIT VENDOR PART NO.	802996	32000409	803701
SQUIB PART NO.	800545-2	30903835	872627-01
VENDOR RECHARGE INSTRUCTIONS NO	AP108 △2	2241-005	F-40826

△1 FOLLOW INSTRUCTIONS DECALS ON BOTTLES. DO NOT REMOVE VALVE ASSEMBLY WHEN SPHERE IS PRESSURIZED.

△2 VENDOR OVERHAUL INSTRUCTIONS NO. 26-20-75 ALSO AVAILABLE

JS-1-2-X0/5-1206

Figure 12-6. Engine Extinguisher Servicing Chart

SECTION 12.4

SERVICING THE HYDRAULIC SYSTEM

Contents — Page

GENERAL 12-19
  Hydraulic Fluid 12-19
  Suitable Hydraulic Fluids 12-20

FILLING THE NORMAL HYDRAULIC SYSTEM RESERVOIR 12-20

DRAINING THE NORMAL HYDRAULIC SYSTEM RESERVOIR 12-20A

FILLING THE STANDBY HYDRAULIC SYSTEM RESERVOIR 12-20A

DRAINING THE STANDBY HYDRAULIC SYSTEM RESERVOIR 12-20A

HYDRAULIC ACCUMULATORS PRECHARGE PRESSURES 12-20A

CHARGING THE HYDRAULIC ACCUMULATORS 12-20A

BLEEDING THE HYDRAULIC ACCUMULATORS 12-20B

SERVICING CAPACITIES 12-20B

USING EXTERNAL HYDRAULIC GROUND POWER 12-20C

FIGURES

12-7 Hydraulic Reservoir Servicing Points 12-22

12-8 Accumulator Servicing 12-24

12-8A Hydraulic System Schematic Typical Hydraulic Ground Power Unit 12-25

12.4.1 GENERAL. There are two hydraulic systems in this airplane, the NORMAL (#1) system and the STANDBY (#2) system. Each of these systems has its own reservoir and must be serviced separately (see figure 12-7).

12.4.1.1 HYDRAULIC FLUID. Hydraulic fluid conforming to specification MIL-H-5606 is used throughout the hydraulic systems of the JetStar airplanes. Only fluid confirming to revision "B", or subsequent, of the specification should be used.

WARNING

Hydraulic fluid may contain Tricresyl Phosphate (TCP) which may be absorbed through the skin and produces paralysis if taken internally. Appropriate protective measures should be taken to avoid such exposures.

CAUTION

Use of hydraulic fluid conforming to a revision prior to "B" may result in particle contamination in the affected systems.

12.4.1.2 SUITABLE HYDRAULIC FLUIDS.

HYDRAULIC FLUID MIL-H-5606 SUPPLIERS	
SOURCE	**MANUFACTURER'S DESIGNATION**
American Oil & Supply Co. 238-260 Wilson Ave. Newark, NJ 07105	PQ9300 PQ4630 PQ4328 PQ4140 PQ3808 PQ2950 PQ2905 PQ2903 PQ2890 PQ2863
Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave. Suite 202 Irvine, CA 92714	Brayco Micronic 757B Brayco Micronic 756H Brayco Micronic 756G Brayco Micronic 756F Brayco Micronic 756ES Brayco Micronic 756E
Castrol Oils, Inc. Continental Plaza 401 Hackensack Ave. Hackensack, NJ 07601	Castrol Hyspin A
Chevron USA, Inc. 555 Market St. P O Box 3069 San Francisco, CA 94119	Chevron Aviation Hydraulic Fluid E Chevron Aviation Hydraulic Fluid D
Convoy Oil Corp. 1412 N Front St. Philadelphia, PA 19122	Convoy 606
Deutsche Shell AG Uberseering 35 2000 Hamburg 60 West Germany	Aeroshell Fluid 41
ESSO SAF Centre de Recherches Rue du Tromquet, BP 6-76130 Mont Saint Aighan, France	ESSO Fluid Aviation Invarol FJ13
Mobil Oil Corp. 150 E 42nd St. New York, NY 10017	Mobil Aero HFD
MZF Associates 11200 Homedale St. Los Angeles, CA 90049	25606

HYDRAULIC FLUID MIL-H-5606 SUPPLIERS	
SOURCE	**MANUFACTURER'S DESIGNATION**
NYCO S. A. 66 Avenue des Champs-Elysees 51 Rue der Ponthieu 75008 Paris, France	NYCO Hydraunycoil FH 51
Rohm & Haas Co. Independence Mall West Philadelphia, PA 19105	PA 4394
Royal Lubricants Co., Inc. P O Box 518 1 Merry Lane East Hanover, NJ 07936	ROYCO 756E-1 ROYCO 756E ROYCO 756D ROYCO 756C (C730-4) DS-437
Shell International Petroleum Co. Shell Centre London SE 1 7NA, UK	Aeroshell Fluid 41
Stauffer Chemical Co. 380 Madison Ave. New York, NY 10017	Stauffer Aero Hydroil 500
Technolube Products Div. 5814 E 61st St. Los Angeles, CA 90040	Technolube FB003
Texaco Refining & Marketing, Inc. Product & Pricing Group P O Box 52332 Houston, TX 77052	Aircraft Hydraulic Oil 15

12.4.2 FILLING THE NORMAL HYDRAULIC SYSTEM RESERVOIR. (See figure 12-7.) Before filling the reservoir, make sure that all engines and APU (if installed) are shut down, the landing gear is down and the main gear inboard doors are open. (On airplanes 5093 and subsequent, and those having Service Bulletin 329-238 incorporated, the main landing gear inboard doors are closed.) In addition, before filling the reservoir, make sure that the speed brake is open, the thrust reversers retracted (normal), and accumulators are fully charged with air, and that the system pressure is zero. Fill the reservoir by the following procedure when the fluid level falls below the yellow dot on the reservoir indicator. On airplanes equipped with tube type sight gages, fill the reservoir when the fluid reaches the indicated refill level.

Note

On some airplanes there are two indicators on the reservoir, one above the other. Only the upper indicator should be used.

a. Open the bleeder nut or bleeder valve above the reservoir to relieve the reservoir pressure.

b. Remove the filler cap.

c. Fill the reservoir to overflowing with Specification MIL-H-5606 hydraulic fluid.

Note

Approximately 2 pints are required to fill reservoir from the center of the dot on the indicator to the overflow level.

d. Replace the filler cap.

e. On airplanes 5046 through 5092 not having Service Bulletin 329-238 incorporated, an auxiliary hydraulic reservoir is connected in series with the normal system reservoir on the left side of the ship. This reservoir is serviced by means of the filler port on the normal system reservoir.

Note

A sight glass is incorporated in the auxiliary reservoir to allow a visual check of hydraulic fluid. This sight glass is not a servicing level.

f. Close the bleeder nut or bleeder valve.

12.4.3 DRAINING THE NORMAL HYDRAULIC SYSTEM RESERVOIR.

(See figure 12-7.)

a. Reduce the system pressure to zero.

b. Open the bleeder nut or bleeder valve above the reservoir.

c. Open the drain valve below the reservoir and catch the fluid in a suitable container.

d. Close the bleeder nut or bleeder valve.

e. Close the drain valve.

12.4.4 FILLING THE STANDBY HYDRAULIC SYSTEM RESERVOIR. (See figure 12-7.) Before filling this reservoir, be sure that all engines and APU (if installed) are shut down, the system accumulator is fully charged with air and that the system pressure is zero. Fill the reservoir by the following procedure when the fluid level falls below the yellow dot on the reservoir indicator. On airplanes equipped with tube type gages, fill the reservoir when the fluid reaches the indicated refill level.

a. Open the bleeder nut or bleeder valve above the normal hydraulic system reservoir to relieve the pressure in the reservoir.

b. Remove the filler cap.

c. Fill the reservoir to overflowing with Specification MIL-H-5606 hydraulic fluid.

Note

Approximately 1 pint is required to fill the reservoir from the center of the dot on the indicator to the overflow level, on airplanes 5001 through 5045. On airplanes 5046 and up, 2 pints are required.

d. Replace the filler cap.

e. Close the bleeder nut or bleeder valve.

12.4.5 DRAINING THE STANDBY HYDRAULIC SYSTEM RESERVOIR.

(See figure 12-7.)

a. Reduce the system pressure to zero.

b. Open the bleeder nut or bleeder valve above the normal hydraulic system reservoir.

c. Open the drain valve below the reservoir, and catch the fluid in a suitable container.

d. Close the bleeder nut or bleeder valve.

e. Close the drain valve.

12.4.6 HYDRAULIC ACCUMULATORS PRECHARGE PRESSURES. The hydraulic accumulators and their correct precharge pressures are listed below. (See figure 12-1 and figure 12-8.)

a. Normal hydraulic system accumulator - 1700 PSI.

b. Standby hydraulic system accumulator - 1700 PSI.

c. Thrust reverser accumulator - 1200 PSI.

d. Brake system accumulator - 800 PSI.

e. Main landing gear accumulator (2 each) - 500 PSI (airplanes 5041 and subsequent and those having Service Bulletin 329-25 incorporated).

12.4.7 CHARGING THE HYDRAULIC ACCUMULATORS. Before charging these accumulators, be sure the pressure on both the normal and standby hydraulic systems is zero, otherwise the accumulators cannot be correctly serviced. With both hydraulic systems at zero pressure, charge the accumulators with dry air to the correct precharge pressure given above. Use the following procedure:

Note

The brake system and thrust reverser accumulators both have check valves in the pressure supply lines and do not deplete pressure when the hydraulic system is shut down. Reduce this pressure to zero by operation of the brakes and thrust reversers prior to accumulator precharging.

a. Remove the valve cap.

b. Attach the pressure source to the valve.

c. Loosen the hexagonal swivel nut about a quarter turn counterclockwise and apply pressure.

WARNING

There are two hexagonal nuts on the valve assembly. Do not turn the nut that has the safety wire attached. This nut loosens the valve assembly, which can blow out explosively if loosened while pressure is on the unit.

d. When the desired pressure is obtained, retighten the hexagonal nut, reduce the pressure source to zero, and disconnect the pressure source.

WARNING

Do not remove the pressure source until the hexagonal nut is retightened and the pressure source is cut off.

e. Replace the valve cap.

12.4.8 BLEEDING THE HYDRAULIC ACCUMULATORS. Before bleeding the air charge from hydraulic accumulators, ensure that the pressure on both the normal and standby hydraulic systems is zero. Bleed using the following procedure:

Note

The brake system and thrust reverser accumulators both have check valves in the pressure supply lines and do not deplete pressure when the hydraulic system is shut down. Reduce this pressure to zero by operation of the brakes and thrust reversers prior to accumulator bleeding.

a. Remove the air valve cap.

b. Rotate the hexagonal swivel nut slowly counterclockwise until air begins to bleed from the valve, and allow pressure to bleed slowly from the unit.

WARNING

There are two hexagonal nuts on the valve assembly. Do not turn the nut that has the safety wire attached. Turning this nut will loosen the entire assembly, which may blow out explosively if loosened while pressure is in the unit. Also do not expose any part of the body to the stream of bleed air. This air is expelled with considerable force and can easily cause bodily injury.

c. Allow the air to bleed completely from the accumulator.

d. Rotate swivel nut to full open position.

12.4.9 SERVICING CAPACITIES.

HYDRAULIC FLUID MIL-H-5606 (RED)								
	AIRPLANES 5001 THROUGH 5045				AIRPLANES 5046 THROUGH 5092			
	U.S. GAL.	QTS	IMP GAL.	LITERS	U.S. GAL.	QTS	IMP GAL.	LITERS
NO. 1								
RESERVOIR SERVICING CAPACITIES	1.8	7.2	1.5	6.8	2.5	10.0	2.1	9.1
AIR SPACE	0.8	3.2	0.7	3.0	0.8	3.2	0.7	3.0
SYSTEM CAPACITY	5.3	21.2	4.4	20.0	6.0	24.0	5.0	22.7
NO. 2								
RESERVOIR SERVICING CAPACITIES	0.6	2.3	0.5	2.2	1.8	7.2	1.5	6.8

HYDRAULIC FLUID MIL-H-5606 (RED)								
	AIRPLANES 5001 THROUGH 5045				AIRPLANES 5046 THROUGH 5092			
NO. 2 (Continued)	U.S. GAL.	QTS	IMP GAL.	LITERS	U.S. GAL.	QTS	IMP GAL.	LITERS
AIR SPACE	0.1	0.5	0.1	0.5	0.8	3.2	0.7	3.0
SYSTEM CAPACITY	2.3	9.3	1.9	8.8	3.5	14.2	2.9	13.4
	AIRPLANES 5093 AND UP △1							
NO. 1								
RESERVOIR SERVICING CAPACITIES	2.6	10.3	2.1	9.7				
AIR SPACE	0.8	3.2	0.7	3.0				
SYSTEM CAPACITY	6.0	24.2	5.0	22.9				
NO. 2								
RESERVOIR SERVICING CAPACITIES	1.8	7.2	1.5	6.8				
AIR SPACE	0.8	3.2	0.7	3.0				
SYSTEM CAPACITY	3.5	14.2	2.9	13.4				

△1 **ALSO AIRCRAFT WITH SERVICE BULLETIN 329-238 INCORPORATED**

12.4.10 USING EXTERNAL HYDRAULIC GROUND POWER. There are external hydraulic power connections for each hydraulic system. These connections are described in Chapter 29. The external hydraulic power unit should have the capability of supplying 10 GPM fluid at 3000 PSIG. (See Figure 12-8A.)

CAUTION

Use only MIL-H-5606 (red) hydraulic fluid for servicing the airplane hydraulic systems. Other type hydraulic fluids (specifically Skydrol) contain agents that will damage the hydraulic systems seals and these type fluids should never be used.

Note

The following procedure is for general guidance only. An experienced operator should be available and the power unit's manufacturer's instructions should be followed.

a. Ensure that the ground power unit contains MIL-H-5606 (red) hydraulic fluid.

b. Sample and analyze the hydraulic reservoir for possible water contamination in accordance with instructions in Chapter 29 when the following conditions occur.

  (1) When the ground power unit has not been used for an appreciable period of time.

  (2) When the ground power unit has been exposed to environmental conditions that would result in water condensation in the hydraulic reservoir.

WARNING

The ground power unit shall be located at a sufficient distance from the airplane to provide adequate ventilation and heat dissipation.

c. Ensure that the ground power unit is located as far from the airplane as the hoses will permit without strain.

d. Set parking brakes securely.

e. Open all necessary access doors.

f. Check that the hydraulic fluid level of the power unit is adequate. Add new fluid if required.

g. Check fuel gauge, radiator level, and engine oil level in engine-driven units. Ensure that levels are adequate for anticipated operating period. Check power connections in electric-powered units for correct phasing and frequency.

h. Check unit hydraulic accumulator pressure if unit is so equipped. If required, charge accumulator to required pressure.

i. Check that the pointers of all gauges rest at or near zero.

j. Clean and connect service ends of the external pressure and return line hoses to the hose storage (recirculation) manifold provided on the unit. If manifold is equipped with shutoff valves, place valves in open position.

k. Start power unit engine (or motor) in accordance with manufacturer's operating instructions. Allow engine to warm up to normal operating temperature.

Note

Steps l. through q. clean and deaerate the hydraulic fluid in the hydraulic power unit. Both operations are performed concurrently.

l. Set up power unit to provide fluid flow from the internal reservoir through the external service hoses and interconnecting manifold. Place pressure compensator at its lowest setting and ensure that the manifold and service outlet valves (if present) are in the open position. The high pressure gauge should indicate a value less than 600 PSI.

m. Allow test stand to recirculation clean for 3 to 5 minutes. Monitor fluid temperature throughout the cleaning cycle and ensure that maximum operating limits are not exceeded.

n. Monitor all filter differential pressure indicators. If indications of a loaded filter are observed after fluid reaches normal operating temperature, shut down power unit and have replacement filter elements installed.

o. On power units equipped with a fluid sight glass and manual air bleed valve, operate valve and monitor sight glass periodically throughout cleaning cycle to eliminate visible indications of entrapped air.

p. Upon completion of recirculation cleaning and deaeration, terminate fluid flow to the external service hoses in preparation for connecting to airplane. Disconnect service hoses from manifold assembly and reinstall manifold dust covers.

q. Shut down power unit prior to connecting hoses to airplane.

CAUTION

Ensure that the power unit pressure hose connects with the 1/2 inch tube size quick-disconnect labeled "PRESS LINE" and that the return hose connects with the 3/4 inch tube size quick-disconnect labeled "SUCTION LINE." An improper hookup may overpressurize the airplane return system.

r. Connect hydraulic power unit service hoses to the airplane external power connection quick-disconnects, ensuring that all connectors are clean prior to connection. Mate all attached dust covers and plugs to protect against their contamination during power unit operation.

s. Check airplane reservoir level in accordance with instructions on placard located above reservoir. Fill if required in accordance with procedure in this chapter.

t. Set up power unit to provide for either airplane or power unit reservoir operation. Required mode of operation can be obtained by use of the reservoir selector valve on stands so equipped or by use of the reservoir fluid supply valve. Closure of the power unit reservoir supply valve will enable airplane reservoir operation.

Note

Use of hydraulic power unit reservoir is desirable in that the vented reservoir will enable airplane fluid deaeration during system operation. This mode should be employed wherever practical.

u. Start power unit and allow unit to warm up with controls set for bypass fluid flow.

v. Adjust flow rate and operating pressures to required values by means of the volume and pump compensator controls. Set emergency relief valve (if so equipped) to the desired operating pressure plus 10 percent.

Note

Bypass control should be fully closed during airplane operation. Adjust operating pressure using pump compensator control only.

w. Hydraulic power unit is now ready to provide powered operation of the airplane hydraulic system. Use specific checkout procedures as provided in appropriate chapters of this manual.

Note

Steps x. through ab. provide items which must be checked during operation of the power unit.

x. Periodically check condition of system fluid through sight glass. If evidence of air is seen, the system should be bled at the power unit until fluid appears clear. It may also be necessary to conduct bleeding procedures on the airplane system being operated. (Refer to appropriate section of this manual.)

Note

In some cases, loaded filter indicators may extend due to cold starting conditions. Reset and continue to monitor indicator until the equipment reaches the normal operating temperture.

y. Monitor filter differential pressure indicators. Should a loaded filter be indicated, shut down power unit and have replacement filter elements installed.

z. If fault indicators light, shut down unit and have problem located and corrected.

aa. If an emergency arises (ruptured hydraulic hose in airplane, for example), open the bypass valve to relieve pressure and stop flow of hydraulic fluid to airplane.

ab. Stop engine immediately if any engine parts fail. Heed warning signs such as sudden drop in engine oil pressure or any unusual engine noise.

Note

Steps ac. through ag. give shutdown procedure. Reverse flow of fluid through airplane filters may result if proper shutdown procedures are not employed.

ac. Leave bypass valve in closed position. Reduce volume setting to zero and adjust pressure compensator to minimum. Allow several minutes for stored pressure in aircraft to bleed off via normal internal leakage. Alternately, on stands so equipped, pressure and return line shutoff valves may be closed instead of reducing the volume and the pressure compensator.

ad. Slowly open pressure bypass valve.

ae. Let engine run at 1000 rpm for about 5 minutes (engine driven models only).

af. *Push throttle down completely.*

ag. Place panel light switch in OFF position.

CAUTION

Do not drag hose ends on apron or floor or otherwise expose to contamination.

ah. Remove external hoses from airplane ground service connections and connect loose ends to hose storage manifold disconnects on power unit. Install all dust caps and plugs, including those at aircraft quick-disconnects.

ai. Close all access doors to protect instruments and controls. Close airplane ground service door.

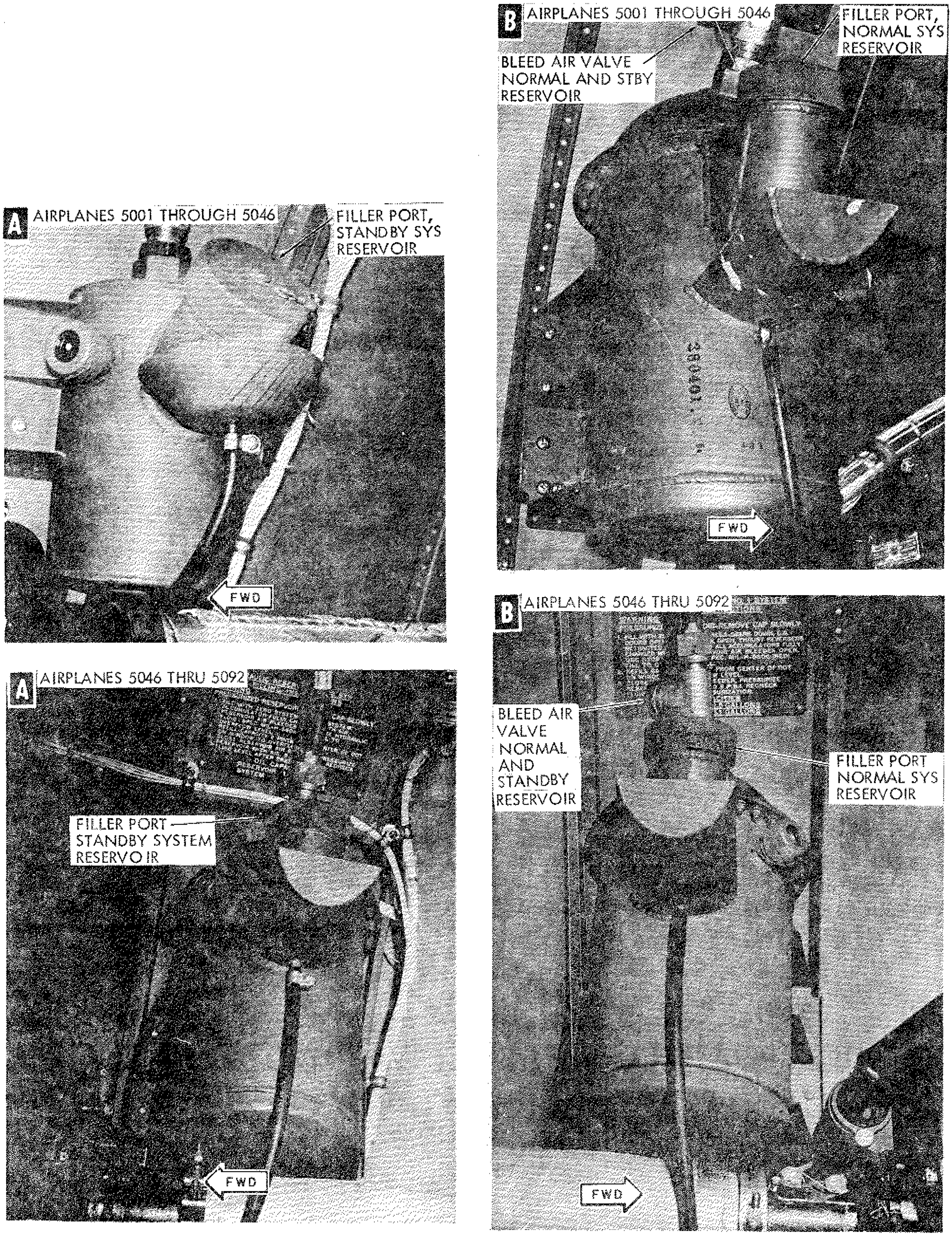
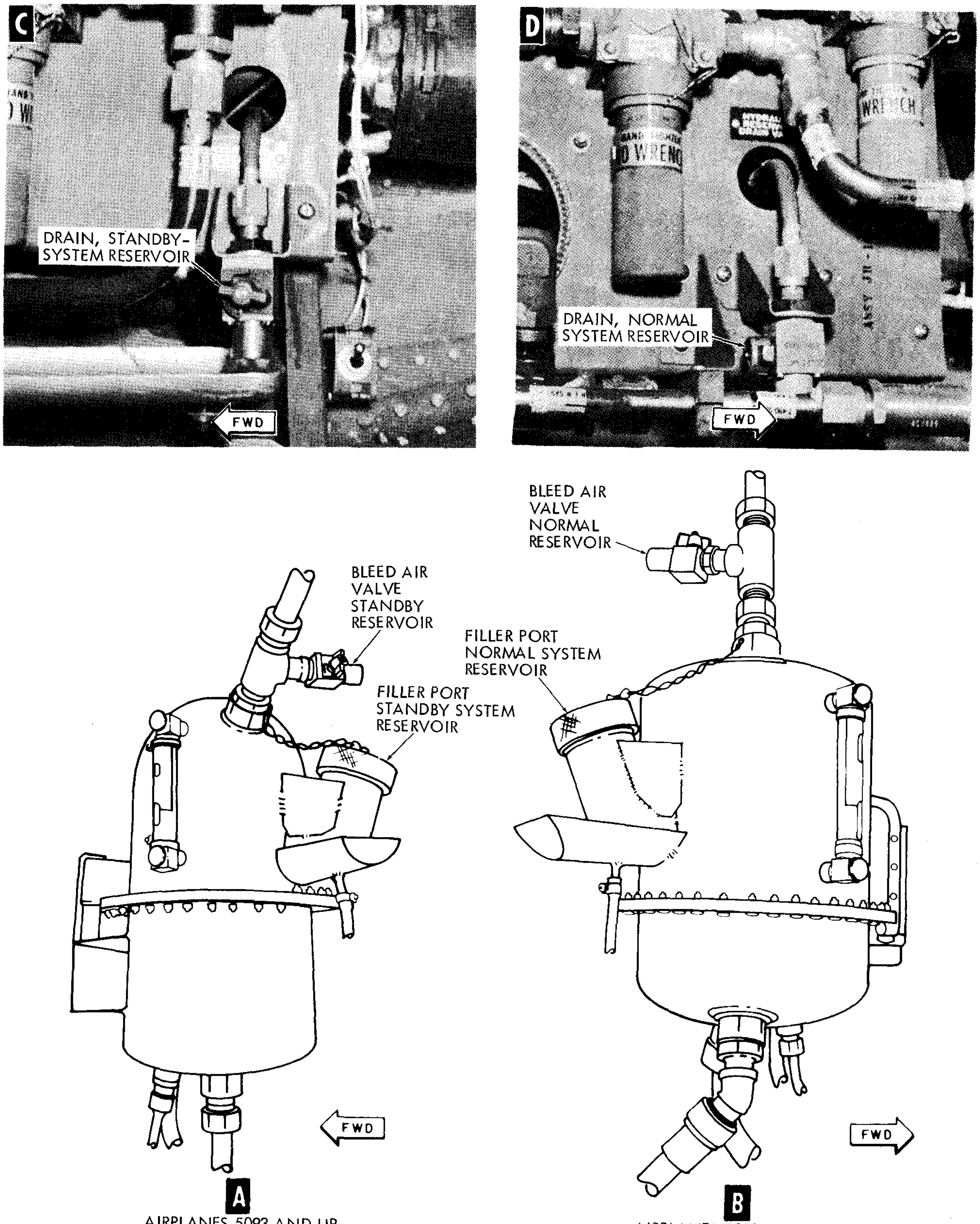

A AIRPLANES 5001 THROUGH 5046
FILLER PORT, STANDBY SYS RESERVOIR
FWD
A AIRPLANES 5046 THRU 5092
FILLER PORT STANDBY SYSTEM RESERVOIR
FWD
B AIRPLANES 5001 THROUGH 5046
FILLER PORT, NORMAL SYS RESERVOIR
BLEED AIR VALVE NORMAL AND STBY RESERVOIR
FWD
B AIRPLANES 5046 THRU 5092
BLEED AIR VALVE NORMAL AND STANDBY RESERVOIR
FILLER PORT NORMAL SYS RESERVOIR
FWD

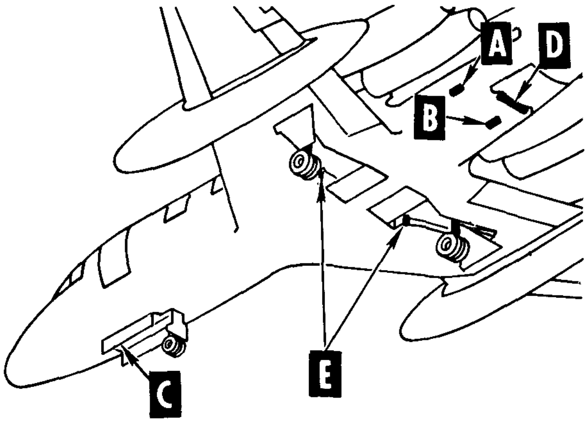
Figure 12-7. Hydraulic Reservoir Servicing Points (Sheet 1 of 2)

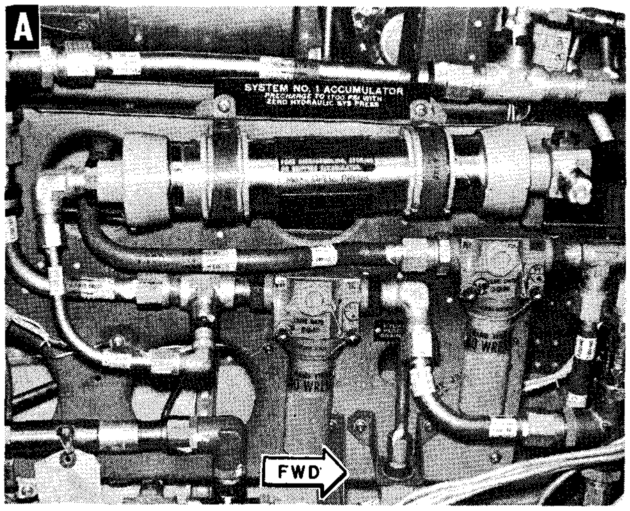

C
DRAIN, STANDBY-SYSTEM RESERVOIR
FWD
D
DRAIN, NORMAL SYSTEM RESERVOIR
FWD
BLEED AIR VALVE NORMAL RESERVOIR
BLEED AIR VALVE STANDBY RESERVOIR
FILLER PORT NORMAL SYSTEM RESERVOIR
FILLER PORT STANDBY SYSTEM RESERVOIR
FWD
FWD
A
B

AIRPLANES 5093 AND UP

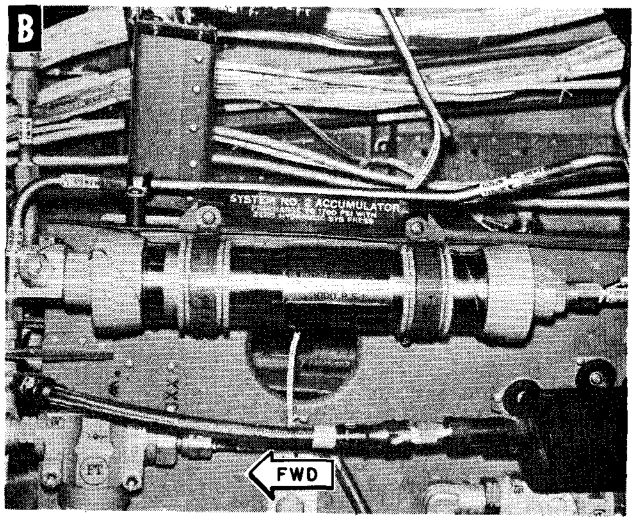
AIRPLANES 5093 AND UP

Figure 12-7. Hydraulic Reservoir Servicing Points (Sheet 2 of 2)



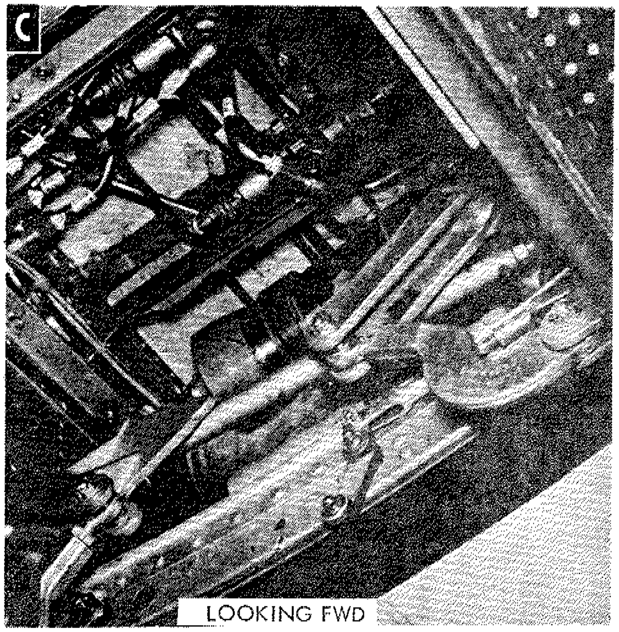


NORMAL HYDRAULIC SYSTEM ACCUMULATOR

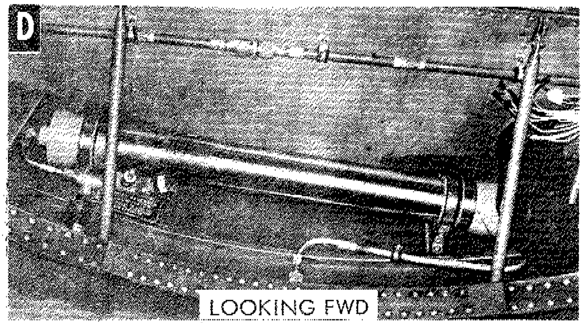


STANDBY HYDRAULIC SYSTEM ACCUMULATOR

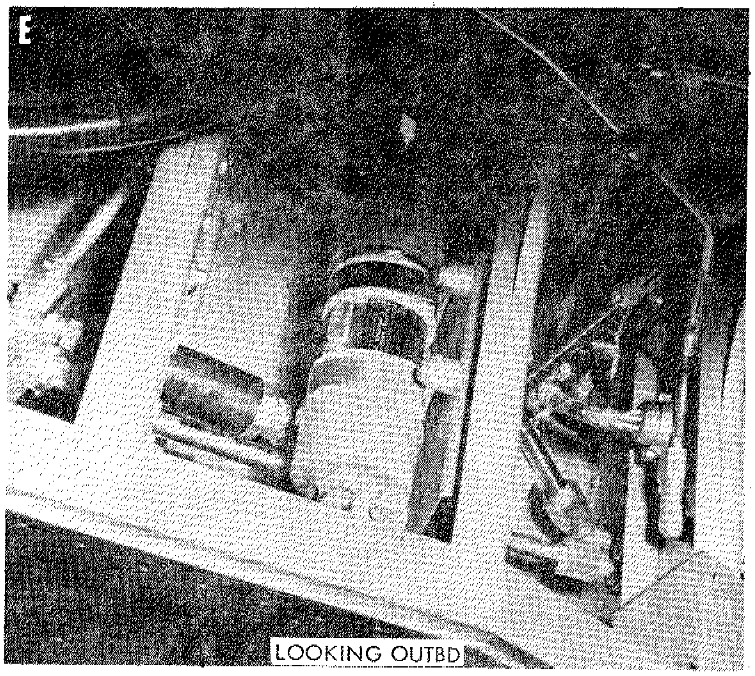
Figure 12-8. Accumulator Servicing (Sheet 1 of 2)



BRAKE SYSTEM ACCUMULATOR

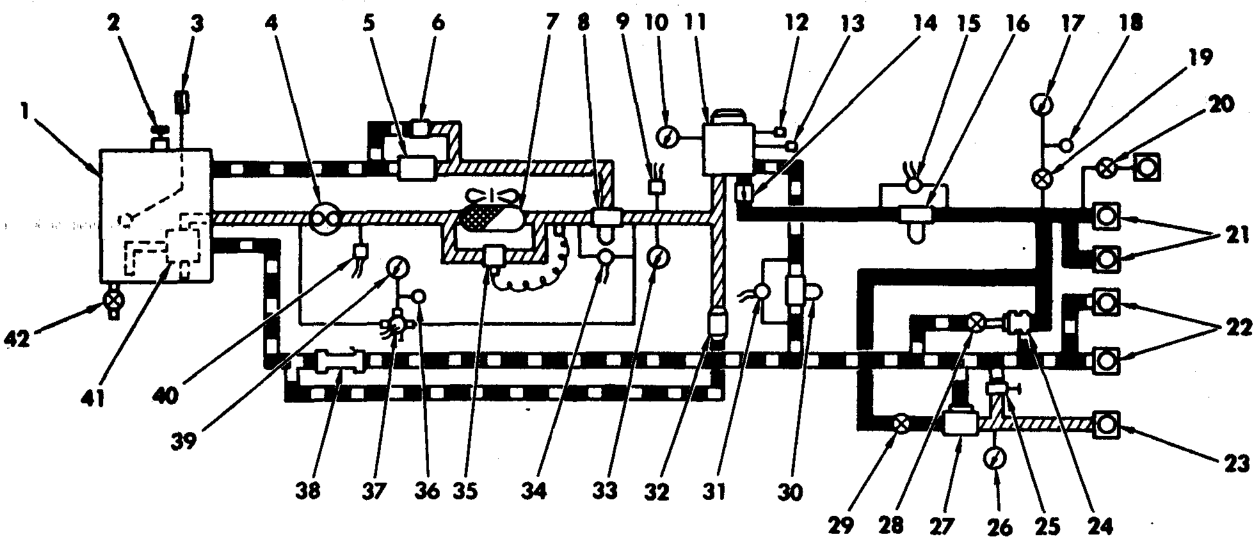


THRUST REVERSER ACCUMULATOR



LANDING GEAR ACCUMULATORS

Figure 12-8. Accumulator Servicing (Sheet 2 of 2)



LEGEND

▨ LOW PRESSURE

▬ RETURN

■ HIGH PRESSURE

1. FLUID RESERVOIR
2. RESERVOIR FILL AND VENT
3. FLUID LEVEL GAGE
4. BOOST PUMP
5. AIR BLEED VALVE
6. THERMAL RELIEF VALVE
7. OIL COOLER
8. LOW PRESSURE FILTER
9. THERMOSWITCH
10. FLOW INDICATOR
11. HIGH PRESSURE PUMP
12. PUMP VOLUME CONTROL
13. PRESSURE COMPENSATOR CONTROL
14. CHECK VALVE
15. PRESSURE DIFFERENTIAL SWITCH
16. HIGH PRESSURE FILTER
17. HIGH PRESSURE GAGE
18. GAGE TEST CONNECTION
19. GAGE SHUT-OFF VALVE
20. FLUID SAMPLING VALVE
21. HIGH PRESSURE DISCONNECT
22. RETURN DISCONNECT
23. FILL SYSTEM DISCONNECT
24. HIGH PRESSURE RELIEF VALVE
25. FILL SYSTEM RELIEF VALVE
26. FILL SYSTEM PRESSURE GAGE
27. PRESSURE REDUCING VALVE
28. HIGH PRESSURE BY PASS
29. FILL SYSTEM SHUT-OFF
30. PUMP CASE FILTER
31. PRESSURE DIFFERENTIAL SWITCH
32. LOW PRESSURE RELIEF VALVE
33. FLUID TEMPERATURE GAGE
34. PRESSURE DIFFERENTIAL SWITCH
35. OIL TEMPERATURE CONTROLLER
36. GAGE TEST CONNECTION
37. GAGE SELECTOR VALVE
38. BACK PRESSURE RELIEF
39. COMPOUND PRESSURE GAGE
40. LOW BOOST SHUT-DOWN
41. RESERVOIR SELECTOR VALVE
42. RESERVOIR DRAIN VALVE

JS-12-0-008A

Figure 12-8A. Hydraulic System Schematic Typical Hydraulic Ground Power Unit

SECTION 12.5

SERVICING THE BATTERIES

Contents

Page
GENERAL 12-27
CHARGING THE BATTERIES (CONSTANT POTENTIAL METHOD) 12-27
STATE OF CHARGE 12-27
POWER CONNECTIONS FOR BATTERY CHARGING 12-27
SERVICING THE BATTERY FOR INSTALLATION 12-27
BATTERY INSPECTION 12-27
CHARGING THE BATTERIES (CONSTANT-CURRENT METHOD) 12-28
REPLENISHING THE BATTERY ELECTROLYTE 12-29
CHECK ELECTROLYTE LEVEL AND INTERCHANGE BATTERIES EVERY 50 HOURS 12-29
STORING THE BATTERIES 12-29
SERVICING BATTERY VENT AND DRAIN TUBES 12-29
FIGURES
12-9 Battery Servicing 12-30A

12.5.1 GENERAL. Two nickel-cadmium batteries of 36 ampere-hour capacity are used in the JetStar. The active material of the positive plates is nickel-oxide, and that of the negative plates is cadmium-oxide. The electrolyte, a solution of potassium hydroxide in distilled water, does not enter into any chemical reaction with the plates; therefore, no significant change in the specific gravity occurs during charging or discharging. At approximately 70 percent charge, the water begins to break down and the battery gasses slightly up to 100 percent charge. On overcharge, gassing becomes progressively more rapid with the positive plates giving up hydrogen and the negative plates giving up oxygen. In flight, these gases are carried off by the venting system; however, when charging the batteries on the ground, some hydrogen may collect in the top of the battery case. For this reason, it is important that the battery be kept clean and the terminals tight so that no hot spots can develop which would ignite the hydrogen. As a safeguard against injury to servicing personnel, the battery case and cover are designed to contain any hydrogen explosion that might occur.

12.5.2 CHARGING THE BATTERIES (CONSTANT POTENTIAL METHOD). The batteries are normally charged by the constant-potential method, using the airplane generating system or a power cart, when connected as shown in figure 24-1. This is the only method that can be used with the battery installed in the airplane. The proper voltage level is 28 volts when charging through the airplane system. When using external power, the voltage may be adjusted to 28.5 volts; however, be careful not to exceed this value when making the adjustment.

CAUTION

Charging at a voltage higher than 28.5 may damage the battery by overcharging. Self-destruction, also known as runaway or vicious cycling, can occur if overcharging is excessive. This condition results because overcharging generates heat; this reduces the internal resistance of the cell and allows more current to flow, generating still more heat. The charging process absorbs heat up to approximately 70 percent of charge. The heat generated beyond that point is not significant until the overcharge region is reached. Proper battery maintenance and observance of the battery starting duty cycle will prevent self-destruction or a runaway condition. Charging a battery in ambient temperatures over

160°F or with the battery case temperature over 130°F may also damage the battery.

Note

Nickel-cadmium batteries will accept very high charging currents and will therefore charge very rapidly. Power supplies with capacities up to 1,500 amperes may be used without damaging the cells.

12.5.3 STATE OF CHARGE. It is difficult to determine the exact amount of energy stored in the battery. An approximate amount, which is adequate for normal use, can be determined by observing the charging current. Upon application of the charging voltage, a high-inrush current may be noted; the exact amplitude of this current is unimportant, but the length of time required for the current to return to the trickle charge rate is an indication of the state of charge. A fully charged battery will accept a high-inrush current and will return to trickle current in 1 to 2 minutes. A partially discharged battery will accept a high-inrush current, but will not return to the trickle charge rate for approximately 20 minutes. The trickle charge is defined as a value between 1 to 5 amperes. When this value is obtained, the battery is as fully charged as possible for a 28-volt charging voltage, and the battery is charged to approximately 95 percent of its rated capacity. To obtain a charge to 100 percent capacity, a charging voltage of 28.5 volts is required. These values are approximate, as they depend somewhat upon temperature conditions.

12.5.4 POWER CONNECTIONS FOR BATTERY CHARGING. Power for charging the batteries is applied through the SERVICE RECEPTACLE in the compartment for connecting external electrical power (see figure 24-1). This requires an AN2551 DC ground power plug. To charge the batteries, turn the battery switch to "ON" and apply external power.

12.5.5 SERVICING THE BATTERY FOR INSTALLATION. The battery may be received from the manufacturer in a partially discharged condition with no visible liquid in the cells. This is a normal condition; however, before installing, subject the battery to a constant current charge and inspect the liquid level. If equipment for constant current charging is unavailable, install the battery in the airplane, charge at constant voltage, and inspect the liquid level. Charge the battery by the constant current method as outlined in paragraph 12.5.7 as soon as possible.

12.5.6 BATTERY INSPECTION (100 hour). Remove, inspect, and service the batteries at the 100-hour inspection period or after 200 starts, whichever occurs first, or when a symptom of lost capacity develops as indicated in trouble shooting the battery as follows:

Note

Note the location of the batteries before removal, and interchange positions upon reinstallation.

a. Inspect for cleanliness. Remove battery from the airplane. Disassemble and remove the cell interconnecting straps, noting the order of parts removal for reinstallation. Remove the cells from battery case, and flush the cells and the case with tap water to remove dirt, dust, foreign matter, or white powdery deposits. After cleaning, drain the case and allow it to dry. Drying may be accelerated by using compressed air. Clean the interconnecting straps with water and soft bristle brush. Place the cells in the case, and reinstall the interconnecting straps. Torque all screws and double D washers to 50 inch-pounds.

CAUTION

Do not attempt to clean the battery with solvents, acid or any chemical solution. Do not use wire brushes.

b. Inspect for loose connection at quick-disconnect. Check inside and outside for burns or evidence of overheat. Tighten the screws firmly.

c. Inspect the case for distortion. Remove and replace, if required.

d. Charge per paragraph 12.5.7.

12.5.7 CHARGING THE BATTERIES (CONSTANT CURRENT METHOD). Charge the battery as follows:

a. Completely discharge the battery. This is accomplished by discharging the battery into a load drawing approximately 17 amperes for two hours. Then, each cell within the battery must be discharged. When the battery appears to be discharged and current is no longer flowing, before disconnecting the load, apply shorting jumpers across the terminals of each cell. Remove the load, and allow the battery to discharge in this manner for 12 hours. Remove the shorting straps, and check the battery terminal voltage. If the voltage is less than 2.4, the battery is ready for constant-current charging.

Note

Constant-current charging permits an accurate determination of the amount of energy restored to the battery, and it conditions and balances the cells to eliminate faded capacity. To restore a completely discharged battery to 100 percent capacity, between 140 and 160 percent of the rated capacity must be returned to the battery. If the battery is not completely discharged, it is not easy to determine the proportionate amount of returned energy. It is recommended that in all cases of fading or loss of capacity, the battery be completely discharged before constant-current charging.

b. Any type of battery charger that will deliver 8 amperes of charging current for 7 hours may be used for constant-current charging. If a suitable charger is not available, a variable-voltage source capable of the 8-ampere delivery may be used. Near the end of the charge it may be necessary to raise the voltage as high as 35 volts. This type of equipment must be monitored and the voltage adjusted manually to maintain a constant current rate. During the last two hours of the charge cycle, the cell voltages must be checked to determine if all cells are rising evenly. Should some cells indicate a voltage lower than the others by .03 volts or more, leave the battery on charge until these cells come up to 1.60 volts. During the greater portion of charge, the cell voltages will remain between 1.4 and 1.45 volts. When the battery has been charged for approximately 70% of the time necessary to fully charge it, the voltage will sharply rise to 1.60 volts or more. Should any cell fail to rise to or above 1.60 volts, it should be removed and replaced with a good cell. After replacing a cell, the battery should be completely discharged and recharged again to ensure that all cells are balanced. At the end of the charge, check the liquid level and add or remove liquid, if necessary, as instructed in paragraph 12.5.8.

12.5.8 REPLENISHING THE BATTERY ELECTROLYTE. Electrolyte consumption of the nickel-cadmium battery is very low as compared to that of a lead-acid battery. The visible level depends upon the battery state of charge. A battery will appear to have a higher-than-normal electrolyte level immediately after it is charged.

CAUTION

Contamination of the electrolyte can become a problem if lead-acid batteries and nickel-cadmium batteries are stored in the same area or are serviced with the same tools.

a. Remove the battery covers and the battery filler caps.

CAUTION

Over exposure to the atmosphere will contaminate the electrolyte with CO_2. Keep the cell caps in place when possible.

b. After charging the battery and allowing it to sit from 1 to 3 hours, check the electrolyte level. The proper level is just above the top of the barrier (1/16 to 1/8 inch). The level can be correctly determined by the use of a glass tube which is open on both ends. Insert the tube into the cell until it rests on the barrier above the plates. Then close the top of the tube and withdraw the tube from the battery cell. If there is 1/8 inch or less liquid retained in the lower end of the tube, the liquid level is properly adjusted. If no liquid is withdrawn, add distilled or demineralized water until the level is 1/16 to 1/8 inch above the barrier.

c. After electrolyte level check and necessary level adjustments have been made, check the specific gravity of the electrolyte of each cell. If the specific gravity of each cell is between 1.24 and 1.30 at room temperature proceed to paragraph (d). If the specific gravity of the electrolyte of any cell is below 1.24 the battery should be cycled once, then the specific gravity checked again. If the specific gravity of any cell is still below 1.24 replace cell with a fully charged cell.

d. Replace the filler caps and tighten securely.

e. Clean the sump jar and pad, and saturate the pad with a 5-percent boric acid solution.

f. Replace the battery covers.

12.5.9 CHECK ELECTROLYTE LEVEL AND INTERCHANGE BATTERIES EVERY 50 HOURS.

a. Ensure that each battery is fully charged by using the constant-potential method of charging prescribed in paragraph 12.5.2.

b. Remove the batteries from the aircraft, noting the position from which each battery is removed.

c. Check and replenish the electrolyte level as prescribed in paragraph 12.5.8, steps (a), (b), (d), (e), and (f).

d. Check fit of battery connector and receptacle per paragraph 24.0.5.3.

e. Interchange and reinstall the batteries, placing the battery removed from No. 1 position in No. 2 position, and the battery removed from the No. 2 position in No. 1 position.

12.5.10 STORING THE BATTERIES. Nickel-cadmium batteries may be stored for long periods of time in any state of charge. Retention of the charge depends largely upon the ambient temperature of the storage area. A fully charged battery stored at 80°F will retain approximately 60 percent of its rated capacity over a period of one year. A battery stored in ambient temperature of 0°F will retain approximately 90 percent of its rated capacity for a period of one year.

Note

The active materials of the plates are insoluble in the electrolyte when the battery is stored, on open circuit. Before placing a battery in storage, it should be cleaned. Charge a battery that has been in storage, by the constant-current method prior to installing it in the airplane.

12.5.11 SERVICING BATTERY VENT AND DRAIN TUBES. Check battery vent and drain tubes to make certain they are clear of any obstructions. The tubes may be checked for obstructions by blowing through the tubes.

WARNING

Do NOT put mouth in direct contact with vent or drain tube. Use a clean short section of tubing as a mouthpiece that fits snugly inside the vent or drain tube. To safeguard against the possibility of entry of electrolyte into mouthpiece, keep mouthpiece elevated at all times during clearing procedure. Do NOT inhale while mouthpiece is in the mouth. Failure to observe this warning may result in severe mouth and throat burns.

When clearing a battery vent or drain tube of obstructions, disconnect tube at both ends. To catch any accumulation of electrolyte, place lower end of tube in a jar or suitable container. Do not permit electrolyte to drip on airplane structure because it may cause corrosion to develop at that point. After the cleaning procedure has been accomplished, reconnect tubes to vent connections. Make certain tube connections are tight and leakproof.

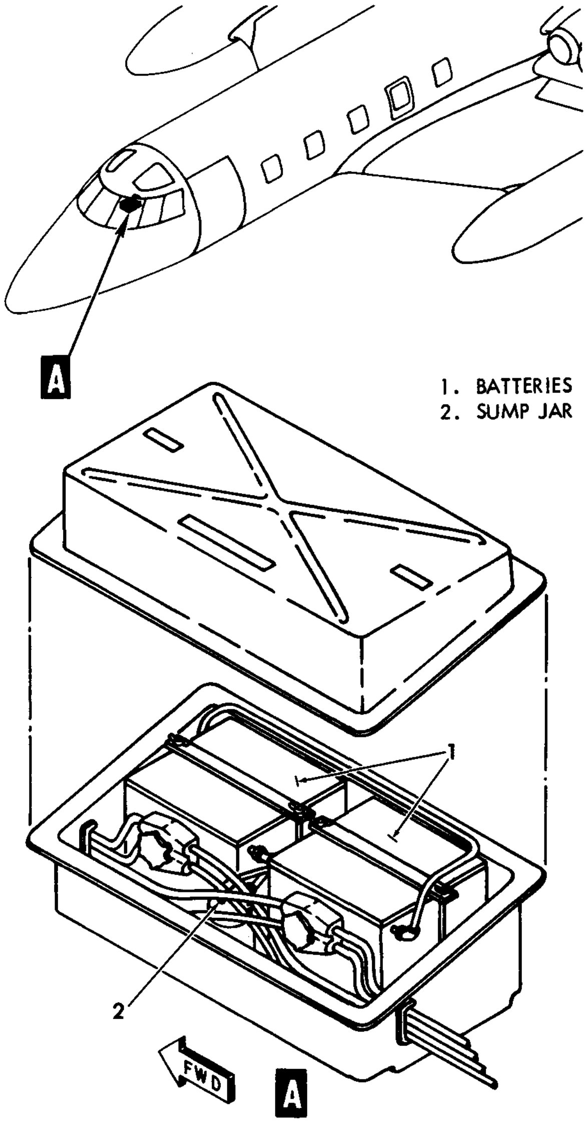


Figure 12-9. Battery Servicing

SECTION 12.6

SERVICING THE LANDING GEAR

Contents Page

LANDING GEAR TIRES 12-31
LANDING GEAR STRUTS 12-32
OIL SERVICING 12-32
ALTERNATE METHOD OF MAIN LANDING GEAR SHOCK STRUT SERVICING 12-32
STRUT INFLATING 12-32
LANDING GEAR EMERGENCY EXTENSION BOTTLES 12-32A
FIGURES
12-10. Landing Gear Servicing 12-33

WARNING

Any equipment used for pressurizing tires should be provided with pressure regulating and relief valves to prevent the applied tire pressure from ever exceeding 300 psi.

12.6.1 LANDING GEAR TIRES. All wheels are provided with conventional valve stems.

12.6.1.1 Dash 8 JetStars 5093, 5097 and up, and all aircraft with S.B. 329-230 or S.B. 329-251 incorporated should have both NLG and all MLG tires serviced to 220 (±5) psi.

12.6.1.2 All other aircraft covered by this manual may use either the specified tire or the Dash 8 JetStar tire with the following restrictions and conditions. Both NLG tires shall be the same part number. The NLG tire pressure shall be either 180 (±5) psi or 220 (±5) psi at the operator's option, but both tires should be at the same pressure. MLG tires may be interchanged at any of the four positions. The MLG tire pressure shall be either 205 (±5) or 220 (±5) psi at the operator's option, but all four tires should be at the same pressure.

WARNING

Improper tire inflation may be hazardous to personnel and property. Proper safety precautions must be adhered to by all personnel involved in servicing tires.

Note

To improve tire life, use of higher tire pressure is recommended.

12.6.1.3 Observe the following safety precautions when servicing tires:

a. Use only dry air or nitrogen.

Note

Uniform use of nitrogen for inflating and servicing tires will reduce the possibility of explosion of an overheated tire.

b. Use a low pressure source with a maximum pressure capability of 300 psi if available.

c. If a low pressure source is not available, use high pressure servicing equipment, protected by a pressure regulating valve and a relief valve, capable of controlling the supply air pressure to 300 psi maximum.

d. Use a tire inflator kit to control and monitor the inflator pressure. Milton Industries manufactures a kit, Part Number 1065.

Milton Industries
4500 West Cortland Street
Chicago, IL 60639
Phone: 312-235-9400

e. Never permit personnel to locate themselves facing a wheel when inflating or deflating a tire. Stand either forward or aft of the landing gear assembly.

f. Never inflate above the maximum specified tire inflation pressure. Excess pressure can cause a tire or wheel to fail with explosive force.

12.6.1.3 (Cont).

CAUTION

Do not deflate tire by removing the valve core. With high air pressure in the tire, the valve core will be expelled with explosive force.

g. When a tire or wheel on an aircraft is defective, remove the pressure from the tire before attempting any removals or repairs.

h. Do not tow or taxi an airplane with underinflated tires.

i. When servicing tires, always ensure that tires are sufficiently inflated in accordance with the requirements in this chapter of the manual.

12.6.2 LANDING GEAR STRUTS. The landing gear struts are of the familiar oil-air type, and must be serviced with both air and oil (see figure 12-10).

12.6.3 OIL SERVICING. Excessive oil leakage on the polished surface of the piston, or spongy action of the strut, even though the air inflation is correct, indicates a possible low oil condition. Check and replenish the oil by the following procedure:

a. Remove the valve cap.

b. Rotate the hexagonal swivel nut slowly counterclockwise until air begins to bleed from the valve, and allow pressure to bleed slowly from the unit.

WARNING

There are two hexagonal nuts on the valve assembly. Do not turn the nut that has the safety wire attached. Turning this nut will loosen the entire valve assembly, which may blow out explosively if loosened while pressure is in the unit. Also do not expose any part of your body to the stream of bleed air. This air is expelled with considerable force and can easily cause bodily injury.

c. Allow the air to bleed completely from the strut, then cut the safety wire on the other hexagonal nut and remove the entire valve assembly from the strut.

WARNING

Be very sure the air pressure in the strut is completely depleted before removing the valve assembly. If the valve is removed with pressure in the strut, the valve may be explosively expelled.

d. With the strut fully compressed, fill it to overflowing with hydraulic fluid (red), Specification MIL-H-5606, through the valve opening.

WARNING

Hydraulic fluid may contain Tricresyl Phosphate (TCP) which can be absorbed through the skin and which can produce paralysis if taken internally. Appropriate protective measures should be taken to avoid such exposures.

Note

When the main landing gear struts are fully compressed, before servicing with oil, the piston assembly may partially block the refill port making it difficult to add oil in the required quantity. This condition can be eliminated by jacking the airplane and allowing the struts to extend approximately 1/2 inch; fill with oil in the normal manner, and lower the airplane. Excess oil will then drain out. This method may also be used to service the nose landing gear strut.

e. Replace the valve assembly and safety wire, and inflate the strut in accordance with paragraph 12.6.5.

12.6.4 ALTERNATE METHOD OF MAIN LANDING GEAR SHOCK STRUT SERVICING. This method does not require jacking the airplane. Instead, a hydraulic hand cart is used. The cart should be serviced with MIL-H-5606 hydraulic fluid. A special union is required to connect the hydraulic hand cart to the strut. The Lockheed part number for the union is 203C004-001C. This method may also be used on the nose landing gear strut.

a. After the strut has been completely deflated (see paragraph 12.6.3), leave the MS28889-1/-2 valve in the open position.

b. Install the special union, and connect the hand cart dispensing hose. Pump until a slight extension of the gear is noticed; then, open the valve on the cart and allow excess oil to drain back to the cart reservoir. Repeat this procedure several times until no air is evident in the oil returning to the cart.

c. Disconnect the cart, remove the union from the strut and inflate the strut.

12.6.5 STRUT INFLATING. Due to a wide variation in airplane weights, an accurate check of strut inflation cannot be made by simply measuring the strut

12.6.5 (Cont).

extension. Use the following procedure to check and inflate the struts.

a. Remove the valve cap, and attach the pressure source.

b. Loosen the hexagonal nut about a quarter of a turn.

WARNING

There are two hexagonal nuts on the valve assembly. Do not turn the nut that has the safety wire attached. Turning this nut will loosen the entire valve assembly, which may blow out explosively if loosened while pressure is in the unit.

c. Apply pressure to the strut until the pressure reading and torque arm center distance agrees with one pair of figures on the inflation chart (see figure 12-10).

d. When the desired inflation is obtained, retighten the hexagonal nut, reduce the pressure source to zero, remove the pressure source, and replace the valve cap.

12.6.6 LANDING GEAR EMERGENCY EXTENSION BOTTLES.

Note

Airplanes 5001 thru 5040 have one main (L.H. side) and one nose landing gear emergency extension bottle. Airplanes 5001 thru 5085 have the nose landing gear emergency extension bottle located above the nose gear strut (F.S. 266 approx.). Airplanes 5086 and up have the nose landing gear emergency extension bottle located in the nose wheel well at F.S. 228 approx. Airplanes 5041 and up, and those incorporating Service Bulletin 329-25 have two main bottles (one L.H. side and one R.H. side) and one nose landing gear emergency extension bottle. Both the main and nose landing gear emergency extension bottles should be serviced when the pressure gages on the bottles indicate 1,800 PSI or less. The normal pressure for these bottles is 2,000 PSI of either nitrogen or dry air. Service these bottles as follows (see figure 12-10):

a. Remove the valve cap and attach the pressure source.

b. Loosen the hexagonal swivel nut about a quarter of a turn.

WARNING

There are two hexagonal nuts on the valve. Do not loosen the nut with the safety wire attached. To loosen this nut will loosen the valve assembly, which may blow out explosively if loosened while pressure is in the bottle.

c. Apply pressure to the bottle until the correct pressure is obtained.

d. Retighten the hexagonal nut, and reduce the pressure source to zero.

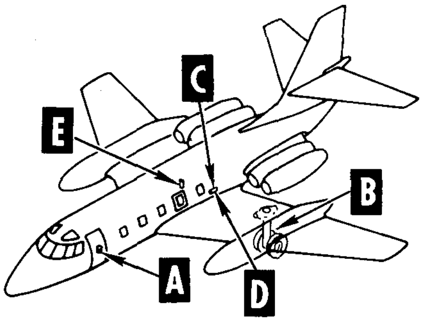
e. Remove the pressure source, and replace the valve cap.

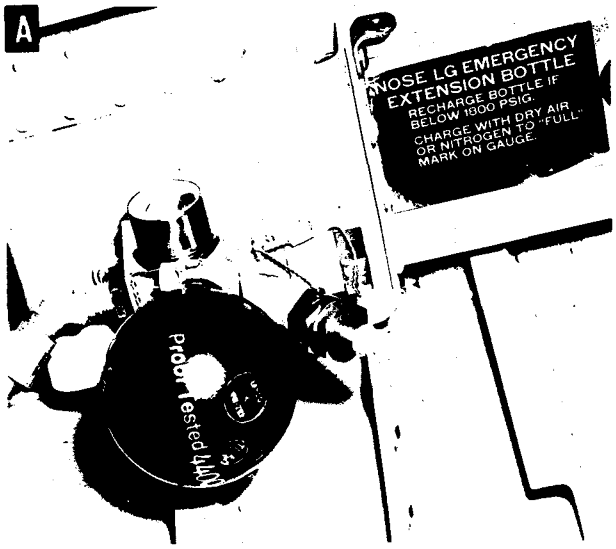
WARNING

Do not remove the pressure source until the hexagonal nut is retightened and the pressure source is cut off.

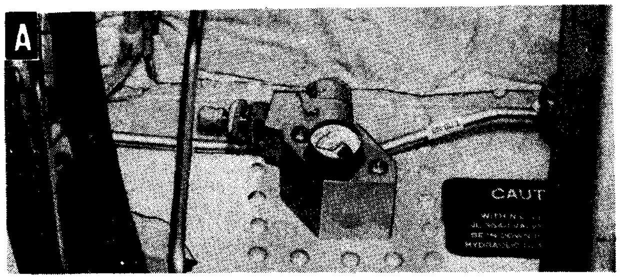
Note

Airplanes 5076 and up are equipped with air bottles which have a non-reseating (rupture disc) type of relief valve. These relief valves are set to relieve at 4,000 ± 250 PSI. If the relief valve operates, replace the valve assembly. Consult the Illustrated Parts Breakdown (IPB) for replacement information. Airplanes 5075 and under may also have obtained air bottles with non-reseating relief valves thru preferred spares and/or rework. The same information is applicable to these bottles.

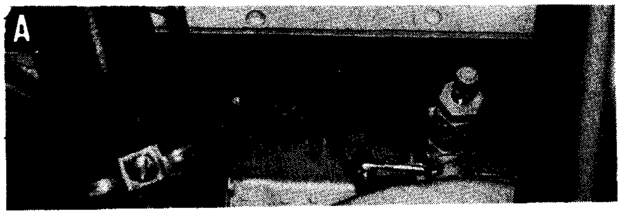



A
NOSE LG EMERGENCY EXTENSION BOTTLE
RECHARGE BOTTLE IF BELOW 1800 PSIG.
CHARGE WITH DRY AIR OR NITROGEN TO "FULL" MARK ON GAUGE.
Proof Tested 4400

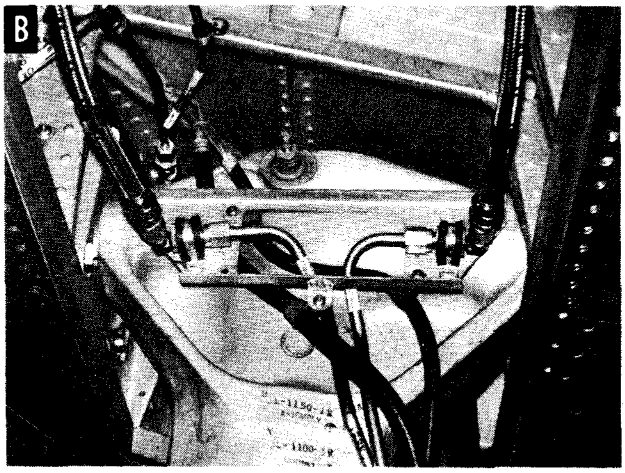
EMERGENCY EXTENSION BOTTLE
AIRPLANES 5086 AND UP



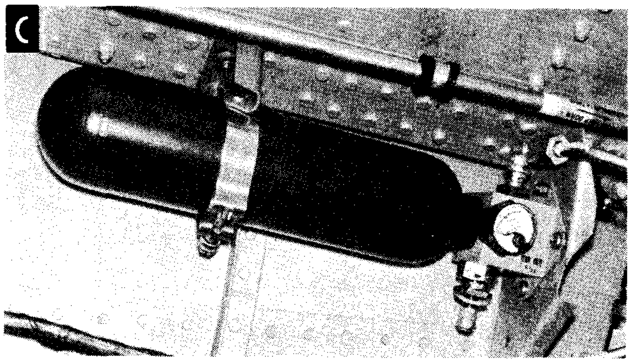
EMERGENCY EXTENSION BOTTLE
AIRPLANES 5001 THRU 5085



NOSE GEAR STRUT



MAIN LANDING GEAR STRUT

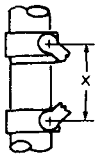


MAIN LANDING GEAR EMERGENCY EXTENSION BOTTLE
AIRPLANES 5001 THROUGH 5043 NOT HAVING
SERVICE BULLETIN 329–96 INCORPORATED

Figure 12-10. Landing Gear Servicing (Sheet 1 of 3)

STRUT SERVICING INSTRUCTIONS (MAIN LANDING GEAR)

WITH AIR PRESSURE RELEASED & STRUT FULLY COMPRESSED, REMOVE AIR VALVE & FILL WITH OIL SPEC. MIL-H-5606 TO LEVEL OF VALVE SEAT. REGARDLESS OF AIRCRAFT WEIGHT AND STRUT FRICTION, INFLATE STRUT UNTIL PRESSURE READING & TORQUE ARM CENTER DISTANCE "X" AGREES WITH ONE PAIR OF FIGURES ON CHART BELOW.



INFLATION CHART

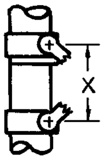
T/A CENTER DISTANCE	5.00	5.50	6.50	7.00	7.38	7.50	7.62	7.75	7.88	8.00	8.12
GAUGE PRESSURE P.S.I.	3019	2368	1613	1383	1247	1208	1170	1135	1102	1070	1040
T/A CENTER DISTANCE	8.25	8.38	8.50	8.62	8.75	8.88	9.00	9.25	9.50	9.75	10.00
GAUGE PRESSURE P.S.I.	1012	985	960	936	913	891	870	830	794	761	731
T/A CENTER DISTANCE	10.50	11.50	12.50	14.00	15.50	16.50	17.50	18.50	19.50	20.00	20.50
GAUGE PRESSURE P.S.I.	676	588	519	441	383	352	325	302	282	273	264

WARNING

RELEASE AIR IN STRUT BEFORE DISASSEMBLING

STRUT SERVICING INSTRUCTIONS (NOSE LANDING GEAR)

WITH AIR PRESSURE RELEASED AND STRUT FULLY COMPRESSED, REMOVE AIR VALVE AND FILL WITH OIL SPEC. MIL-H-5606 TO LEVEL OF VALVE SEAT. REGARDLESS OF AIRCRAFT WEIGHT AND STRUT FRICTION, INFLATE STRUT UNTIL PRESSURE READING AND TORQUE ARM CENTER DISTANCE "X" AGREES WITH ONE PAIR OF FIGURES ON CHART BELOW.



INFLATION CHART

T/A CENTER DISTANCE	7.94	8.06	8.18	8.31	8.44	8.56	8.68	8.81	8.94	9.06	9.18
GAUGE PRESSURE P.S.I.	894	863	833	806	780	756	733	712	692	673	654
T/A CENTER DISTANCE	9.31	9.44	9.56	9.68	9.81	9.94	10.06	10.18	10.31	10.44	10.56
GAUGE PRESSURE P.S.I.	637	620	605	590	576	562	549	537	525	514	503
T/A CENTER DISTANCE	10.68	10.81	10.94	11.06	11.18	11.31	11.44	11.56	11.81	12.06	12.31
GAUGE PRESSURE P.S.I.	492	482	473	463	454	446	438	429	414	400	387

WARNING

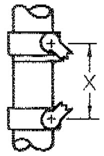
RELEASE AIR IN STRUT BEFORE DISASSEMBLING

Airplanes 5001 thru 5055 and 5059 except those Incorporating Service Bulletin 329-157

Figure 12-10. Landing Gear Servicing (Sheet 2 of 3)

STRUT SERVICING INSTRUCTIONS (NOSE LANDING GEAR)

WITH AIR PRESSURE RELAESED AND STRUT FULLY COMPRESSED, REMOVE AIR VALVE AND FILL WITH OIL SPEC. MIL-H-5606 TO LEVEL OF VALVE SEAT. REGARDLESS OF AIRCRAFT WEIGHT AND STRUT FRICTION. CAUTION - CHOCK MAIN LANDING GEAR. JACK AIRCRAFT NOSE SHOCK STRUT. INFLATE WITH AIR PRESSURE TO 300 ± 25 PSI.



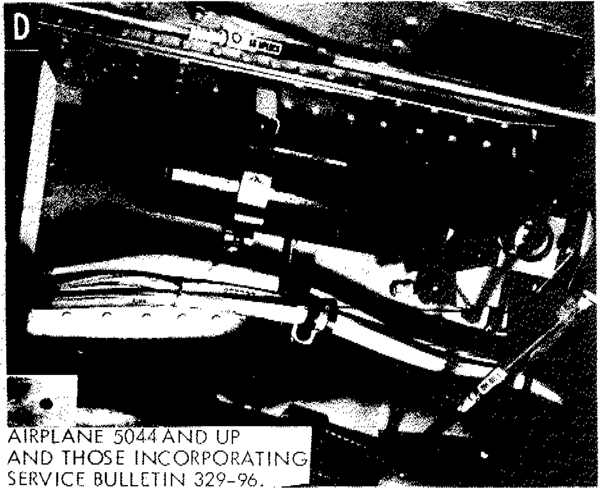
FOR REFERENCE ONLY. ALL PRESSURES WITHIN 100 PSI.

GAUGE PRESSURE PSI	800	900	1000
"X" DISTANCE	9-3/4	9-1/4	8-3/4

WARNING

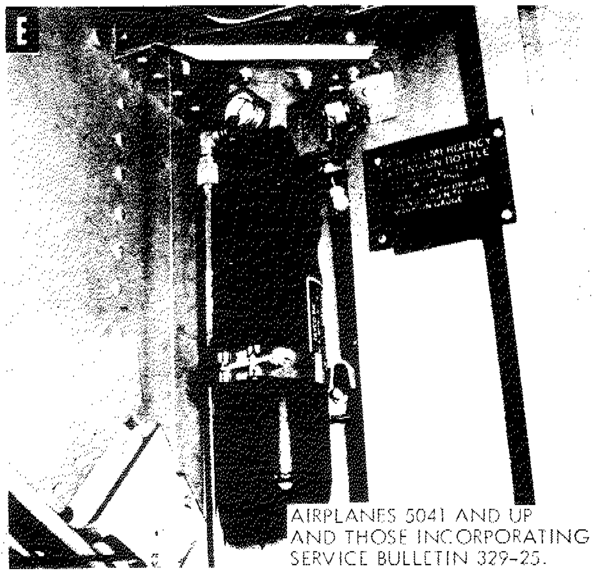
RELEASE AIR IN STRUT BEFORE DISASSEMBLING

(Airplanes 5056, 5057, 5058, 5060 and up) and those Incorporating Service Bulletin 329-157



AIRPLANE 5044 AND UP AND THOSE INCORPORATING SERVICE BULLETIN 329-96.

LEFT MAIN LG EMERGENCY EXTENSION BOTTLE



AIRPLANES 5041 AND UP AND THOSE INCORPORATING SERVICE BULLETIN 329-25.

RIGHT MAIN LG EMERGENCY EXTENSION BOTTLE

JS-1-2-X0/2-1210

Figure 12-10. Landing Gear Servicing (Sheet 3 of 3)

SECTION 12.7

SERVICING THE PITOT STATIC SYSTEM

Contents Page

DRAINING 12-35

FIGURES

12-11. Pitot Static Drain Provisions 12-36

12.7.1 DRAINING. Provisions are included for draining moisture accumulation from the pitot and static tubing (see figure 12-11). Remove the plugs from the tube ends, allow all moisture to drain and replace the plugs. Check out the system.

Note

See Section 21.1 for water drainage instructions relating to the pressurization safety valve static line.

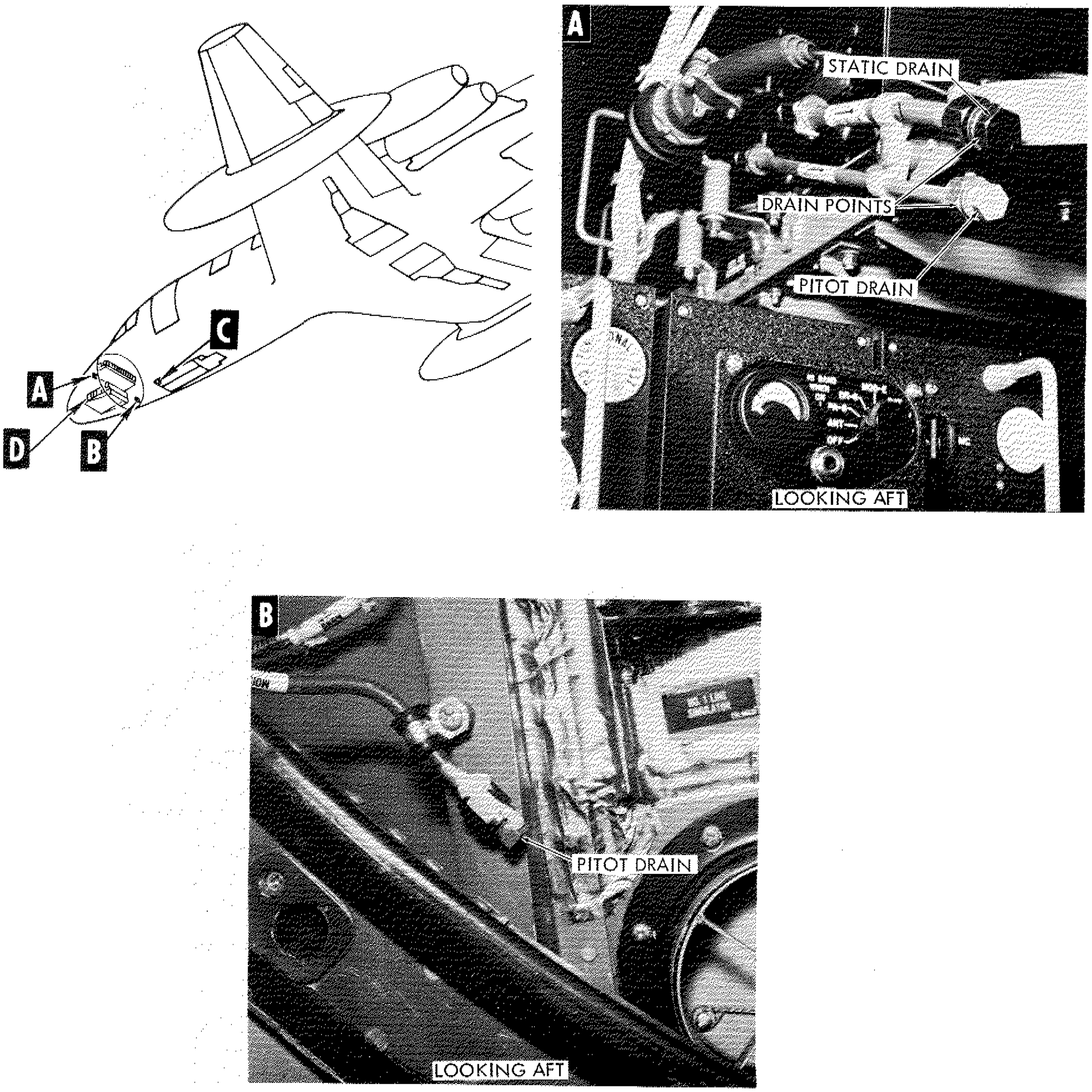


Figure 12-11. Pitot Static Drain Provisions (Sheet 1 of 2)

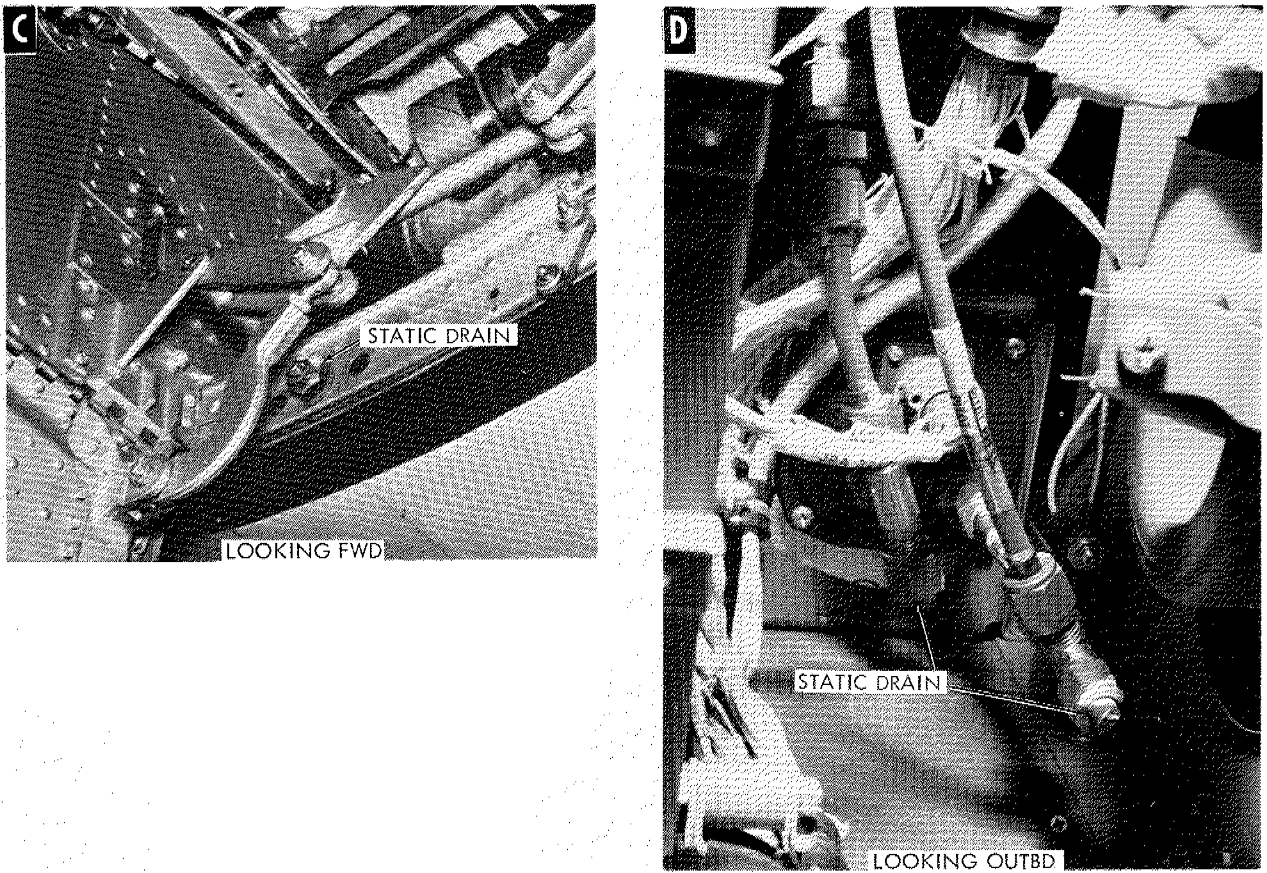


Figure 12-11. Pitot Static Drain Provisions (Sheet 2 of 2)

SECTION 12.8

SERVICING THE ENVIRONMENTAL SYSTEMS

Contents Page

SERVICING THE AIR CONDITIONING SYSTEM 12-39

Oil Servicing - Air Conditioning Package Rotative Assembly 12-39
Installation of Air Conditioning Package Rotative Assembly 12-39
Air Conditioning System 12-39

SERVICING THE PRESSURIZATION SYSTEM 12-39

SERVICING THE ELECTRONIC EQUIPMENT COOLING SYSTEM 12-39

SERVICING THE TOILET VENTILATION SYSTEM 12-39

FIGURES

12-12 Cooling Turbine Oil Reservoirs 12-40
12-12A Sources For MIL-L-6085 Oil 12-40A

12.8.1 SERVICING THE AIR CONDITIONING SYSTEM.

12.8.1.1 OIL SERVICING - AIR CONDITIONING PACKAGE ROTATIVE ASSEMBLY. Two identical cooling turbines, on opposite sides of the aft compartment, each have a small reservoir of translucent plastic (see figure 12-12). Drain and fill reservoirs every 500 hours, using oil conforming to MIL-L-6085 (see figure 12-12A). Each time oil is added, the filler plug preformed packing, part number AN6290-2, should be replaced.

CAUTION

Remove oil with suitable syringe. Do not attempt to remove oil by disassembling the sump from the unit and pouring the oil from the sump. When the sump is reassembled to the unit, it is possible the lubrication wicks may be cut, causing subsequent oil starvation of the bearings.

Note

Moisture is indicated if a white cloudiness appears in the bottom of the reservoir. This moisture may be removed, using a hypodermic syringe.

Note

Auxiliary power unit (APU) operating hours should be considered in addition to engine operating hours when determining the refill schedule.

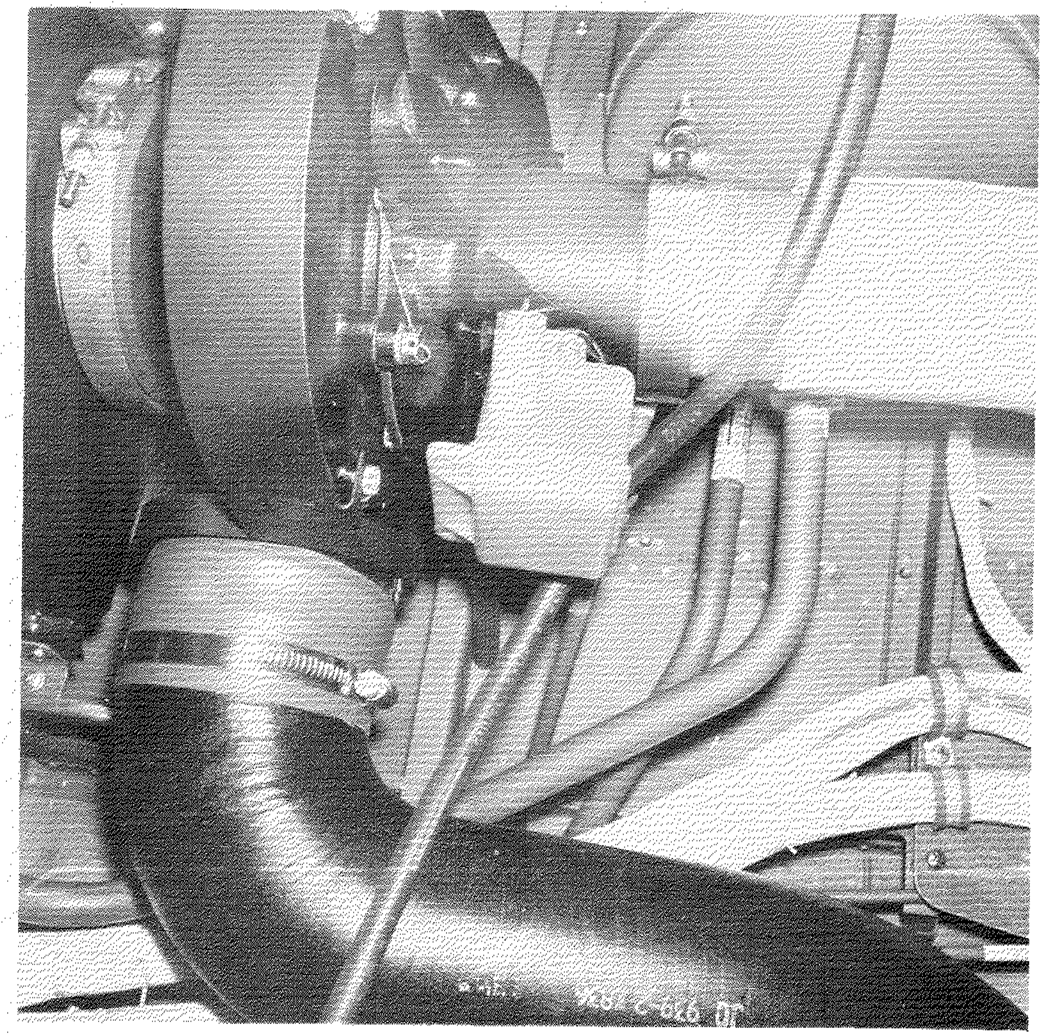
12.8.1.2 INSTALLATION OF AIR CONDITIONING PACKAGE ROTATIVE ASSEMBLY. When installing a turbine fan assembly or refrigeration package, insure that the oil sump is filled in accordance with paragraph 12.8.1.1 at least three hours prior to running the package. The lubrication wicks must be saturated with oil for this minimum time period before they will provide adequate lubrication to the bearing. Install unit in accordance with instructions in Section 21.0. The sump should be filled only after the unit is installed in the aircraft.

12.8.1.3 AIR CONDITIONING SYSTEM. Additional servicing instructions for the air conditioning system are given in Section 21.0.

12.8.2 SERVICING THE PRESSURIZATION SYSTEM. Servicing instructions for the pressurization system are given in Section 21.1.

12.8.3 SERVICING THE ELECTRONIC EQUIPMENT COOLING SYSTEM. Servicing instructions for the electronic equipment cooling system are given in Section 21.2.

12.8.4 SERVICING THE TOILET VENTILATION SYSTEM. Servicing instructions for the toilet ventilation system are given in Section 21.3.



JS-2-2-1212

Figure 12-12. Cooling Turbine Oil Reservoirs

Source	Product Designation
American Oil & Supply Co. 238 Wilson Avenue Newark, NJ 07105	"PQ" Rust Preventive No. 160
Anderson Oil & Chemical Co. Portland, CT 06480	L-245X or Winsor Lube L-245X
Bray Products Div 16815 Von Karman Ave Suite 202 Irvine, CA 92714	Brayco 885
Exxon Company, USA P.O. Box 2180 Houston, TX 77252	Univis P-12
Hatco Chemical Corporation King George Post Road Fords, NJ 08863	Hatcol -3885
E.F. Houghton & Company Madison & VanBuren Avenues Valley Forge, PA 19482	Cosmolubric No. 270-A
Octagon Process, Inc. 596 River Road Edgewater, NJ 07020	Product 80
Royal Lubricants Company, Inc. P O Box 518 1 Merry Lane East Hanover, NJ 07936	Aeroshell Fluid 12, Royco 885 or Royco 885A
Tenneco Chemicals Turner Place, P.O. Box 365 Piscataway, NJ 08854	Anderol L-401D
Viscosity Oil Company 3200 S. Western Avenue Chicago, IL 60609	Visconorust 1615

Figure 12-12A. Sources For MIL-L-6085 Oil

SECTION 12.9

SERVICING THE PITCH TRIM ACTUATOR

Contents .... Page

GENERAL 12-1

PITCH TRIM ACTUATOR INSPECTION AND SERVICING 12-1

FIGURE

12-13 Pitch Trim Actuator Servicing Provisions 12-2

12.9.1 GENERAL. The pitch trim actuator is filled with lubricating oil. Varying flight conditions, such as changes in altitude and temperature, can cause condensation inside the actuator assembly, contaminating the oil. This water could freeze and impair the operation of the actuator. To guard against this condition, inspect the actuator: (1) at intervals prescribed in Chapter 5, (2) when the oil level drops below a level visible in the upper sight glass at the 4.50-degree nose up position, or (3) whenever oil leaks from the actuator are detected. Use MIL-L-7808 oil for servicing the pitch trim actuator.

12.9.2 PITCH TRIM ACTUATOR INSPECTION AND SERVICING.

a. Remove sump drain plug and drain oil until all evidence of suspended water disappears. Actuators not provided with a sump (Part No. JC1002-8, prior to incorporation of Sargent Industries Service Bulletin 1506A1073) are drained at four places (see figure 12-13). Drain the two upper positions until all evidence of water disappears and then drain the two lower positions. Should a large amount of suspended water be found, retract the actuator to full nose up trim and drain all oil.

CAUTION

Catch this lubricating oil in a container. This oil is harmful to painted surfaces.

b. After draining all water, position the actuator at six degrees nose up, as shown on the horizontal stabilizer indicator, and check to see if oil is visible through sight glass at the upper end of the housing on each outboard side. If oil is not visible at either glass, position the actuator at 4.50 degrees nose up and fill both actuator halves until the oil level reaches the center of both sight glasses. Each actuator half will require about one quart of oil to fill if empty. This also fills the sump.

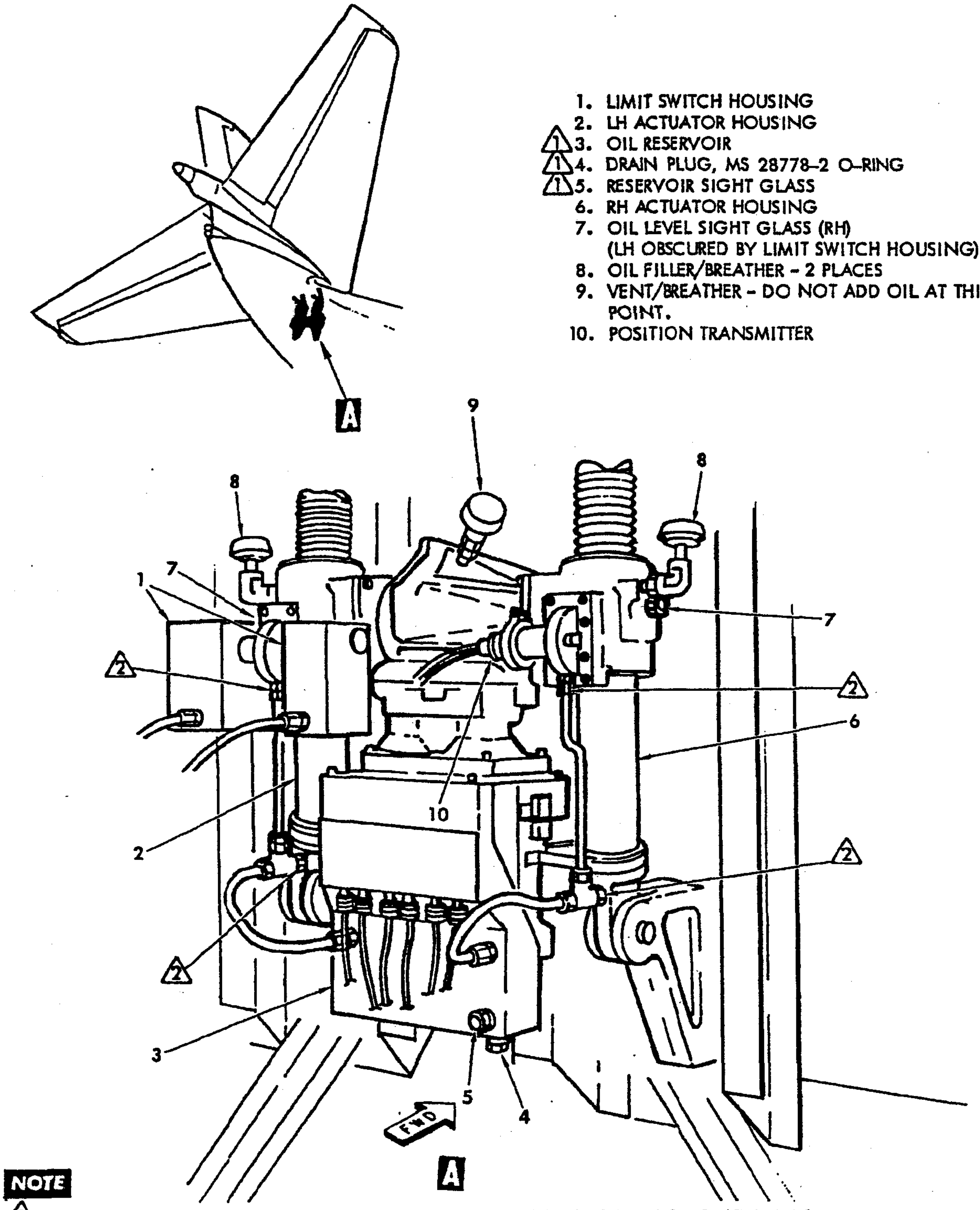
Note

Oil leaks at bolts should be corrected by removal of the bolt and reinstallation with sealant (Permatex Form-A-Gasket No. 2 or equal). Bolts should be replaced with standard torque values (Chapter 51) and resafety-wired. Leaks at gaskets and seals are in most cases not repairable in the field.

CAUTION

The limit switch drive between the two limit switch housings should never be removed in the field because of the critical adjustments involved.

c. The position transmitter drive can be removed if, before removal, notation is made of the position shown in the cockpit. After adjustment, the drive must then be replaced to show the same position without any movement of the actuator. In any such operation, care should be observed to prevent oil spills. It is usually best to drain all oil from the actuator when any removals are made for leak repair. Boot and/or rainshield replacement would not require draining but length and preload adjustments (see figure 27-18) should follow anything that requires disconnecting the actuator rod ends.



1. LIMIT SWITCH HOUSING
2. LH ACTUATOR HOUSING
3. △1 OIL RESERVOIR
4. △1 DRAIN PLUG, MS 28778-2 O-RING
5. △1 RESERVOIR SIGHT GLASS
6. RH ACTUATOR HOUSING
7. OIL LEVEL SIGHT GLASS (RH) (LH OBSCURED BY LIMIT SWITCH HOUSING)
8. OIL FILLER/BREATHER - 2 PLACES
9. VENT/BREATHER - DO NOT ADD OIL AT THIS POINT.
10. POSITION TRANSMITTER

NOTE

△1 THESE PARTS ARE NOT PROVIDED FOR PART NO. JC1002-8 PRIOR TO INCORPORATION OF SARGENT INDUSTRIES SERVICE BULLETIN 1506A1073.

△2 DRAIN PLUG FOR JC1002-8 LOCATED HERE. (AIRPLANES 5001 THRU 5045 PRIOR TO INCORPORATION OF SARGENT INDUSTRIES SERVICE BULLETIN 1506A1073)

JS-1-2-X0-3-1213

Figure 12-13. Pitch Trim Actuator Servicing Provisions

Use and/or disclosure is governed by the statement on the title page of this document.

SECTION 12.10

SERVICING THE OXYGEN SYSTEM

Contents Page

GENERAL 12-43

OXYGEN FIRE POTENTIAL 12-43

SERVICING PRECAUTIONS 12-44
  Service Hose Leakage Test 12-44A

SERVICING PROCEDURES 12-44A
  On-Board Servicing 12-44A
  Servicing by Cylinder Removal 12-44B
    Cylinder Removal 12-44B
    Cylinder Installation 12-44C

INFORMATION NOTES 12-44C

PURGING THE OXYGEN SYSTEM 12-44C

OXYGEN CYLINDERS, PRESSURE-TEMPERATURE 12-44C

SUGGESTED MATERIAL LIST FOR ASSEMBLY OF OXYGEN SERVICE CART 12-44C

FIGURE
  12-13A Recommended Oxygen Service Cart Arrangement 12-44F

12.10.1 GENERAL.

WARNING

The oxygen system should be serviced by qualified personnel only. It is essential that dirt, dust, oil, and grease be prevented from entering the system. The outside surfaces of all components must be kept clean of these contaminants. Tools, hands, and clothing must also be kept clean.

a. The methods most commonly used to service an oxygen system are as follows:

  (1) Replacing a depleted or partially depleted cylinder with a fully charged cylinder. This method requires the availability of a spare cylinder.

  (2) Removing a cylinder from the airplane and hangar for reservicing.

  (3) Servicing a cylinder in the airplane by connecting a service hose to the cylinder from a portable oxygen service cart located just outside the airplane.

  (4) Servicing a cylinder in the airplane by means of connecting a service cart hose to an external service system fitting permanently mounted on the airplane.

b. There is some degree of hazard associated with all of the above methods but the greatest potential for fire occurs during the transfer of high pressure oxygen from one cylinder to another. When this transfer is accomplished at the airplane not only is there the risk of personnel injury, but the airplane may be severely damaged if a fire occurs.

12.10.2. OXYGEN FIRE POTENTIAL.

a. In order for a fire to be started and/or maintained, three ingredients have to be present at the same time. These three ingredients are as follows:

  (1) Any combustible material such as rubber, oils, plastics, metals, etc.

  (2) Source of oxygen to support combustion.

Note

Oxygen does not burn itself but merely supports the combustion, or burning, of other materials.

  (3) A source of ignition.

b. In an oxygen system, oxygen is normally present to support combustion and the plumbing distribution system is made up of some combustible materials, therefore, the only ingredient for a fire that is left to control is the ignition source. Special effort was made to control the ignition source in the JetStar oxygen system by careful selection and examination of materials, routing of lines, and keeping the pressure level low in the distribution lines.

c. Some of the fire potentials that exist when servicing oxygen supply cylinders at the airplane are as follows:

(1) A fire may be created by adiabatic compression (dieseling effect) in the servicing system. Adiabatic compression is created when a gas, oxygen in this case, is contained in a tube or confined chamber at a relatively low pressure level, or ambient pressure, and is quickly forced to the end of the tube or chamber, as a block of oxygen and compressed to a pressure level equal to the pressure of the service cart cylinder. (Always make it a practice to open the oxygen valve slowly.) Servicing cylinders when fully charged normally contain 2200 psi. The compression of this block of oxygen will cause work to be done on it which in turn will create a temperature rise that may reach 1200°F. Should any contaminant, such as oil, metal slivers, or pieces of rubber be present at the end of the tube, or chamber, a fire will very likely occur. The rapid temperature rise, due to adiabatic compression, can be reduced by a sized orifice at the upstream end of the servicing line/hose. The suggested hose in figure 12-14 has an orifice built into it for this purpose.

(2) A fire may be created by the inadvertent contamination of the oxygen servicing equipment and system. This can occur through the use of servicing hoses that are not compatible with high pressure oxygen. For example, the inner liner of certain hydraulic or pneumatic hoses are constructed of nitrile or neoprene rubber. These rubbers use plasticizers that with time, age, or heat, tend to "bloom" and surface an oily substance on the inner liner. The presence of this oily flammable plasticizer in the inner liner of the servicing hose can create a serious fire hazard. For this reason, a hose with an inner liner of polytetrafluoroethylene (teflon) should be used.

(3) A fire may be created by friction heat that is generated by high pressure oxygen escaping through a very small opening such as a leaking coupling nut, a pin hole in a line or hose, or a crack in a piece of equipment.

(4) A fire may be created by throttling a valve for a low flow condition in a high pressure system that will create friction heat within the valve itself which may be sufficient to ignite the valve seat material.

(5) A fire may be created by heat generated from sonic vibration. Sonic vibration can be generated when high pressure oxygen is transferred at a high velocity.

(6) A fire may be created by arcing of static electricity at a time when the atmosphere oxygen is enriched from escaping oxygen during the purging or filling operation.

(7) A fire may be created by unauthorized or makeshift equipment being used adjacent to the airplane or carried on board the airplane to service the oxygen system.

12.10.3 SERVICING PRECAUTIONS.

a. The following precautions apply to both on board servicing and cylinder removal. Where a removed cylinder is sent to a commercial oxygen shop for servicing, the precautions relating to servicing equipment may not apply.

(1) Service the system with the airplane outside the hangar, never inside. The airplane and all service equipment should be properly electrostatically grounded.

(2) Personnel who are smoking, carrying matches, cigarette lighters, or wearing exposed steel plates on their shoes should not be permitted within 50 feet of the airplane during the servicing operation.

(3) Open flames or spark producing equipment of any type should not be permitted within 50 feet of the airplane during the servicing operation.

(4) Other maintenance or operations should not be performed on the airplane while the oxygen system is being serviced nor should anyone not directly related to the servicing operation be on board the airplane during the servicing operation.

(5) Assure that all airplane electrical power is off and maintenance extension cords are disconnected prior to servicing. Do not permit connecting or disconnecting of ground power during the time of servicing the oxygen system.

(6) Complete cleanliness must be observed. Never permit oil, grease, or other combustible substance to come into contact with components of the airplane oxygen system or charging equipment. Avoid the use of flammable polish or protective compounds on surfaces adjacent to the oxygen supply compartment. Do not handle or work on any oxygen equipment or component of the oxygen system with dirty or oily hands, gloves, or tools or perform the charging operation wearing oily or greasy clothing. The servicing and maintenance equipment and tools should be kept free of dust and other contaminants at all times, and should not be used for other maintenance. Avoid storing flammable materials in the oxygen supply compartment.

(7) The operator should avoid wearing clothes made of synthetic fibers, such as nylon, orlon, and dacron when servicing the oxygen system to prevent arcing from static electricity. These fibers are usually found in wash and wear and permanently pressed type clothes. Clothes made of 100% cotton are ideal. Gaseous oxygen should not be directed at the body or clothing of any person.

CAUTION

Should a person be subjected to escaping oxygen he should not smoke or go near any heat or ignition source for a least 20 minutes.

(8) Assure that personnel performing the servicing operation are trained and experienced in oxygen handling. Should only one person be servicing the system, he should not do the job in an isolated area. Someone else should be within immediate calling distance in case of an emergency.

(9) Oxygen servicing equipment should be checked for accuracy, operation, wear and leaks on a scheduled basis. Lockheed recommends that the service hose be checked for leaks on a yearly basis or whenever a leak in the hose is suspected.

(10) Use only servicing equipment suitable for the aircraft breathing oxygen systems. Never interchange oxygen equipment intended or used for other gases or materials. Oxygen charging hoses should be kept clean, capped when not in use, and clearly marked "For Oxygen Use Only". See figure 12-14 for a description of any oxygen charging system showing minimum requirements.

(11) Ensure that only isopropyl alcohol is used to clean oxygen system components.

(12) Use only gaseous oxygen conforming to MIL-O-27210 Type I or its equivalent to service the oxygen system.

12.10.3.1 SERVICE HOSE LEAKAGE TEST. The hose assembly shall be pressurized with clean air or nitrogen to 3,000 psig pneumatic pressure with the assembly under clear water. The pressure shall be maintained for not less than 1 minute nor more than 15 minutes. Indication of rapid forming and breaking away of bubbles from any point on the outer surface of the hose or any part of the end fitting shall be cause for rejection. Alternate test method: The total leakage from any 12-inch section of the hose assembly shall be collected for 15 minutes. At the end of 15 minutes, the amount of gas accumulated shall not exceed 25 cc.

12.10.4 SERVICING PROCEDURES.

12.10.4.1 ON-BOARD SERVICING. On-board servicing, either at oxygen compartment or external servicing fitting.

a. Prior to the actual charging operations, the following should be accomplished and/or checked:

(1) Fire extinguishers available and ready.

(2) Oxygen service cylinder(s) have adequate pressure to service the airplane oxygen system.

(3) Airplane electrostatically grounded to earth ground.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

(4) Charging equipment electrostatically grounded to airplane and to same earth ground as airplane.

(5) All airplane electric power off and all external power cables disconnected:

(6) Airplane oxygen cylinder valve in closed position. (See Information Notes.)

(7) Charging equipment set up as follows :

Charging cylinder valve - closed

Pressure reducer valve - set a minimum outlet pressure

Shut-off valve - closed

Bleed valve - closed

b. Inspect and purge the service hose as follows:

(1) Remove plastic bag and dust plug from service hose and dust cap from airplane supply fill valve and inspect the service hose fitting and the airplane fill valve connection for evidence of contamination. Clean, if necessary, with a dry, lint free cloth. Replace dust cap on airplane supply cylinder valve until ready to make connection.

(2) Turn on the supply cylinder valves on the service cart, adjust the regulator for approximately 100 psi delivery pressure and open supply shut-off valve to permit a flow from the end of service hose. Readjust regulator to provide a brisk flow but below a frosting condition. Purge the service hose for a minimum of from one to two minutes. Shut off flow by adjusting the pressure reducer down stream (delivered) pressure to zero (0) psi and close the service cart shut-off valve. In the event of a discharge of fog or mist, the purging should be continued until the condition is eliminated.

CAUTION

Do not release oxygen where it can contact oil, grease, etc, or where it can contact any source of heat or open flame. Do not direct flow into interior or on exterior of airplane.

c. After purging the hose, immediately attach the service hose to either the oxygen cylinder supply fill valve (for filling at the oxygen compartment) or at the airplane service fitting (for remote filling). When the external servicing fitting is utilized, the end of the pressure fill line in the oxygen compartment must be connected to the cylinder fill valve. Always double wrench when installing or removing oxygen service hose or dust caps.

Note

When using multi-bottle carts, use only one cylinder at a time. When the cylinder pressure drops below usable pressure shut it off before opening the valve on another cylinder.

d. Check to ensure that the bleed valve on the service cart is closed, that the service carts supply cylinder valve is open, and then proceed as follows in the order as listed below:

(1) Open the service cart shut-off valve.

(2) Adjust the service cart reducer to the desired servicing pressure. See paragraph 12.10.7 for charge pressure at various temperatures.

Note

Do not fill the airplane oxygen supply cylinder too fast by adjusting service cart reducer to full open position. Instead use a step filling procedure by maintaining 200 psi to 500 psi differential between the airplane cylinder pressure and regulated filling pressure.

(3) Check for leaks at the service hose to airplane fill valve connection and any other area by listening and by using a leak detector fluid per MIL-L-25567A (ASG) Type I or equivalent. If leaks are detected, close service cart cylinder valve or shut-off valve and open service cart bleed valve. After pressure in service hose has escaped, close bleed valve. Repair leak as necessary using approved method/procedures and materials before proceeding with step c again.

CAUTION

Ports and lines should be capped after disconnecting to prevent entry of contaminants to eliminate need to repurge the system.

e. When the airplane supply cylinder is serviced to the desired pressure, return the oxygen service cart to normal condition as follows:

(1) Close service cart cylinder valve.

(2) Adjust reducer to minimum delivery pressure setting.

(3) Adjust bleed valve to relieve internal pressure in service cart hose and lines.

(4) Adjust reducer to relieve spring tension on its diaphragm by adjusting the reducer to the off position.

(5) Remove service hose from airplane supply fill valve and replace dust cap on fill valve and check for leaks.

(6) Replace the service cart hose dust plug, close bleed valve and shut-off valve, and place clean plastic bags over end of hose and bleed valve outlet before storing hose on cart. The service cart should be stored in a clean area with a minimum of dust settlement and grounded.

CAUTION

Do not coil a teflon hose too tight as it may kink and develop a leak. A kinked hose must be replaced.

f. Before restoring airplane electrical power the oxygen system should be sniff tested as follows:

Place the crew regulator in "ON", "100% OXYGEN" and "EMERGENCY" positions and permit oxygen to flow from the mask or regulator to mask extension hose for approximately 10 to 15 seconds. Breathe from the mask or hose for an additional 5 seconds. There should be no odor, irritation of the nose or throat, or any evidence of smoke or vapor coming from the mask or hose. Any of the above conditions is indication of a burned valve seat or other contamination in the system, which must be corrected.

g. If the airplane is to be flown within eight hours, open the cylinder shut-off, reducer valves. If the airplane is not to be flown within eight hours, leave the valves in the off position.

12.10.4.2 SERVICING BY CYLINDER REMOVAL. Prior to cylinder removal, the following should be accomplished or checked:

a. Fire extinguishers available and ready.

b. Airplane electrostatically grounded to earth ground.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

c. All airplane electric power off and all external power cables disconnected.

d. Cylinder valve closed. If the airplane is configured with high pressure capillary lines connected to gages on the cockpit bulkhead, ensure that the gages read zero pressure before cylinder removal.

12.10.4.2.1 CYLINDER REMOVAL.

a. Disconnect all lines to the cylinder mounted pressure reducer valve and plug them or protect them with a clean small plastic bag secured around the tube with a tight rubber band or equivalent.

b. Loosen the cylinder retention strap T-bolt nut approximately 10 turns and lift the T-bolt out of the detent so that the straps may be spread apart to clear cylinder.

c. Lift the cylinder out of the cradles horizontally.

12.10.4.2.2 CYLINDER INSTALLATION.

a. Install the serviced or spare cylinder by repeating 12.10.4.2.1, steps a, b, and c in reverse order.

b. Leak Check all connections and perform sniff test in accordance with Servicing Procedure.

c. If the airplane is to be flown within eight hours, leave the cylinder shut-off valve open. If the airplane is not to be flown within eight hours, the shut-off valves should be in the off position.

12.10.5 INFORMATION NOTES.

a. The purpose of closing the airplane oxygen cylinder pressure reducer shut-off valve during the on-board charging operation is to isolate the remaining portion of the aircraft system from the cylinder being charged and from the charging equipment to provide some degree of protection should an emergency condition occur.

b. The charging refill valve on the airplane cylinder is a simple spring loaded valve (similar to an automobile tire valve) that is connected directly to the inside of the cylinder. This completely bypasses the regulating section of the valve. The charging valve is open automatically when the differential pressure from the servicing equipment exceeds the internal pressure of the airplane cylinder plus the valve's own spring force. No provisions have been made for mechanically opening the charging valve. Should the charging valve develop a leak, both the manufacturer of the valve and Lockheed recommend that the complete refill valve assembly be replaced rather than replacing the valve core.

12.10.6 PURGING THE OXYGEN SYSTEM.

Purge the oxygen system when any part of the system has been left open or when the system pressure has been reduced to zero. Purge the system as follows, using only Specification MIL-O-27210, Type I oxygen.

a. Drop the passenger masks by turning the cylinder valve on and positioning the selector knob on the PASSENGER OXYGEN control panel to "MAN" Then turn the cylinder valves off.

b. On the OXYGEN REGULATOR panels, select "EMERGENCY" and "100% OXYGEN" and turn the supply lever to "ON."

c. Disconnect the supply line flexible hose at the structure end. Slightly open the valve on the oxygen supply cart to allow flow through the supply line for two minutes to purge the line, and then close the valve.

d. Connect the supply line of the oxygen supply cart to the disconnected airplane system supply line. Pull the lanyards on the lanyard-operated valves to open the outlets in the system.

e. Slowly open the valves on the oxygen supply cart, and apply a pressure of 50 PSI to the airplane system for 10 minutes.

f. With the airplane outlets still open, disconnect the cart supply line, and connect the airplane system supply line flexible hose. Turn the cylinder valve on slowly to apply cylinder pressure to the system. Maintain this pressure for 2 minutes, then turn the cylinder valve off and close all outlets.

12.10.7 OXYGEN CYLINDERS, PRESSURE-TEMPERATURE.

Temperature (Degrees F)	Charge Pressure (PSI)	Temperature (Degrees F)	Charge Pressure (PSI)
-40	1287	50	1708
-30	1334	60	1754
-20	1381	70	1800
-10	1428	80	1846
0	1475	90	1892
10	1522	100	1938
20	1568	110	1984
30	1615	120	2029
40	1661	130	2075

12.10.8 SUGGESTED MATERIAL LIST FOR ASSEMBLY OF OXYGEN SERVICE CART. The following is a list of materials required to assemble an oxygen service cart. See figure 12-13A for recommended arrangement of components.

a. Leak test compound, oxygen systems per MIL-L-25567A (ASG) Type 1

Suppliers: (1) American Gas and Chemical Inc.
New York, N.Y. 10021 (or equal)

(2) Puritan Bennett Aero Systems
111 Penn Street
El Segundo, CA 90245 (or equal)

b. Valve, Shut-off and/or Bleed (2 required)

(1) To be mounted on front face of panel (4 hole)
Part No. OV601-1P (Ports 1/8 NPT female)
Part No. OV601-4T (AND10050-4) 1/4 in. flared tube boss

Supplier: Robbins Aviation, Inc.
3817 Santa Fe Avenue
Vernon, CA 90058 (or equal)

(2) Optional Mounting:
To be mounted behind panel with stem through panel
Part No. INSG103-1P (Ports 1/8 NPT female)

Supplier: Robbins Aviation, Inc.
3817 Sante Fe Avenue
Vernon, CA 90058 (or equal)

c. Refill (Charging) valve assembly

Part No. F1263-5

Supplier: The ARO Corporation
3695 Broadway
Buffalo, NY 14227

d. Cylinder, Oxygen Purifier Type MB-1 or NOP-1
Per: MIL-C-4810D
(Used with MIL-C-49520 Type MA-1 Cartridge)

Supplier: Puritan Bennett Aero Systems
111 Penn Street
El Segundo, CA 90245 (or equal)

e. Cartridge, Oxygen Purifier, Type MA-1
Per: MIL-C-4952D
(Used with Type MB-1 or NOP-1 cylinder)

Supplier: Puritan Bennett Aero Systems
111 Penn Street
El Segundo, CA 90245 (or equal)

f. Gage (0-3000 PSI) (2 required)

Marked on face as follows: "Oxygen use only - use no oil"

Supplier: Manning, Maxwell & Moore
Stratford, Conn. (or equal)

g. Thread Anti-seize Tape (Preferred)
Per MIL-T-27730

Suppliers: Phillips Sales Co.
Industrial Tape Supply
1236 Logan Circle, N.W.
Atlanta, Ga. 30318 (or equal)
Telephone: (404) 351-3190

Dodge Industries
Hoosick Falls, N.Y. 12090 (or equal)

Thread Anti-seize Compound (acceptable)
Per MIL-T-5542B

Supplier: Puritan Bennett Aero Systems
111 Penn Street
El Segundo, CA 90245 (or equal)

Note

Anti-seize is only to be used on pipe threads. When applying anti-seize do not apply it to the first 2 threads. Remove all old anti-seize left from previous installation before making new installation.

h. Pressure Reducer
Part No. 1342X4P
Consolidated Controls Corp.
2338 Alaska Avenue
El Segundo, CA 90245 (or equal)

Note

Operating range: 0 to 3000 psi
Inlet and outlet ports: 1/4 inch NPT (female)
Cleaned for oxygen use
Automatic self relieving

i. Aircraft Oxygen System Service Hose

The hose is constructed with an extruded polytetrafluoroethylene (Teflon) inner core that is reinforced with two layers of braided corrosion resistant stainless steel wire over the Teflon core and an outer polyolefin scuff sleeve over the reinforcing wire braid. Both ends of the hose are protected from being kinked by a 12 inch long spring guard adjacent to both end fittings. The inlet end of the hose will mate with a MS33656-5 flared tube end fitting and has a 0.070 orifice built into it to help reduce adiabatic compression at the outlet end of the hose when it is connected to the airplane supply cylinder refill valve. The outlet end of the hose will mate with a AND10089-3 end fitting which is the type of fitting on the airplane oxygen cylinder refill valve. The outlet end of the hose is also provided with a permanent dust plug attached to the hose by means of a chain. When selecting the hose length (see examples below) always remember that the service cart, or hose inlet, should always be a minimum of 8 to 10 feet from the cylinder or airplane to be serviced. The hose should be one continuous length as supplied from the vendor and not spliced to other hoses. No other connector or adapter is required to connect to the airplane oxygen supply cylinder fill valve when this hose is used.

Note

The supply cylinder fill valve part number is F1263-5.

Hose Supplier: Stratoflex Inc.
220 Roberts Cutoff
Fort Worth, TX 76114

Stratoflex example part number:

166023 - 4 - 240 - 2

2 — Assembly length in eights of an inch (2/8")
240 — Assembly length in inches (240")
4 — Hose Size
166023 — Basic Part No.

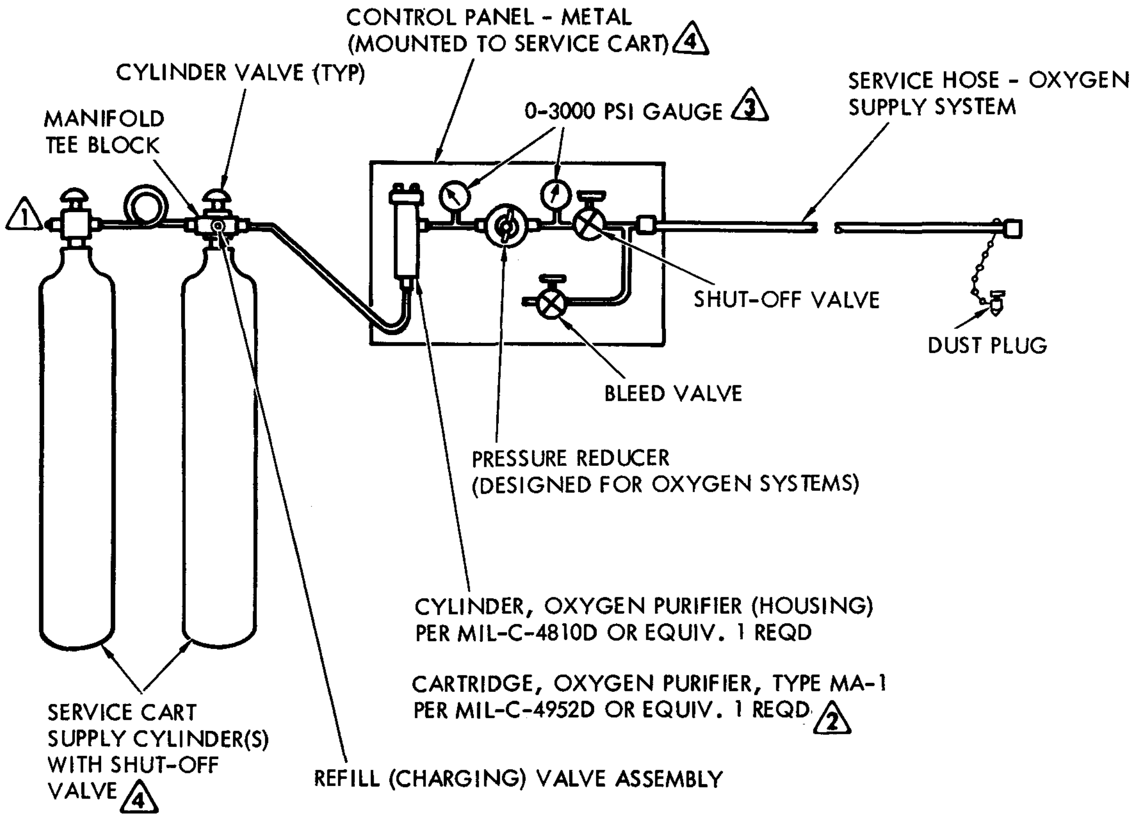
Supplier: Titeflex
A Division of Atlas Corp.
603 Hendee Street
Springfield, Mass. 01109

Titeflex example part number:

S5759 - 240 - 2

2 — Assembly length in eighths of an inch (2/8)
240 — Assembly length in inches (240")
S5759 — Basic Part No.

j. Oxygen Cylinder - Quantity as required


CONTROL PANEL - METAL (MOUNTED TO SERVICE CART) △4
CYLINDER VALVE (TYP)
MANIFOLD TEE BLOCK
0-3000 PSI GAUGE △3
SERVICE HOSE - OXYGEN SUPPLY SYSTEM
△1
SHUT-OFF VALVE
DUST PLUG
BLEED VALVE
PRESSURE REDUCER (DESIGNED FOR OXYGEN SYSTEMS)
CYLINDER, OXYGEN PURIFIER (HOUSING) PER MIL-C-4810D OR EQUIV. 1 REQD
CARTRIDGE, OXYGEN PURIFIER, TYPE MA-1 PER MIL-C-4952D OR EQUIV. 1 REQD △2
SERVICE CART SUPPLY CYLINDER(S) WITH SHUT-OFF VALVE △4
REFILL (CHARGING) VALVE ASSEMBLY

NOTE

△1 CYLINDER VALVE RUPTURE DISK - DO NOT LOOSEN NUT OR TAMPER WITH.

△2 REPLACE THE CARTRIDGE AFTER 16 CYLINDERS OF OXYGEN HAVE PASSED THROUGH IT (INSTALLED INSIDE OF PURIFIER CYLINDER)

△3 GAGES TO BE INSTALLED IN THE LINE ADJACENT TO THE REDUCER ON THE UP-STREAM & DOWNSTREAM SIDES.

△4 THIS IS ALWAYS OUTSIDE THE AIRPLANE.

JS-1-2-X0/2-1213A

Figure 12-13A. Recommended Oxygen Service Cart Arrangement

SECTION 12.11

SERVICING THE LAVATORY WATER SYSTEM

Contents Page

LAVATORY WATER SYSTEM (AIRPLANES 5001 thru 5045) 12-45

LAVATORY WATER SYSTEM (AIRPLANES 5046 AND UP) 12-45

TOILETS 12-45

FIGURES
12-14. Lavatory Water Servicing Panel 12-46

12.11.1 LAVATORY WATER SYSTEM (AIRPLANES 5001 THRU 5045). The lavatory water system should be serviced before each flight. The access door for the servicing provisions is on the right-hand side of the fuselage aft of the nacelles. (See figures 12-1 and 12-14.) Service the system by the following procedure.

a. Open the access door and connect an ordinary garden hose to the filler coupling.

b. Rotate the tee handle 90 degrees and pull it out until it stops.

c. Fill the system with water until water flows from the overflow outlet.

d. Push the tee handle in and rotate it 90 degrees to lock it.

e. Disconnect the hose from the filler coupling and close the access door.

12.11.2 LAVATORY WATER SYSTEM (AIRPLANES 5046 AND UP). The lavatory water system should be serviced according to the recommendations of the interior installer.

12.11.3 TOILETS. Toilets installed in JetStars may be either the chemical or the flushing type. Both types should be serviced according to the toilet manufacturer's instructions.

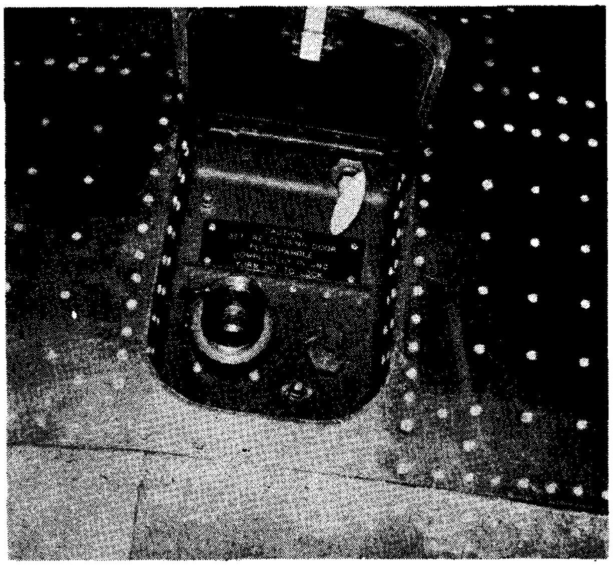


Figure 12-14. Lavatory Water Servicing Panel

JS-2-2-1214

SECTION 12.12
LUBRICATION

Contents Page

GENERAL 12-47

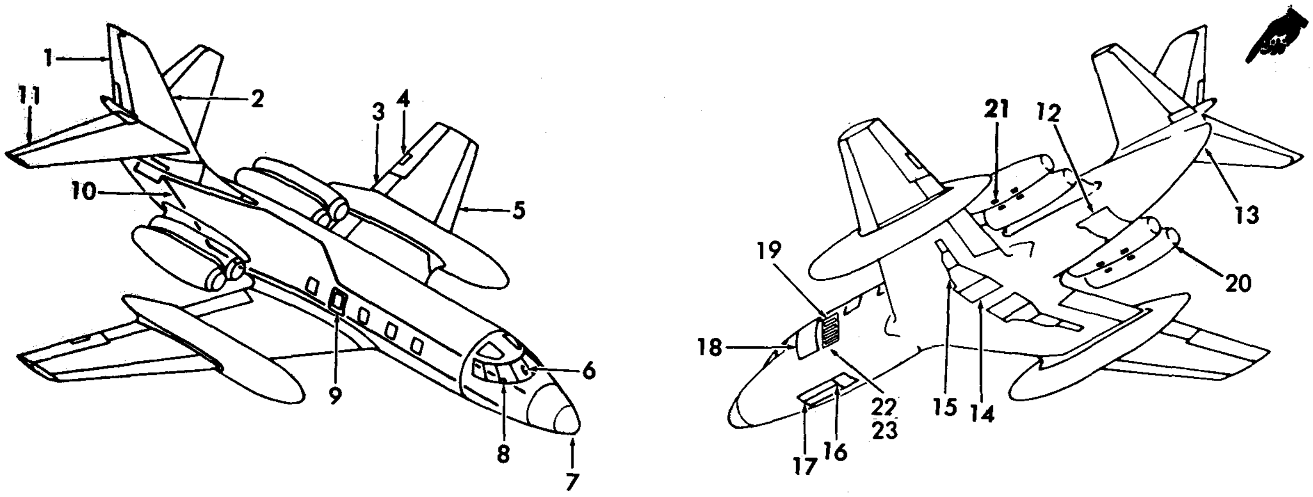
LUBRICATION DURING PERIODS OF INACTIVITY 12-47

FIGURES

12-15. Lubrication of the Airplane 12-48
12-16. Lubrication of the Nose Landing Gear Door Actuating Mechanism 12-52
12-17. Lubrication of the Nose Landing Gear 12-53
12-18. Lubrication of the Main Landing Gear Inboard Door and Release Mechanism 12-55
12-19. Lubrication of the Main Landing Gear and Outboard Doors 12-57
12-20. Lubrication of the Control Cables 12-59
12-21. Lubrication of the Leading Edge Flap System 12-61
12-22. Lubrication of the Trailing Edge Flap System 12-63
12-23. Lubrication of the Aileron and Aileron Tab Systems 12-65
12-24. Lubrication of the Empennage 12-66
12-25. Lubrication of the Rudder and Rudder Trim Tab Systems 12-67
12-26. Lubrication of the Elevator 12-68
12-27. Lubrication of the Speed Brake 12-69
12-28. Lubrication of the Emergency Exit 12-71
12-29. Lubrication of the Removable Nose Cone 12-72
12-30. Lubrication of the Airborne Ladder (Airplanes 5001 through 5065) 12-73
12-30A. Lubrication of Automatic Stair Assembly (Airplanes 5066 and Up) 12-74
12-31. Lubrication of the Entrance Door (Airplanes 5001 through 5065) 12-74A
12-31A. Lubrication of the Entrance Door (Airplanes 5066 and Up) 12-74B
12-31B. Lubrication of the Automatic Stair Gearbox Assembly (Airplanes 5066 through 5120) 12-74C
12-31C. Lubrication of Manual Stair Gearbox Assembly (Airplanes 5121 and Up) 12-74D
12-32. Lubrication of the Clear Vision Windshield 12-75
12-33. Lubrication of the Pilots' Seats 12-76
12-34. Lubrication of the Drag-Chute Mechanism (Airplanes 5001 through 5120 and/or Those Airplanes Maintaining Operable Drag-Chute Systems) 12-77
12-35. Lubrication of the Thrust-Reverser 12-78
12-36. Lubrication of Nacelle Latches 12-79

12.12.1 GENERAL. Lubricate the airplane in accordance with the instructions given in figure 12-15. This figure gives special lubrication instructions, a lubrication index and a locator diagram which references other figures that show lubrication points in detail, together with application and frequency symbols. The lubrication index lists types of lubricants and defines application and frequency symbols.

12.12.2 LUBRICATION DURING PERIODS OF INACTIVITY. For out of service periods of more than 3 months, the aircraft should be lubricated at the beginning of the out of service period and also prior to being returned to service. When outdoor storage is used, a periodic lubrication of all areas not made inaccessible by some preservation seal or cover should be accomplished at least once every 3 months.



1. RUDDER AND RUDDER TRIM TAB (SEE FIGURE 12-25)
2. EMPENNAGE (SEE FIGURE 12-24)
3. TRAILING EDGE FLAPS (SEE FIGURE 12-22)
4. AILERON AND AILERON TRIM TAB (SEE FIGURE 12-23)
5. LEADING EDGE FLAPS (SEE FIGURE 12-21)
6. PILOT'S SEATS (SEE FIGURE 12-33)
7. REMOVABLE NOSE CONE (SEE FIGURE 12-29)
8. CLEAR VISION WINDSHIELD (SEE FIGURE 12-32)
9. EMERGENCY EXITS (SEE FIGURE 12-28)
10. CONTROL CABLES (SEE FIGURE 12-20)
11. ELEVATOR (SEE FIGURE 12-26)
12. SPEED BRAKE (SEE FIGURE 12-27)
13. DRAG CHUTE RELEASE MECHANISM (SEE FIGURE 12-34)
14. MAIN LANDING GEAR INBOARD DOORS (SEE FIGURE 12-18)
15. MAIN LANDING GEAR AND OUTBOARD DOORS (SEE FIGURE 12-19)
16. NOSE LANDING GEAR (SEE FIGURE 12-17)
17. NOSE LANDING GEAR DOOR ACTUATING MECHANISM (SEE FIGURE 12-16)
18. ENTRANCE DOOR (SEE FIGURE 12-31 & 12-31A)
19. AIRBORNE LADDER (SEE FIGURE 12-30 & 12-30A)
20. THRUST REVERSERS (SEE FIGURE 12-35)
21. NACELLE ACCESS DOOR LATCHES (SEE FIGURE 12-36)
22. AUTOMATIC STAIR GEAR BOX ASSEMBLY (SEE FIGURE 12-31B)
23. MANUAL STAIR GEAR BOX ASSEMBLY (SEE FIGURE 12-31C)

SPECIAL SERVICE NOTES

1. CLEAN FITTINGS BEFORE LUBRICATION. INJECT LUBRICANT UNTIL ALL OLD GREASE IS DISPLACED. WIPE OFF EXCESS.
2. USE STEWART-WARNER CORPORATION NOZZLE NO. 314150 ON GREASE GUNS WHEN ADDING LUBRICANT TO A NAS516 GREASE FITTING.
3. ASSURE THAT ALL SURFACES TO BE LUBRICATED ARE CLEAN AND FREE FROM MOISTURE, SOLVENTS, AND OTHER CONTAMINANTS BEFORE LUBRICATING.
4. NEVER APPLY A LUBRICANT OVER A RUSTED OR CORRODED SURFACE.
5. BE CAREFUL WHILE HANDLING HIGHLY POLISHED SURFACES PRIOR TO LUBRICATING, TO AVOID CONTAMINATING THE SURFACES WITH BODY MOISTURE OR OTHER AGENTS THAT COULD CAUSE CORROSION.
6. NEVER HEAT OR MELT LUBRICATING GREASE TO AID IN ITS APPLICATION.
7. BE SURE TO LUBRICATE ALL POINTS REQUIRED, USING THE CORRECT LUBRICANT IN THE SPECIFIED METHOD OF APPLICATION WITH ADHERENCE TO THE FREQUENCY REQUIREMENTS.
8. AFTER LUBRICATION, REMOVE ALL EXCESS LUBRICANT.

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Figure 12-15. Lubrication of the Airplane (Sheet 1 of 7)

APPLICATION SYMBOLS			
✋	🖌		
HAND	BRUSH	CAN	GREASE GUN

FREQUENCY SYMBOLS			
□	○	⬡	△
100 FLT HRS OR 30 DAYS	400 FLT HRS OR 6 MONTHS	1000 FLT HRS OR 1 YEAR	3000 FLT HRS OR 3 YEARS

LUBRICANTS

SYMBOL	SPECIFICATION	MANUFACTURER'S DESIGNATIONS	SOURCE
GIA	MIL-G-23827		
		AO SYN Grease 354	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		Supermil Grease A-72832	Amoco Oil Co. 200 E Randolph Dr. Chicago, IL 60601
		Braycote 627S	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Castrolease A1 Grease	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Royco 27A	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Royco 27A1	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell Grease 7	Shell Oil Co. P O Box 2463 Houston, TX 77001
		Southwest Grease No. 16215	Southwest Petro-Chem, Inc. 1400 S Harrison Olathe, KS 66061
		TG-11900 Low Temp Grease EP	Texaco, USA P O Box 52332 Houston, TX 77052
GMD	MIL-G-21164		
		PQ Moly Grease 353	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105

Figure 12-15. Lubrication of the Airplane (Sheet 2 of 7)

LUBRICANTS			
SYMBOL	**SPECIFICATION**	**MANUFACTURER'S DESIGNATIONS**	**SOURCE**
GMD (CONT.)	MIL-G-21164 (CONT.)	Braycote 664	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Everlube 211-G Moly Grease	E/M Corporation P O Box 2200 West Lafayette, IN 47906
		Royco 64	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell Grease 17	Shell Oil Co. P O Box 2463 Houston, TX 77001
LP	COMMERCIAL		
		Lubriplate No. 130AA	Fiske Bros Refining Co. 121 Lockwood St. Newark, NJ 07105
LTG	MIL-L-7808		
		AOSyn Jet III	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		PQ Turbine Oil 4236	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		PQ Turbine Oil 4706	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		PQ Turbine Oil 8365	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		PQ Turbine Oil 9900	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		PQ 4707	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		Castrol 399	Burmah-Castrol (UK) Ltd. Pipers Way Swindon Wiltshire SN3 1RE, UK

Figure 12-15. Lubrication of the Airplane (Sheet 3 of 7)

	LUBRICANTS		
SYMBOL	SPECIFICATION	MANUFACTURER'S DESIGNATION	SOURCE
LTG (CONT.)	MIL-L-7808 (CONT.)	Brayco 880	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Castrol 399	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Castrol 399	Castrol Canada, Inc. 3660 Lakeshore Blvd W Toronto M8W 1P2, Canada
		Delta Jet Lube 7808	Delta Petroleum Co. P O Box 10397 Jefferson, LA 70181
		Exxon Turbo 2389	Exxon Company USA P O Box 2180 Houston, TX 77252
		Exxon Turbo 2391	Exxon Company USA P O Box 2180 Houston, TX 77252
		HATCOL 1278	Hatco Chemical Corp. King George Post Rd. Fords, NJ 08863
		HATCOL 1280	Hatco Chemical Corp. King George Post Rd. Fords, NJ 08863
		RM-248A	Mobil Research & Dev. Corp. Paulsboro Research Lab. Paulsboro, NJ 08066
		RM-272A	Mobil Research & Dev. Corp. Paulsboro Research Lab. Paulsboro, NJ 08066
		TURBONYCOIL 160	NYCO S. A. 66 Avenue des Champs-Elysees 51 Rue der Ponthieu 75008 Paris, France
		Royco 808	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell Turbine Oil 308	Shell International Petro. Shell Centre London SE1 7NA, UK

Figure 12-15. Lubrication of the Airplane (Sheet 4 of 7)

LUBRICANTS

SYMBOL	SPECIFICATION	MANUFACTURER'S DESIGNATION	SOURCE
LTG (CONT.)	MIL-L-7808 (CONT.)	Technolube SYN TURBO No. 3	Technolube Products Co. 5814 E 61st St. Los Angeles, CA 90040
MDS	MIL-M-7866		
		Brayco 866	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Moly-Disulfide Powder No. 2	Climax Molybdenum Co. 1 Greenwich Plz Greenwich, CT 06830
		Molycote Z	Dow Corning Corp. P O Box 1-M Midland, MI 48640
		Electro-Moly, Grade 2	Electro-Film, Inc. P O Box 3930 N Hollywood, CA 91605
		Royco 66MS	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
OGP	MIL-L-7870		
		PQ Rust Preventive No. 107	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		Winsor Lube L-1018	Anderson Oil & Chem Co. Portland, CT 06480
		Brayco 363	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Convoy BH 7870	Convoy Oil Corp. 1412 Front St. Philadelphia, PA 19122
		OCTOIL 70	Octagon Process, Inc. 596 River Rd. Edgewater, NJ 07020
		Petrotect 7870A	Penreco 106 S Mail St. Butler, PA 16001
		Royco 363	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936

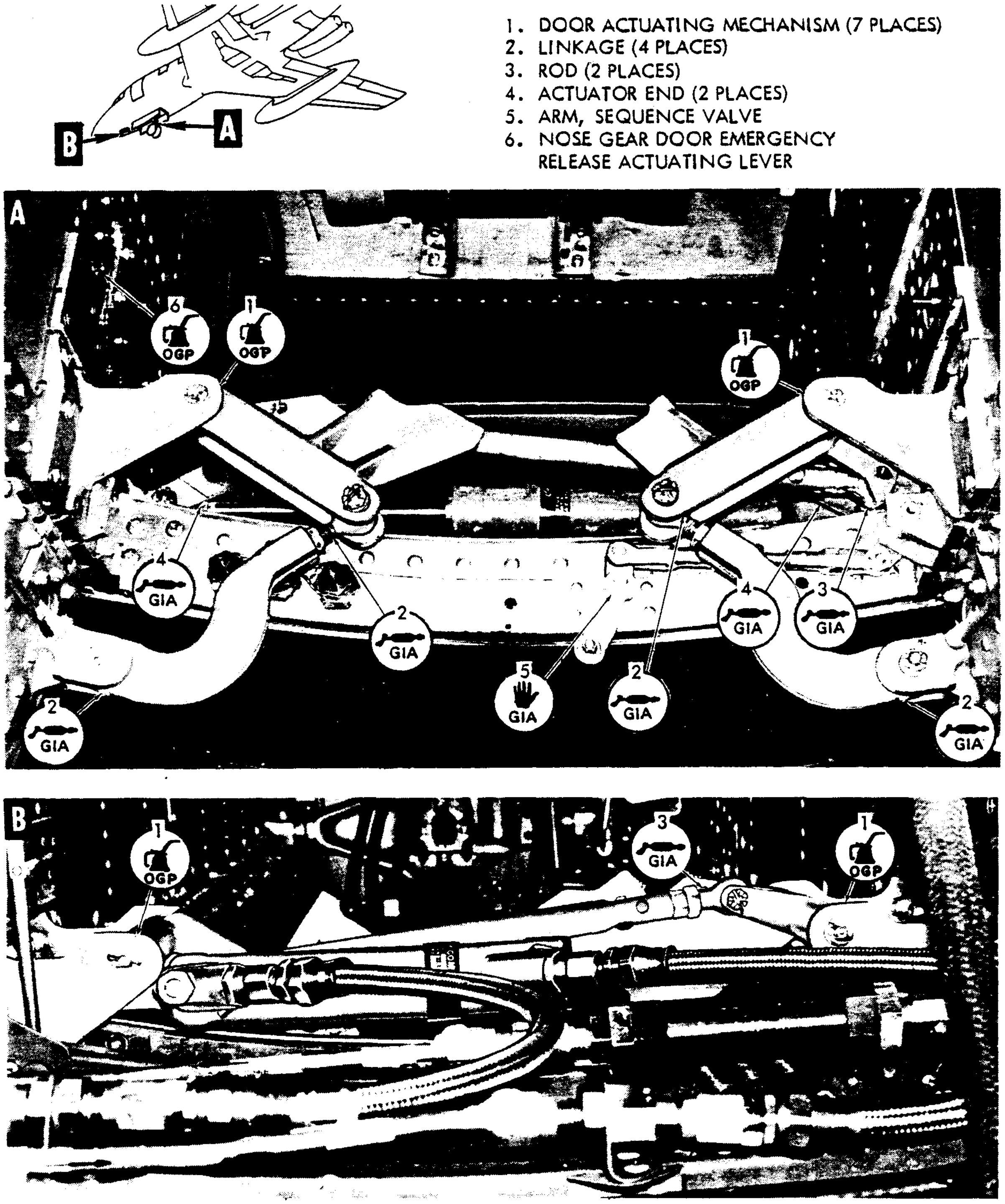
Figure 12-15. Lubrication of the Airplane (Sheet 5 of 7)

LUBRICANTS			
SYMBOL	**SPECIFICATION**	**MANUFACTURER'S DESIGNATION**	**SOURCE**
OGP (CONT.)	MIL-L-7870 (CONT.)	Royco 363A	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell Fluid 3	Shell International Petro. Shell Centre London SE1 7NA, UK
OGR	MIL-L-6086		
		Bruncor L, Grade L	Bruncor Corp. 632 S Front St. Elizabeth, NJ 07202
		Bruncor M, Grade M	Bruncor Corp. 632 S Front St. Elizabeth, NJ 07202
		Aeroshell 5 L-A, Grade L	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell 5 M-A, Grade M	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Royco 586L, Grade L	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Royco 586M, Grade M	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Royco 586V, Grade L	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Royco 586W, Grade M	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell 5 L-A, Grade L	Shell International Petro. Shell Centre London SE1 7NA, UK

Figure 12-15. Lubrication of the Airplane (Sheet 6 of 7)

LUBRICANTS			
SYMBOL	**SPECIFICATION**	**MANUFACTURER'S DESIGNATION**	**SOURCE**
OGR (CONT.)	MIL-L-6086 (CONT.)	Aeroshell 5 L-A, Grade M	Shell International Petro. Shell Centre London SE1 7NA, UK
		Royco 586W, Grade M	Shell International Petro. Shell Centre London SE1 7NA, UK
		GB 3896L, Grade L	Witco Chemical Corp. P O Box 5446 Oildale, CA 93388
		GB 3896M, Grade M	Witco Chemical Corp. P O Box 5446 Oildale, CA 93388
WTR	MIL-G-81322		
		PQ Aircraft Grease 395	American Oil & Supply Co. 238 Wilson Ave. Newark, NJ 07105
		Braycote 622	Burmah-Castrol, Inc. Bray Products Div. 16815 Von Karman Ave., Suite 202 Irvine, CA 92714
		Chevron W-T	Chevron USA, Inc. 575 Market St. San Francisco, CA 94574
		Convoy 322H	Convoy Oil Corp. 1412 Front St. Philadelphia, PA 19122
		322 Grease	Exxon Company, Intl. 200 Park Ave. Florham Park, NJ
		Mobilgrease 28	Mobil Research & Dev. Corp. Paulsboro Research Lab. Paulsboro, NJ 08066
		Royco 22	Royal Lubricants Co. P O Box 518 1 Merry Lane East Hanover, NJ 07936
		Aeroshell Grease 22C	Shell International Petro. Shell Centre London SE1 7NA, UK
		Aeroshell Grease 22	Shell Oil Co. P.O. Box 2463 Houston, TX 77001

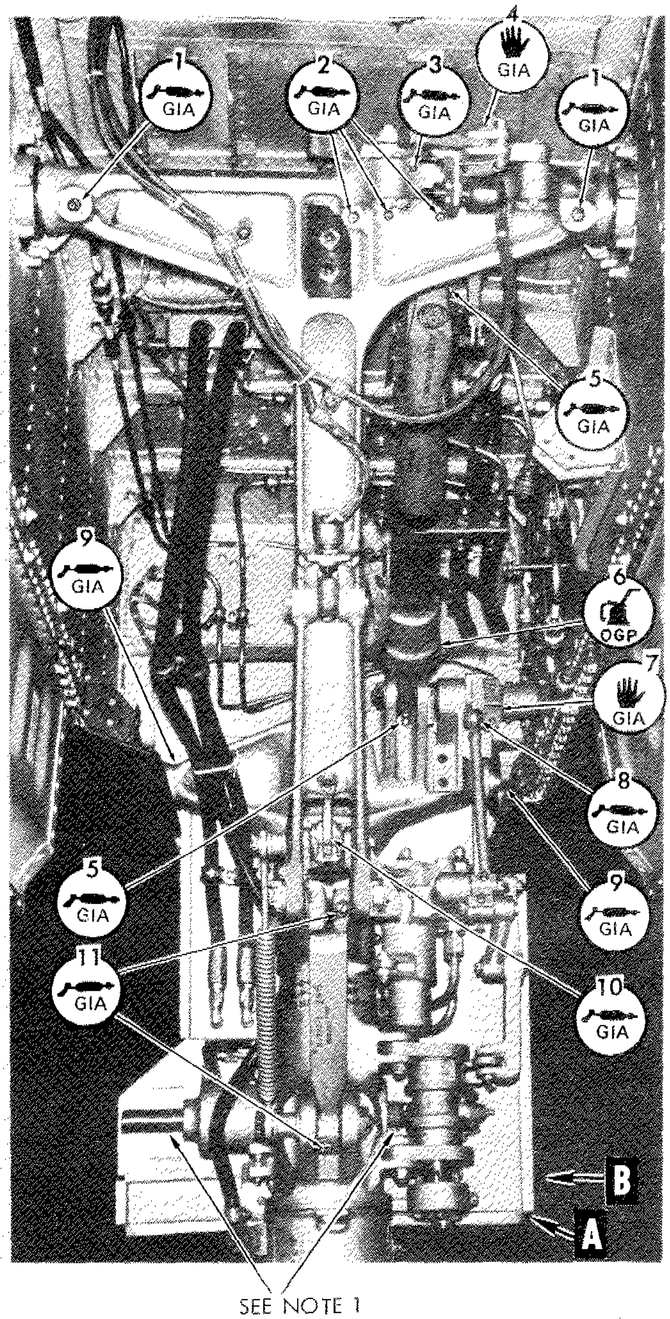
Figure 12-15. Lubrication of the Airplane (Sheet 7 of 7)



1. DOOR ACTUATING MECHANISM (7 PLACES)
2. LINKAGE (4 PLACES)
3. ROD (2 PLACES)
4. ACTUATOR END (2 PLACES)
5. ARM, SEQUENCE VALVE
6. NOSE GEAR DOOR EMERGENCY RELEASE ACTUATING LEVER

JS-1-2-X0/2-1216

Figure 12-16. Lubrication of the Nose Landing Gear Door Actuating Mechanism



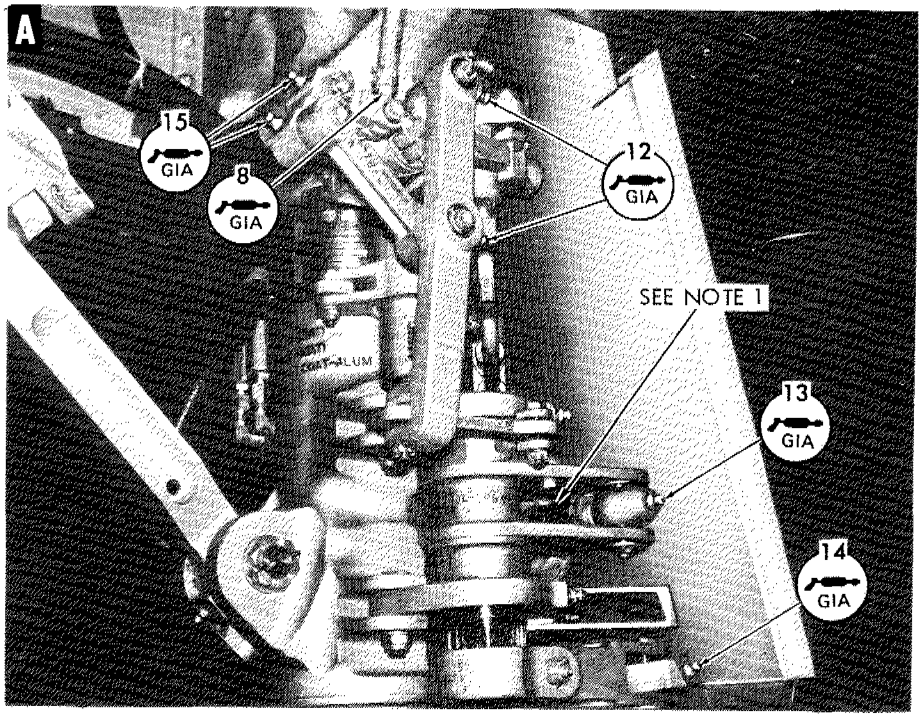
1. DRAG LINK TRUNNIONS (2 PLACES)
2. UPPER DRAG LINK UPLOCK FITTINGS (3 PLACES)
3. UPLOCK LATCH
4. ROD AND ARM GEAR UPLOCK LINKAGE
5. ACTUATOR CYLINDER (2 PLACES)
6. ACTUATOR CYLINDER PISTON WIPER
7. STEERING TORQUE TUBE
8. STEERING CONTROL ROD (2 PLACES)
9. SHOCK STRUT TRUNNION (2 PLACES)
10. LATCH, DOWN LOCK
11. LOWER DRAG LINK ATTACH POINT (2 PLACES)
12. STEERING CONTROL ARM (2 PLACES)
13. STEERING PISTON ROD END
14. STEERING LINK (2 PLACES)
15. STEERING CRANK (2 PLACES)
16. SHOCK STRUT TORQUE ARMS (2 PLACES)
17. EXPOSED SHOCK STRUT PISTON
18. SHOCK STRUT PISTON WIPER
19. STEERING COLLAR

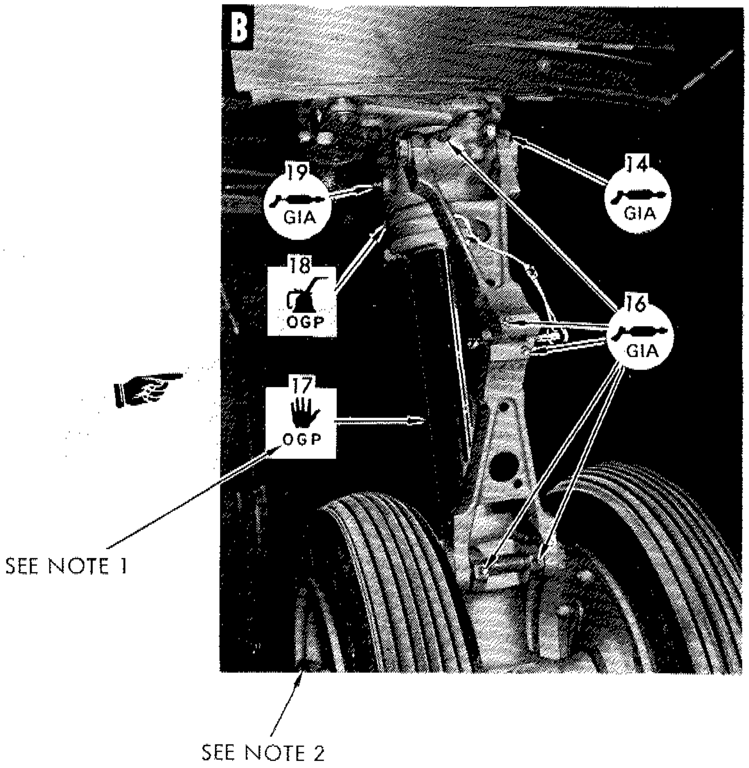
NOTE

1. BEFORE EACH FLIGHT, WIPE PISTON CLEAN OF DIRT AND GREASE AND APPLY LIGHT FILM OF OGP (MIL-L-7870) OIL.
2. AT EACH WHEEL CHANGE, CLEAN OUT ALL OLD GREASE AND REPACK WHEEL BEARINGS AND HUBCAPS WITH WTR. CLEAN AND APPLY LIGHT COAT OF WTR TO AXLE ALSO.

JS-1-2-X0/8-1217-1

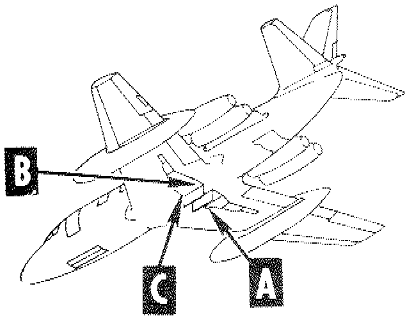
Figure 12-17. Lubrication of the Nose Landing Gear (Sheet 1 of 2)



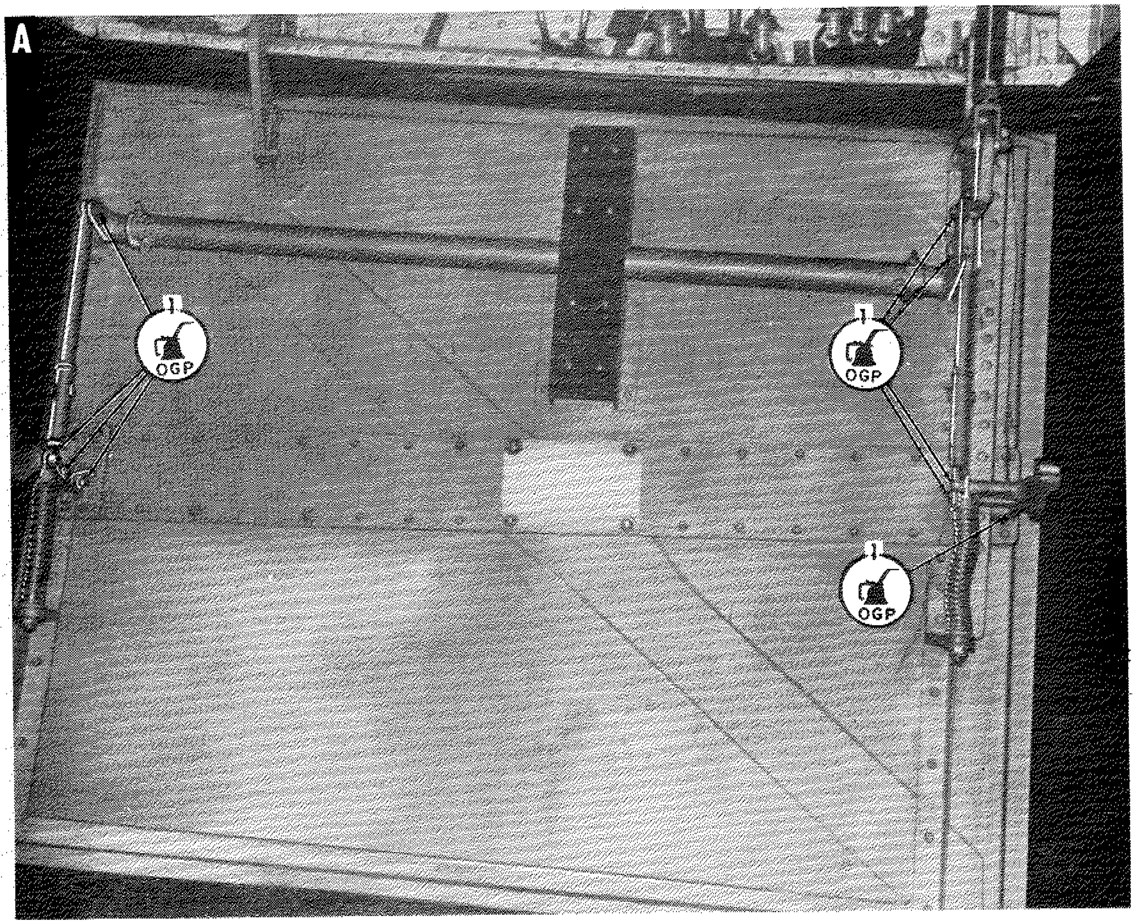


JS-1-2-X0/8-1217-2

Figure 12-17. Lubrication of the Nose Landing Gear (Sheet 2 of 2)

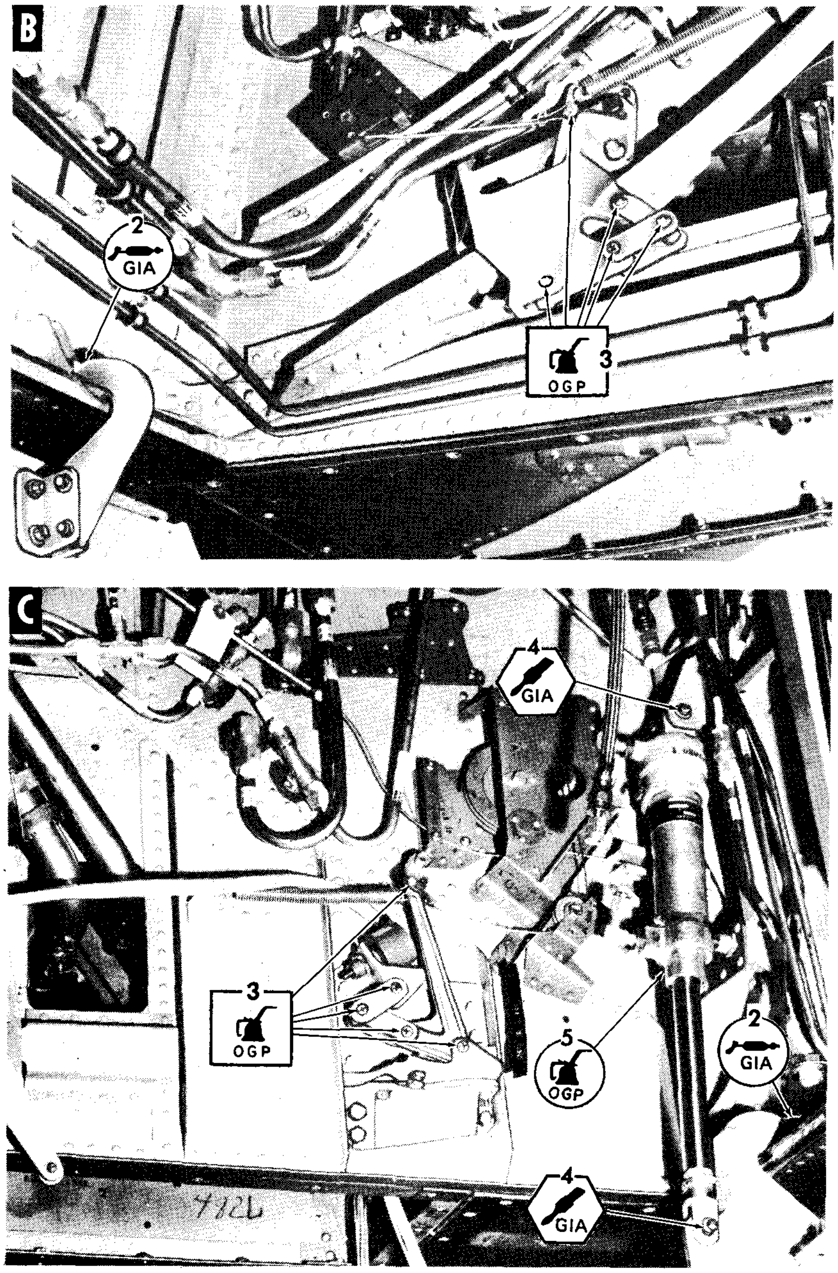


1. INBOARD DOOR (10 PLACES)
2. INBOARD DOOR HINGES (2 PLACES)
3. INBOARD DOOR RELEASE MECHANISM (10 PLACES)
4. INBOARD DOOR ACTUATOR BUSHING (2 PLACES)
5. INBOARD DOOR ACTUATOR WIPER



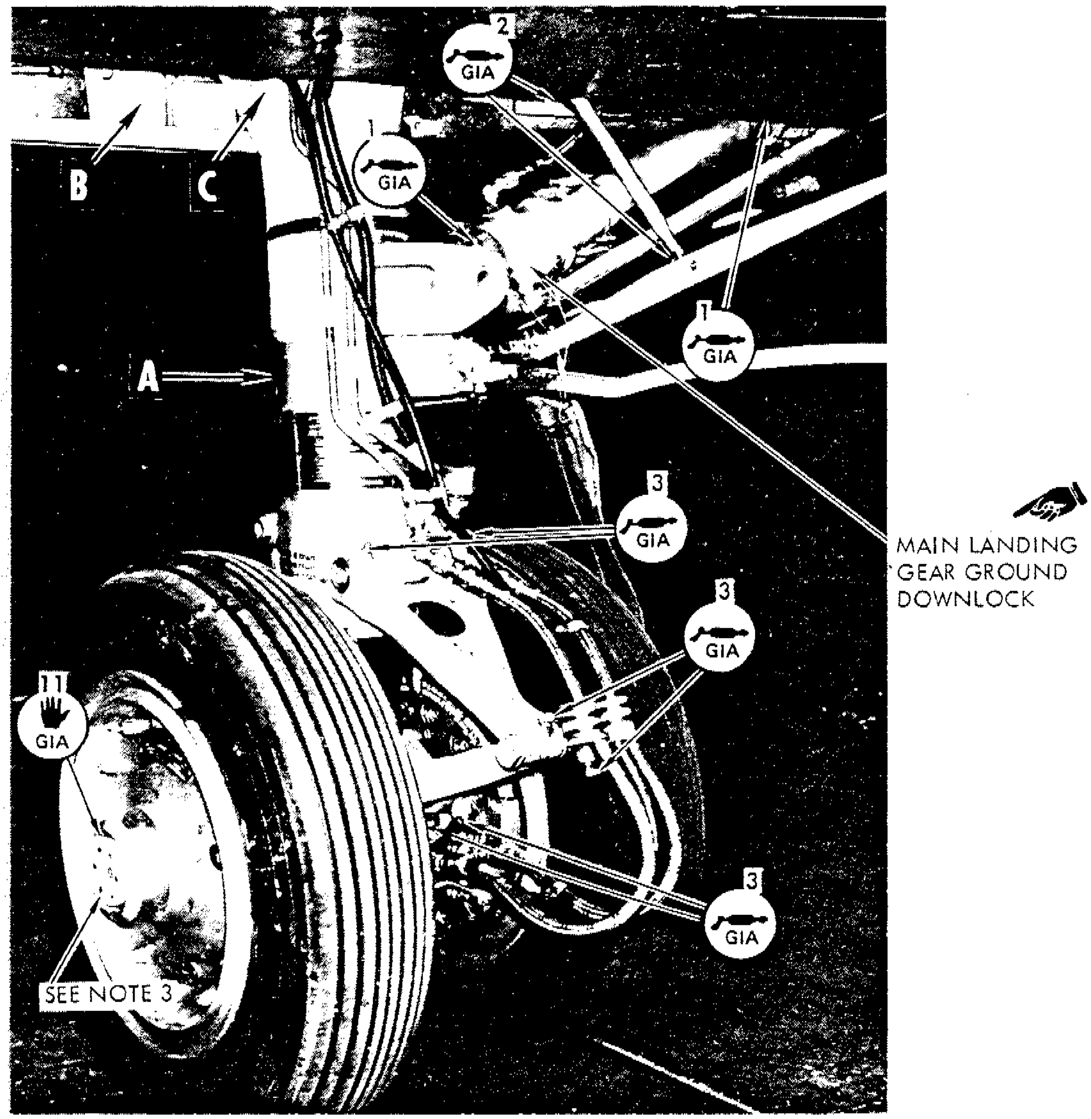
JS-2-2-1218-1

Figure 12-18. Lubrication of the Main Landing Gear Inboard Door and Release Mechanism (Sheet 1 of 2)



JS-1-2-X0/2-1218-2

Figure 12-18. Lubrication of the Main Landing Gear Inboard Door and Release Mechanism (Sheet 2 of 2)



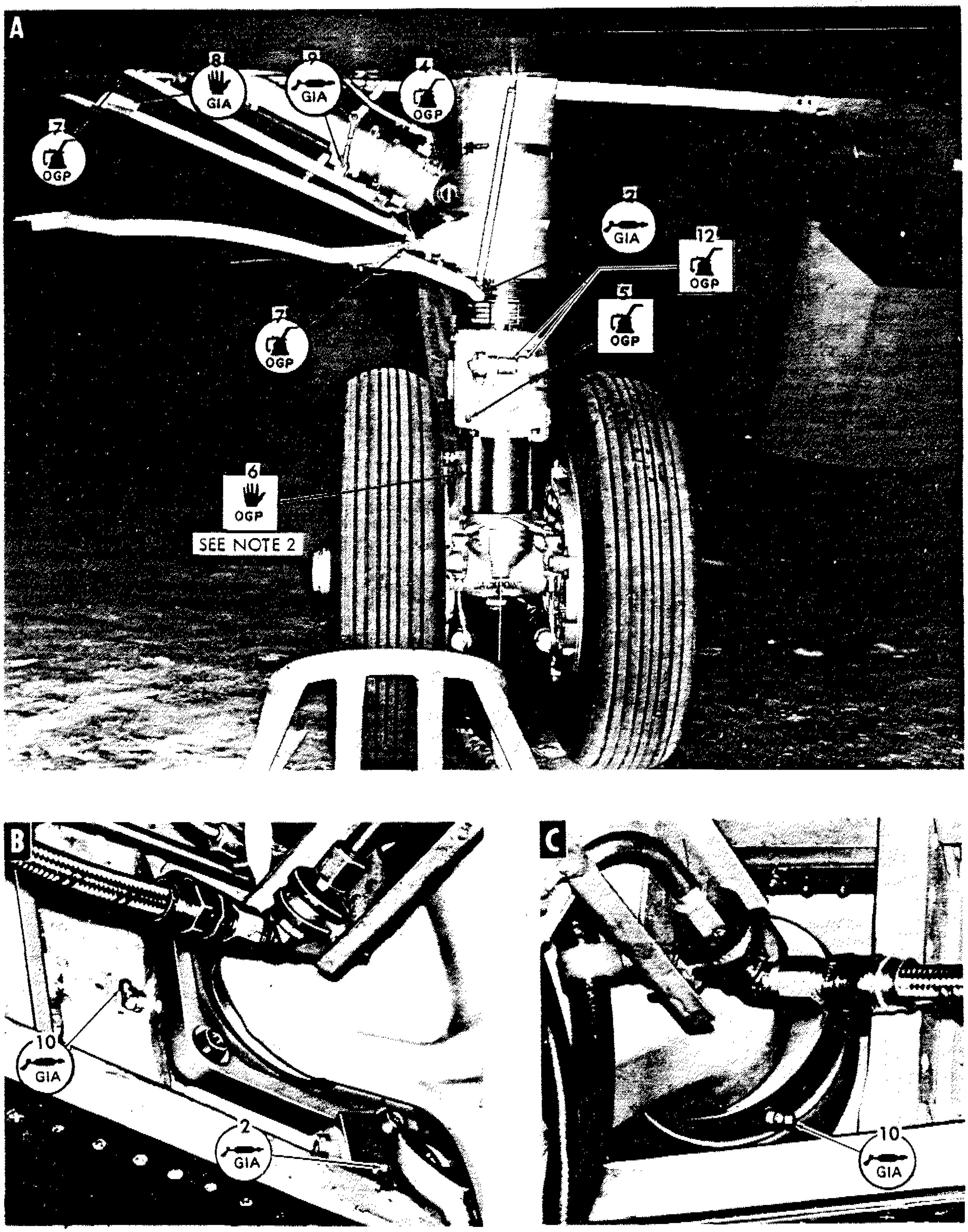
1. ACTUATOR ROD END (2 PLACES)
2. ROD, OUTBOARD WHEEL COVER (4 PLACES)
3. SHOCK STRUT TORQUE ARMS (6 PLACES)
4. ACTUATOR WIPER RING (REMOVE GROUND LOCK FOR ACCESS)
5. SHOCK STRUT PISTON WIPER
6. EXPOSED SHOCK STRUT PISTON
7. OUTBOARD DOOR INSTALLATION HINGES (2 PLACES)
8. LATCH BRACKET
9. ROD, ACTUATOR COVER (2 PLACES)
10. SHOCK STRUT TRUNNION (2 PLACES)
11. HUBCAP SPINNER
12. TOW RING BEARINGS AND SPRING

NOTE

1. RIGHT GEAR SHOWN, LEFT OPPOSITE.
2. BEFORE EACH FLIGHT, WIPE CLEAN OF DIRT AND GREASE AND APPLY LIGHT FILM OF MIL-L-7870 OIL.
3. AT EACH WHEEL CHANGE, CLEAN OUT ALL OLD GREASE AND REPACK WHEEL BEARINGS WITH WTR. CLEAN AND APPLY LIGHT COAT OF WTR TO AXLE ALSO.

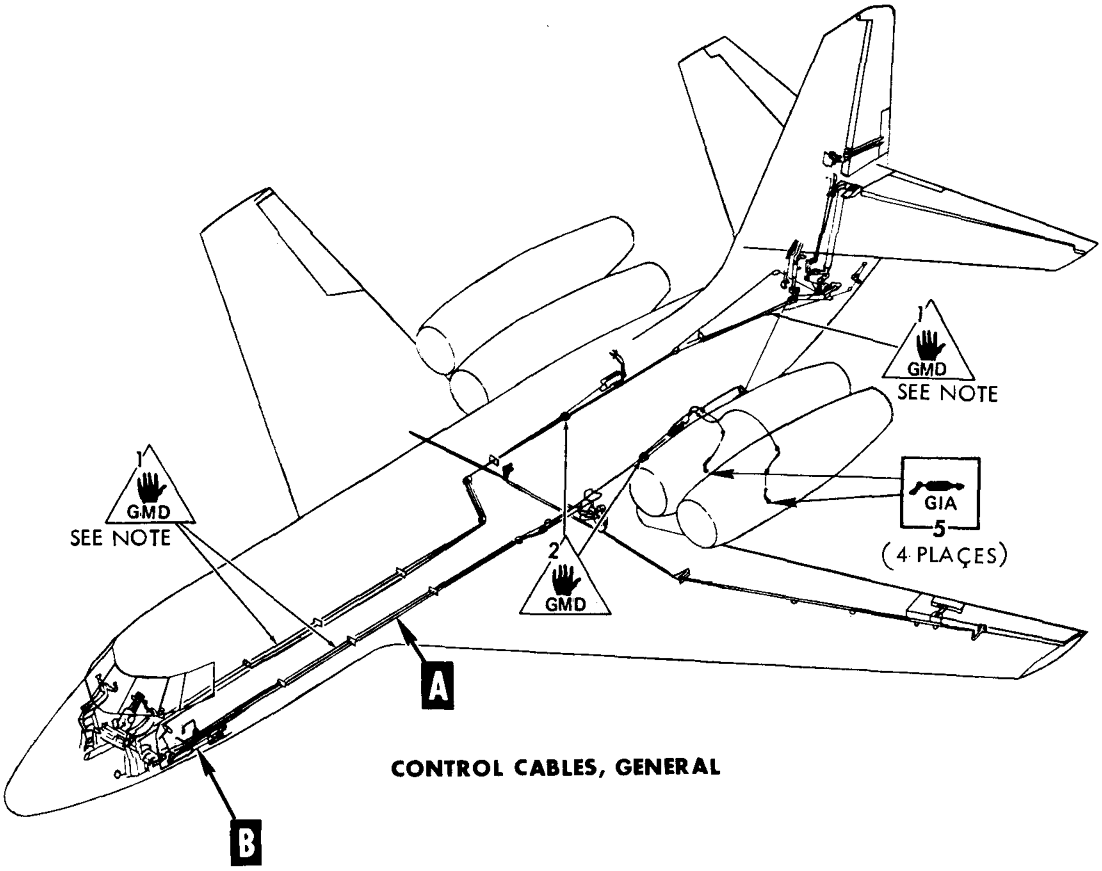
JS-1-2-X0/9-1219-1

Figure 12-19. Lubrication of the Main Landing Gear and Outboard Doors (Sheet 1 of 2)



JS-1-2-X0/2-1219-2

Figure 12-19. Lubrication of the Main Landing Gear and Outboard Doors (Sheet 2 of 2)



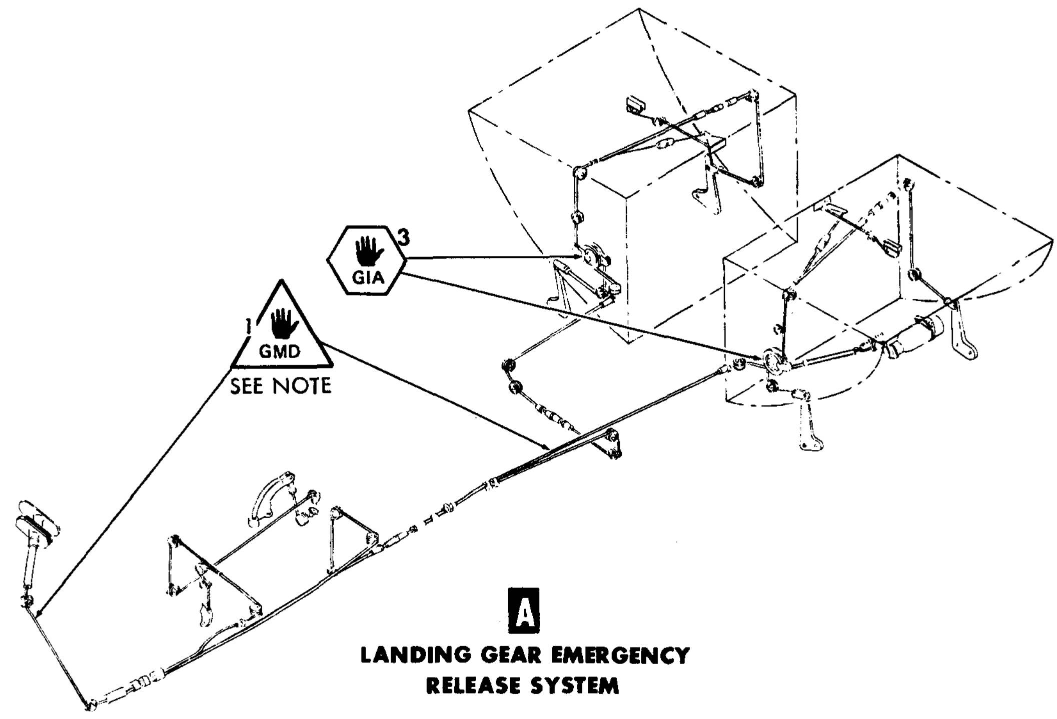
1. CABLES
2. PRESSURE SEALS
3. PULLEY ASSEMBLY, MAIN GEAR RELEASE
4. STEERING SPROCKET AND CHAIN
5. ROD END BEARING

NOTE

CLEAN OLD GREASE FROM CABLES USING CLEAN, DRY CLOTHS AND REAPPLY GREASE.

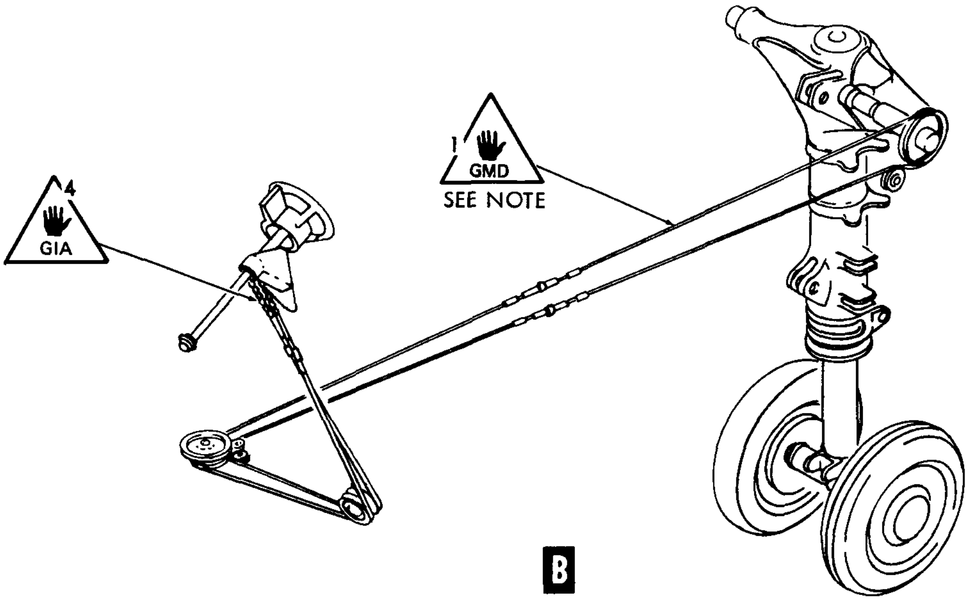
JS-1-2-X0/2-1220-1

Figure 12-20. Lubrication of the Control Cables (Sheet 1 of 2)



A

LANDING GEAR EMERGENCY RELEASE SYSTEM



B

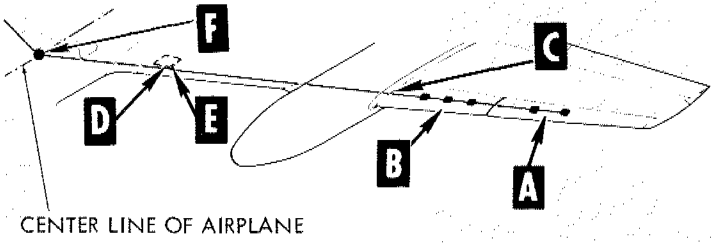
STEERING CONTROLS

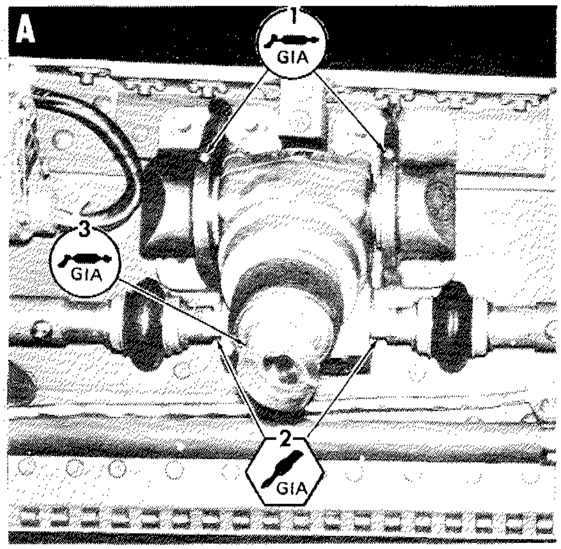
JS-1-2-X0/2-1220-2

Figure 12-20. Lubrication of the Control Cables (Sheet 2 of 2)

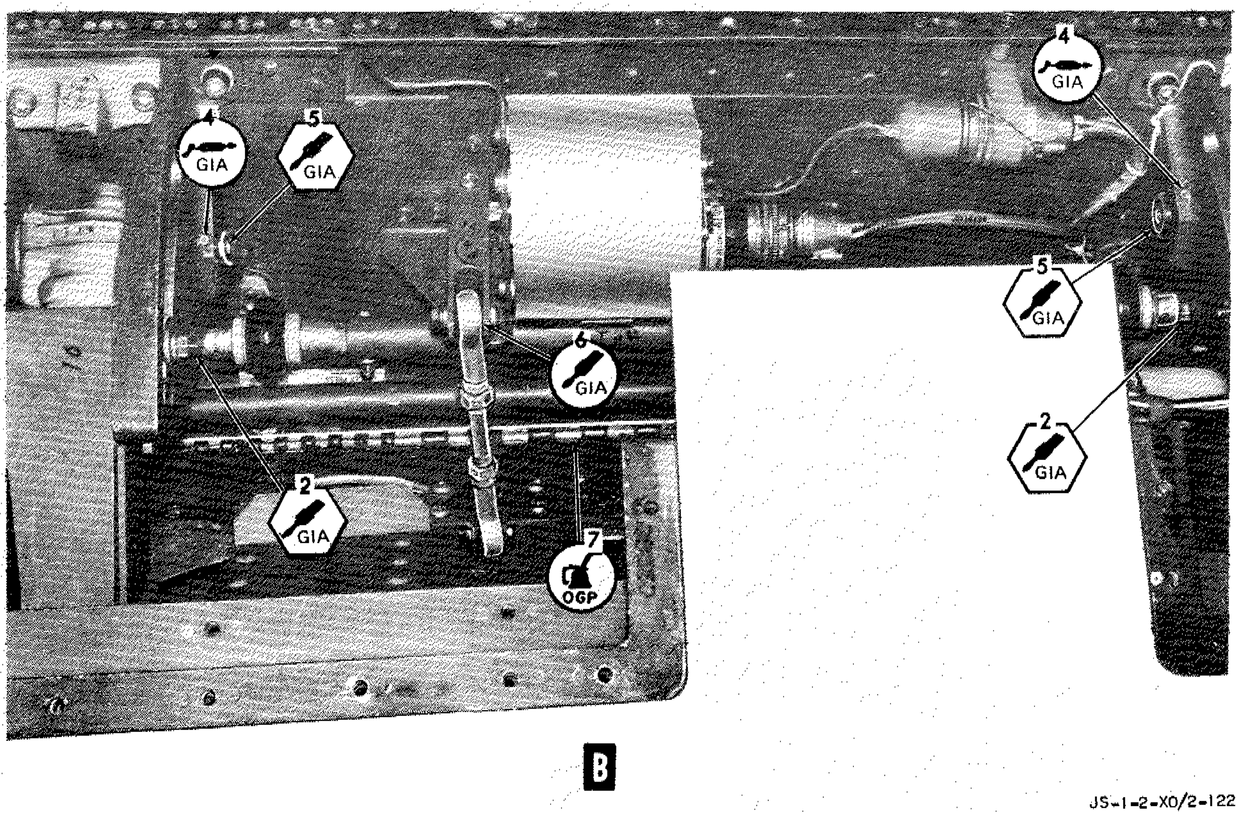
NOTE

1. LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE.
2. EVERY 400 HOURS, LUBRICATE THE MOTOR DRIVEN GEAR BOX. P/N 3201-7
   CALCO GEAR BOXES: P/N 202-
      LUBRICATE WITH GMD GREASE THROUGH THE ALEMITE 47200 BREATHER AND FILLER.
   VARO GEAR BOXES: P/N 965-001-6
      FILL WITH OGR OIL THROUGH THE OIL FILLER CAP ON THE GEAR BOX CASE TO THE OIL LEVEL HOLE. -7



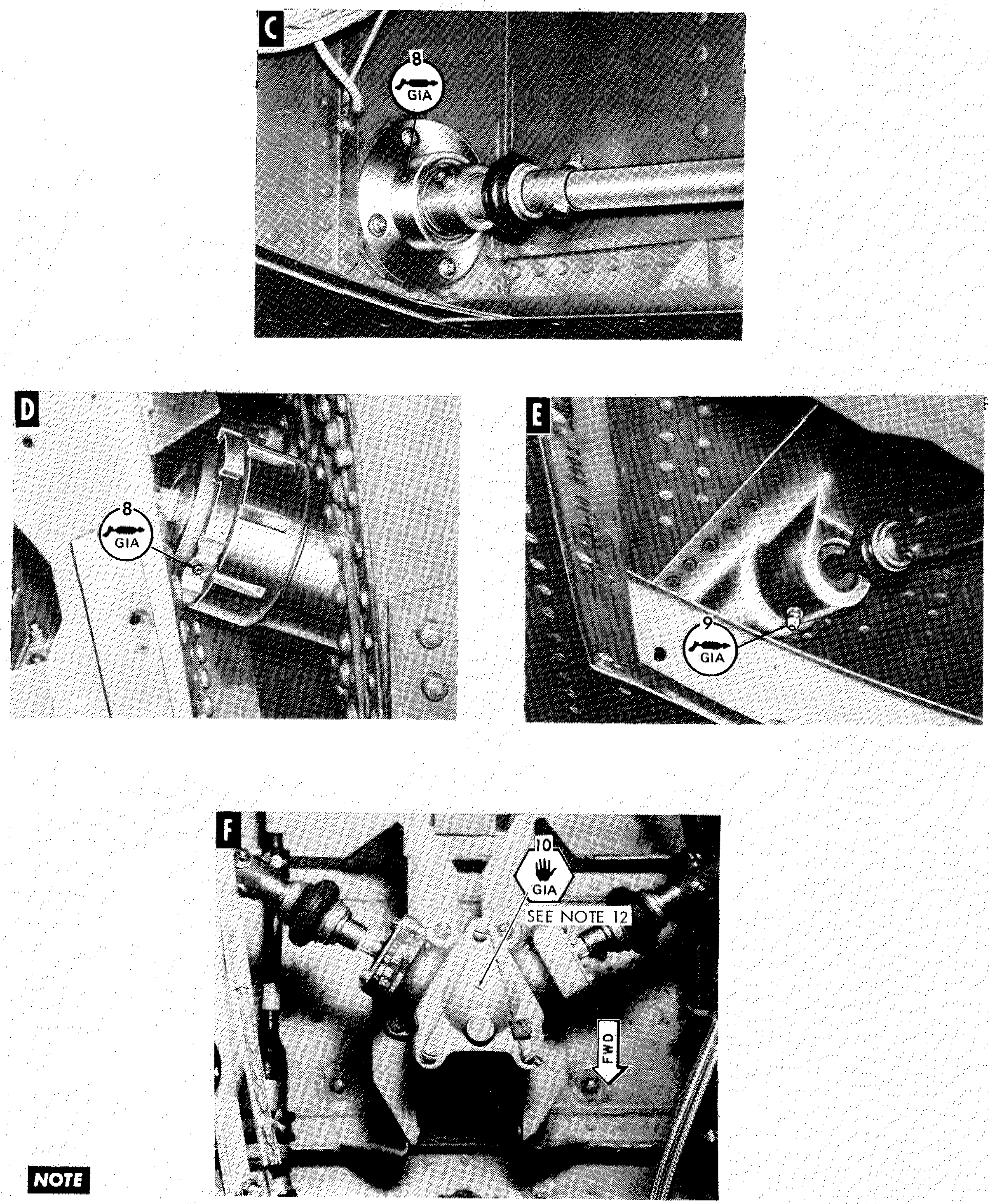


1. FITTING ASSEMBLY OUTBOARD (4 PLACES)
2. SPLINE COUPLINGS, DRIVE MECHANISM (10 PLACES)
3. ACTUATOR ROD ENDS (4 PLACES)
4. FITTING ASSEMBLY INBOARD (4 PLACES)
5. ACTUATOR TRUNNION ENDS (4 PLACES)
6. LINK ASSEMBLY FOLLOW-UP SWITCH (LEFT SIDE ONLY)
7. FLAP HINGE
8. BEARING HOUSING, DRIVE MECHANISM (2 PLACES)
9. PRESSURE FITTINGS, DRIVE MECHANISM (2 PLACES)
10. FLAP DRIVE 120° GEAR BOX
11. LUBRICATE L. E. FLAP ACTUATORS IN ACCORDANCE WITH SECTION 27.4.6.5 AND SECTION 27.4.6.6



JS-1-2-X0/2-1221-1

Figure 12-21. Lubrication of the Leading Edge Flap System (Sheet 1 of 2)

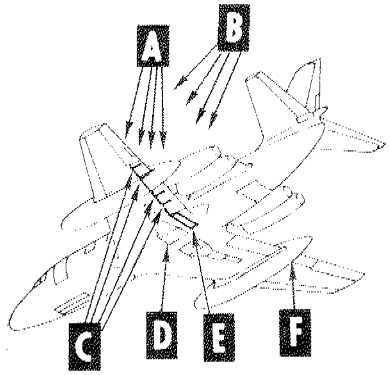


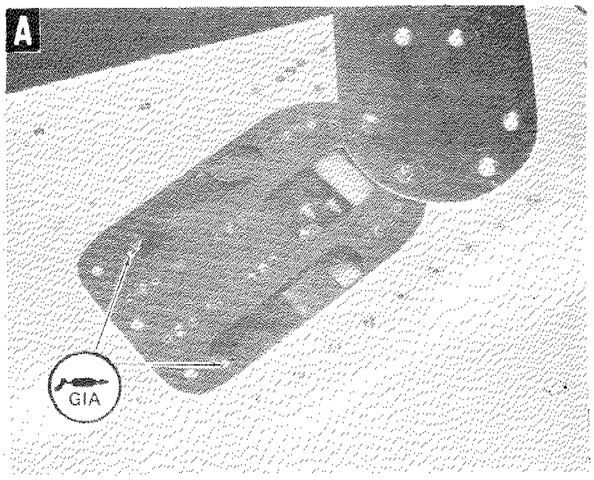
NOTE

12. REMOVE COVER FROM TOP OF GEAR BOX. REMOVE AS MUCH GREASE AS POSSIBLE. FILL HOUSING APPROX. HALF FULL WITH NEW GREASE. REPLACE COVER AND LOCKWIRE

JS-1-2-X0/2-1221-2

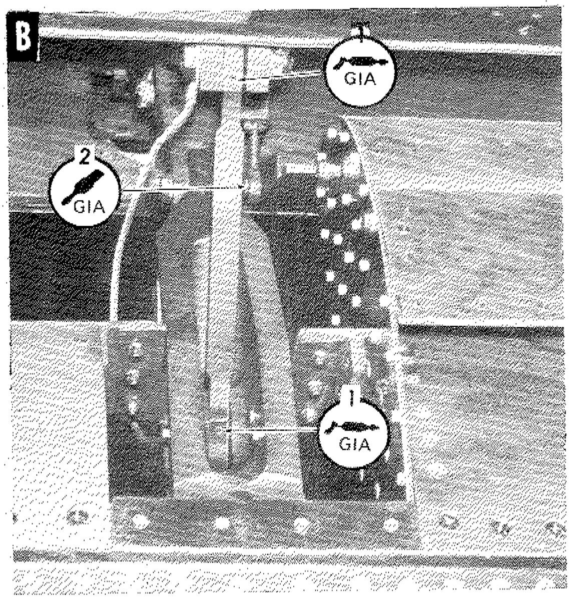
Figure 12-21. Lubrication of the Leading Edge Flap System (Sheet 2 of 2)

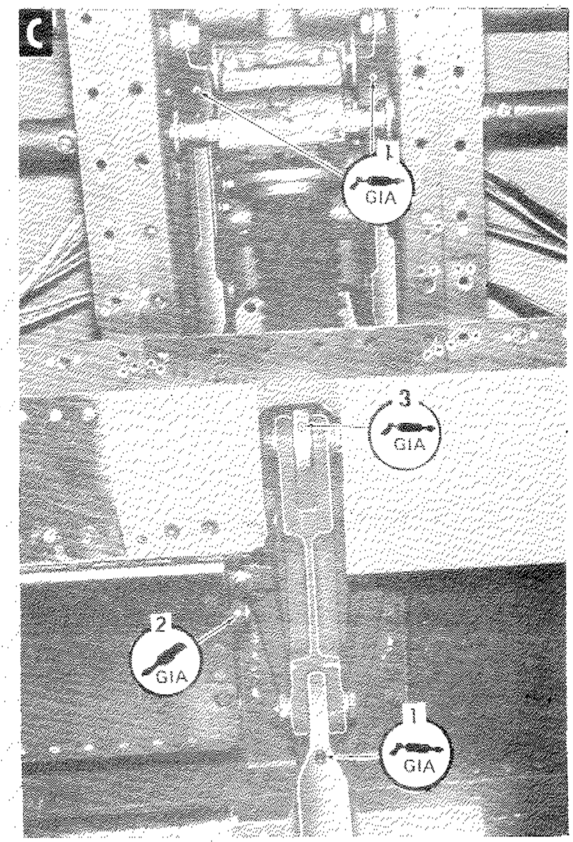




NOTE

EVERY 400 HOURS, LUBRICATE THE MOTOR DRIVEN GEAR BOX.
CALCO GEAR BOXES:
  LUBRICATE WITH GMD GREASE THROUGH THE ALEMITE 47200 BREATHER AND FILLER.
VARO GEAR BOXES:
  FILL WITH OGR OIL THROUGH THE OIL FILLER CAP ON THE GEAR BOX CASE TO THE OIL LEVEL HOLE.

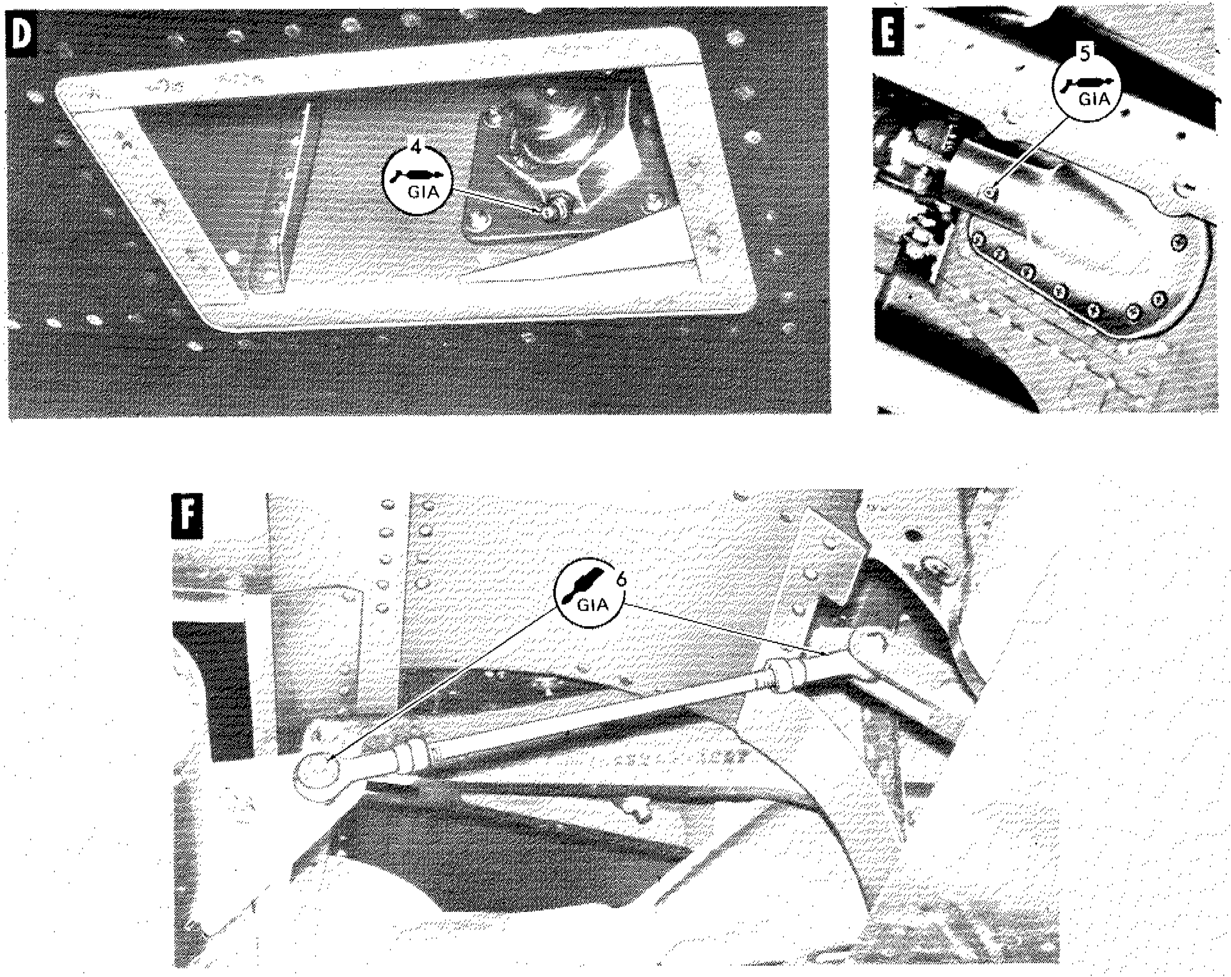




1. ACTUATOR FITTINGS AND LINKAGE MECHANISM (7 PLACES EACH UNIT)
2. FLAP DOOR LINKAGE (4 PLACES)
3. FLAP ACTUATOR ROD END
4. FITTING
5. PRESSURE FITTING
6. LINKAGE, FOLLOW-UP SWITCH (2 PLACES)
7. LUBRICATE T. E. FLAP ACTUATORS IN ACCORDANCE WITH SECTION 27.4.6.5 AND SECTION 27.4.6.6

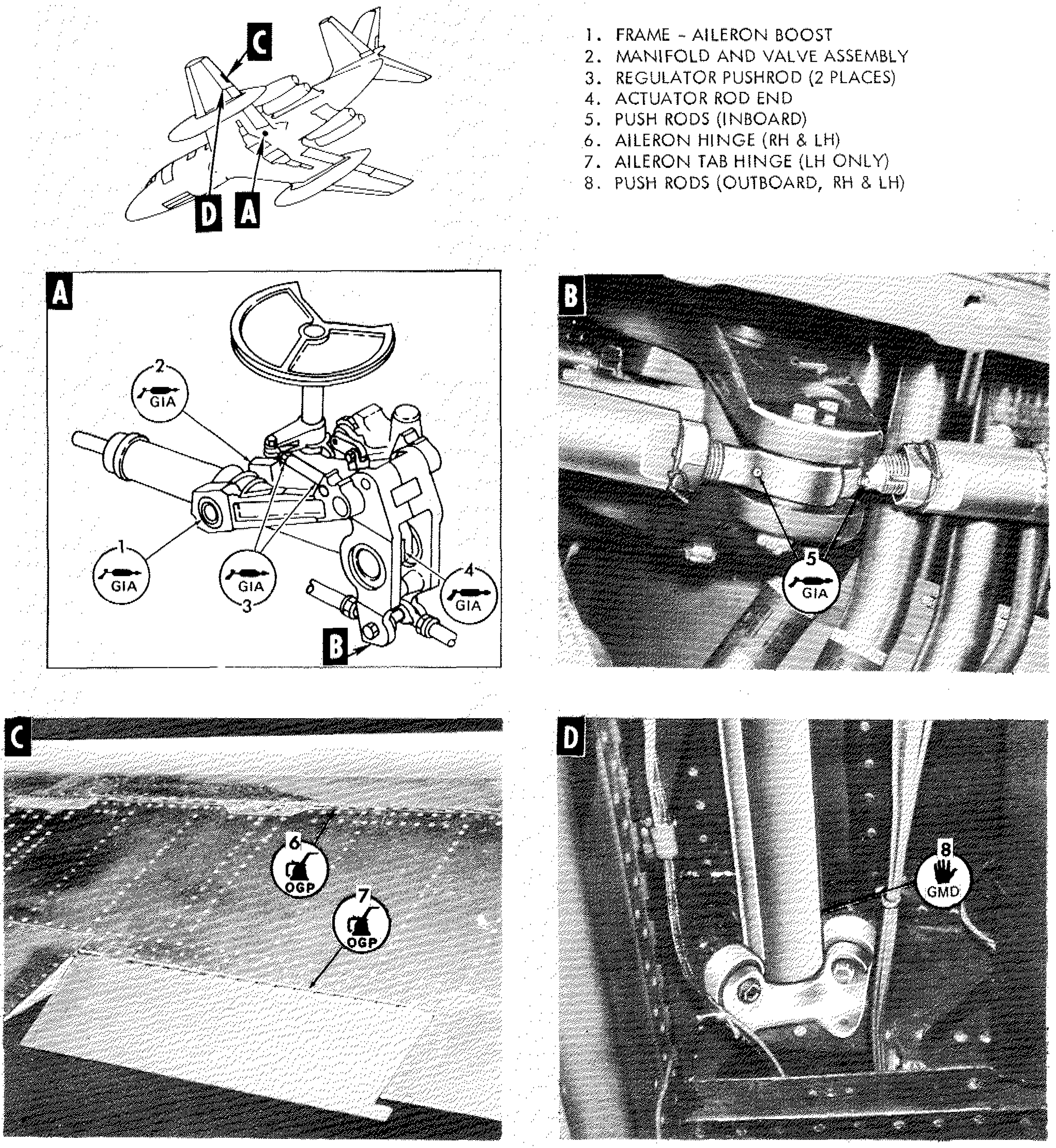
JS-1-2-X0/2-1222-1

Figure 12-22. Lubrication of the Trailing Edge Flap System (Sheet 1 of 2)



JS-1-2-X0/2-1222-2

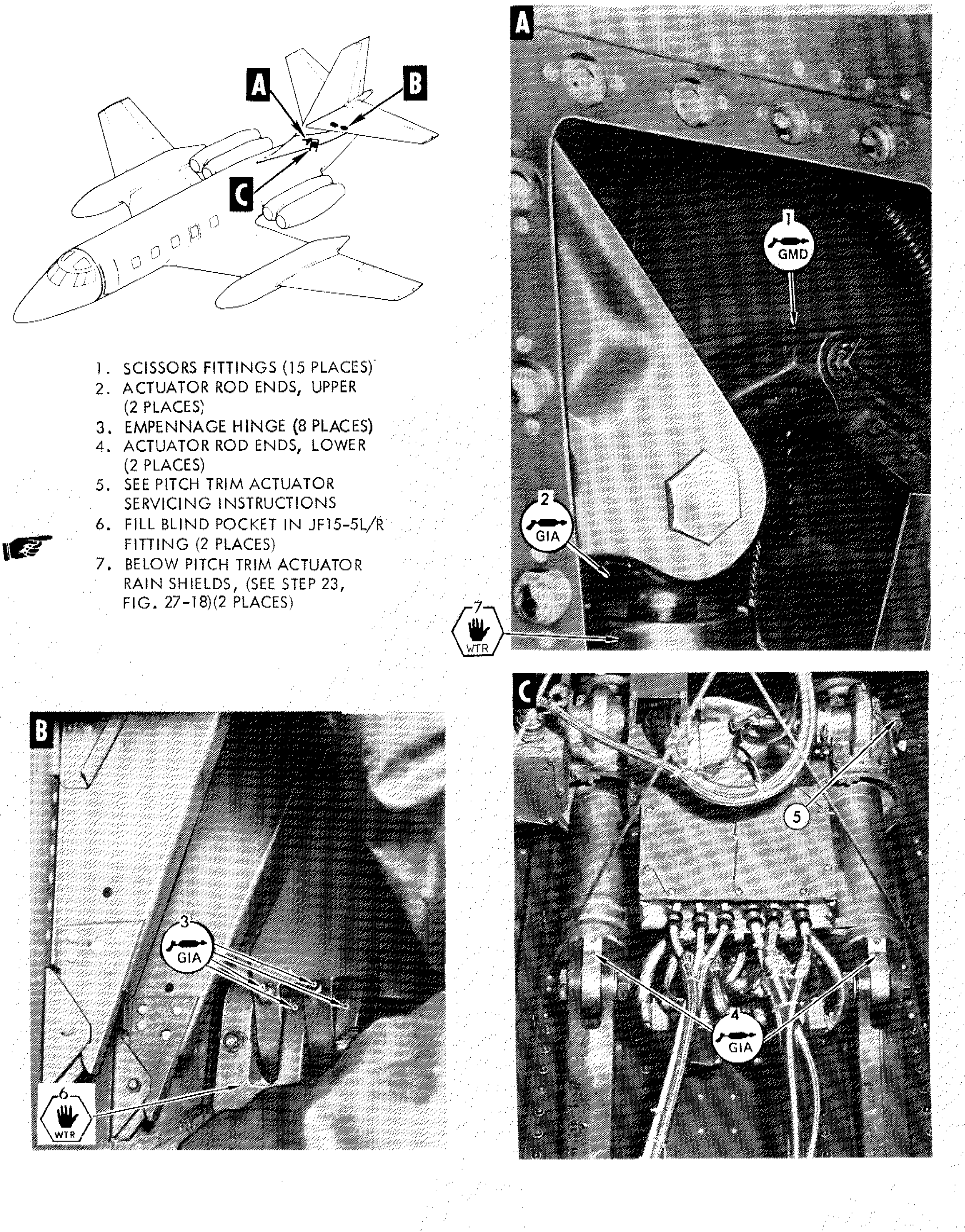
Figure 12-22. Lubrication of the Trailing Edge Flap System (Sheet 2 of 2)



1. FRAME - AILERON BOOST
2. MANIFOLD AND VALVE ASSEMBLY
3. REGULATOR PUSHROD (2 PLACES)
4. ACTUATOR ROD END
5. PUSH RODS (INBOARD)
6. AILERON HINGE (RH & LH)
7. AILERON TAB HINGE (LH ONLY)
8. PUSH RODS (OUTBOARD, RH & LH)

JS-1-2-X0/2-1223

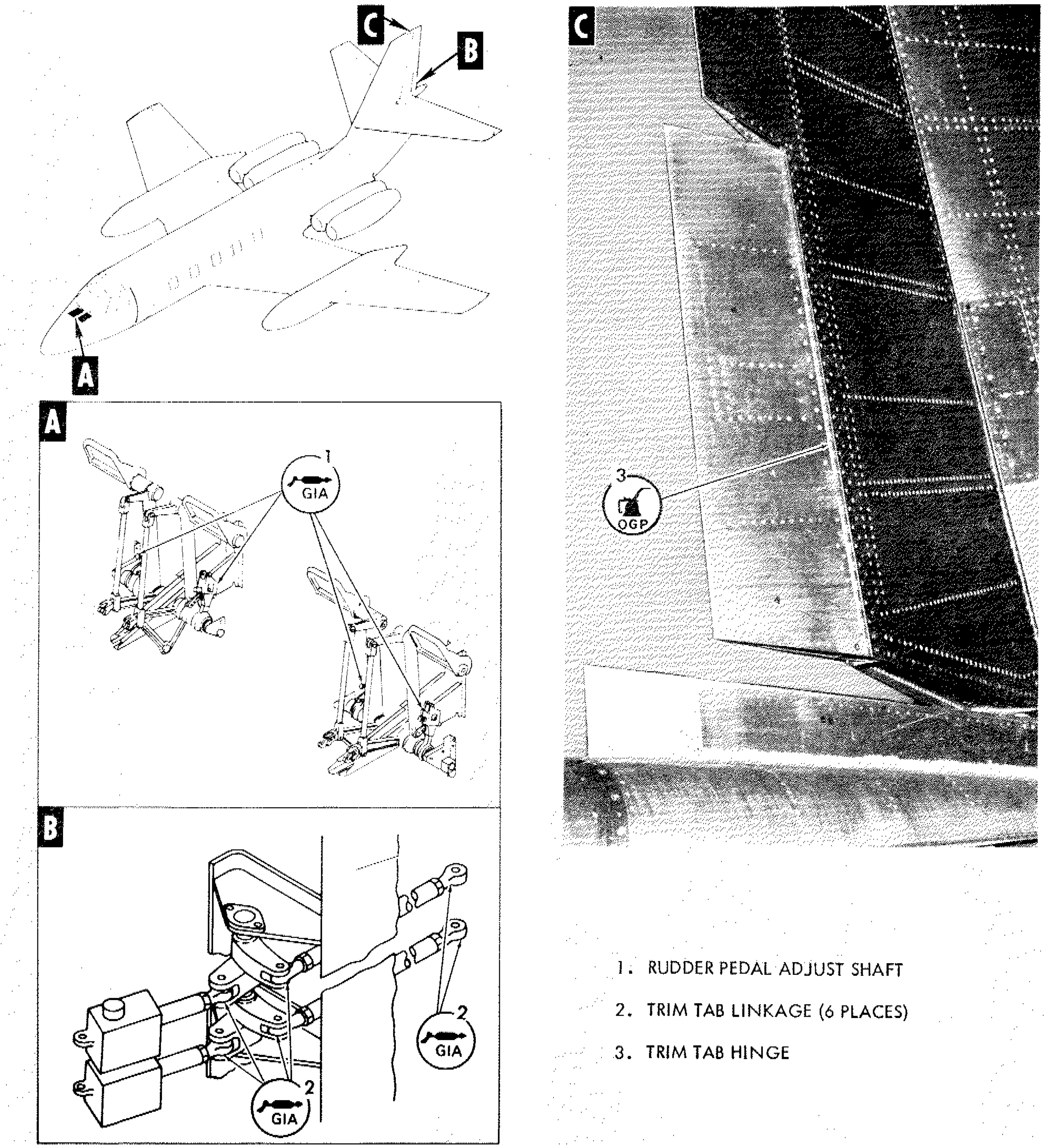
Figure 12-23. Lubrication of the Aileron and Aileron Tab Systems



1. SCISSORS FITTINGS (15 PLACES)
2. ACTUATOR ROD ENDS, UPPER (2 PLACES)
3. EMPENNAGE HINGE (8 PLACES)
4. ACTUATOR ROD ENDS, LOWER (2 PLACES)
5. SEE PITCH TRIM ACTUATOR SERVICING INSTRUCTIONS
6. FILL BLIND POCKET IN JF15-5L/R FITTING (2 PLACES)
7. BELOW PITCH TRIM ACTUATOR RAIN SHIELDS, (SEE STEP 23, FIG. 27-18)(2 PLACES)

JS-1-2-X0/8-1224

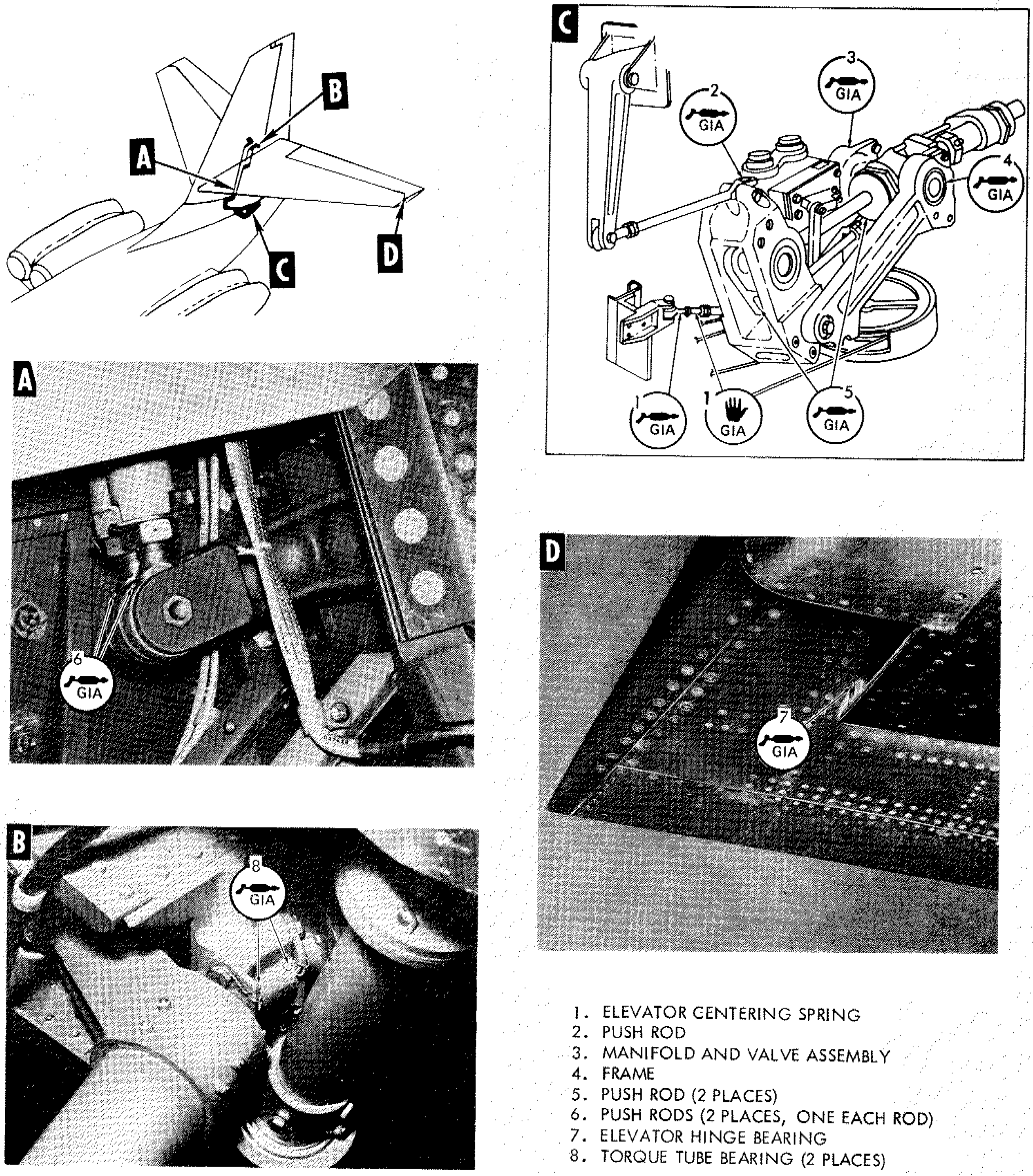
Figure 12-24. Lubrication of the Empennage



1. RUDDER PEDAL ADJUST SHAFT
2. TRIM TAB LINKAGE (6 PLACES)
3. TRIM TAB HINGE

JS-1-2-X0/2-1225

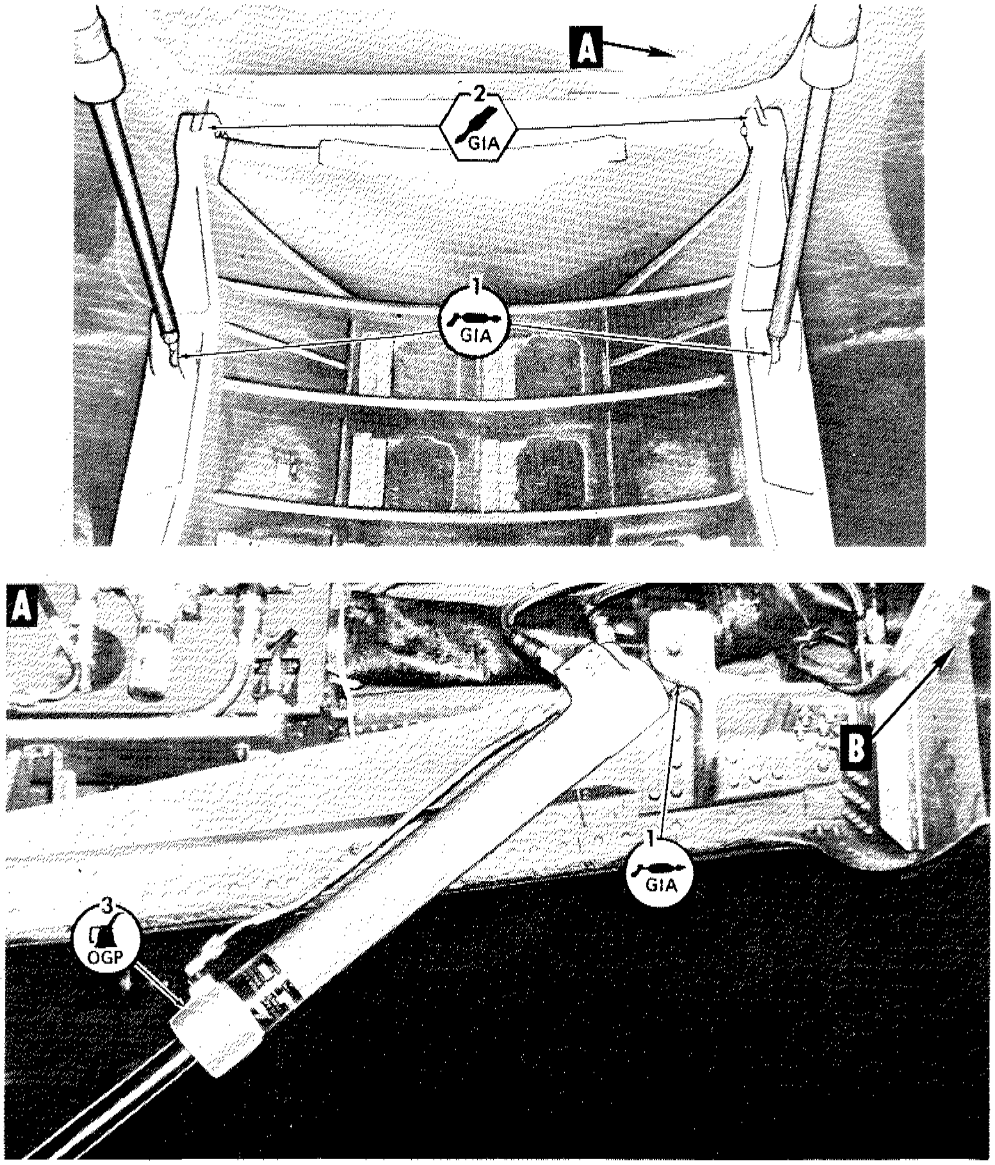
Figure 12-25. Lubrication of the Rudder and Rudder Trim Tab Systems



1. ELEVATOR CENTERING SPRING
2. PUSH ROD
3. MANIFOLD AND VALVE ASSEMBLY
4. FRAME
5. PUSH ROD (2 PLACES)
6. PUSH RODS (2 PLACES, ONE EACH ROD)
7. ELEVATOR HINGE BEARING
8. TORQUE TUBE BEARING (2 PLACES)

JS-1-2-X0/2-1226

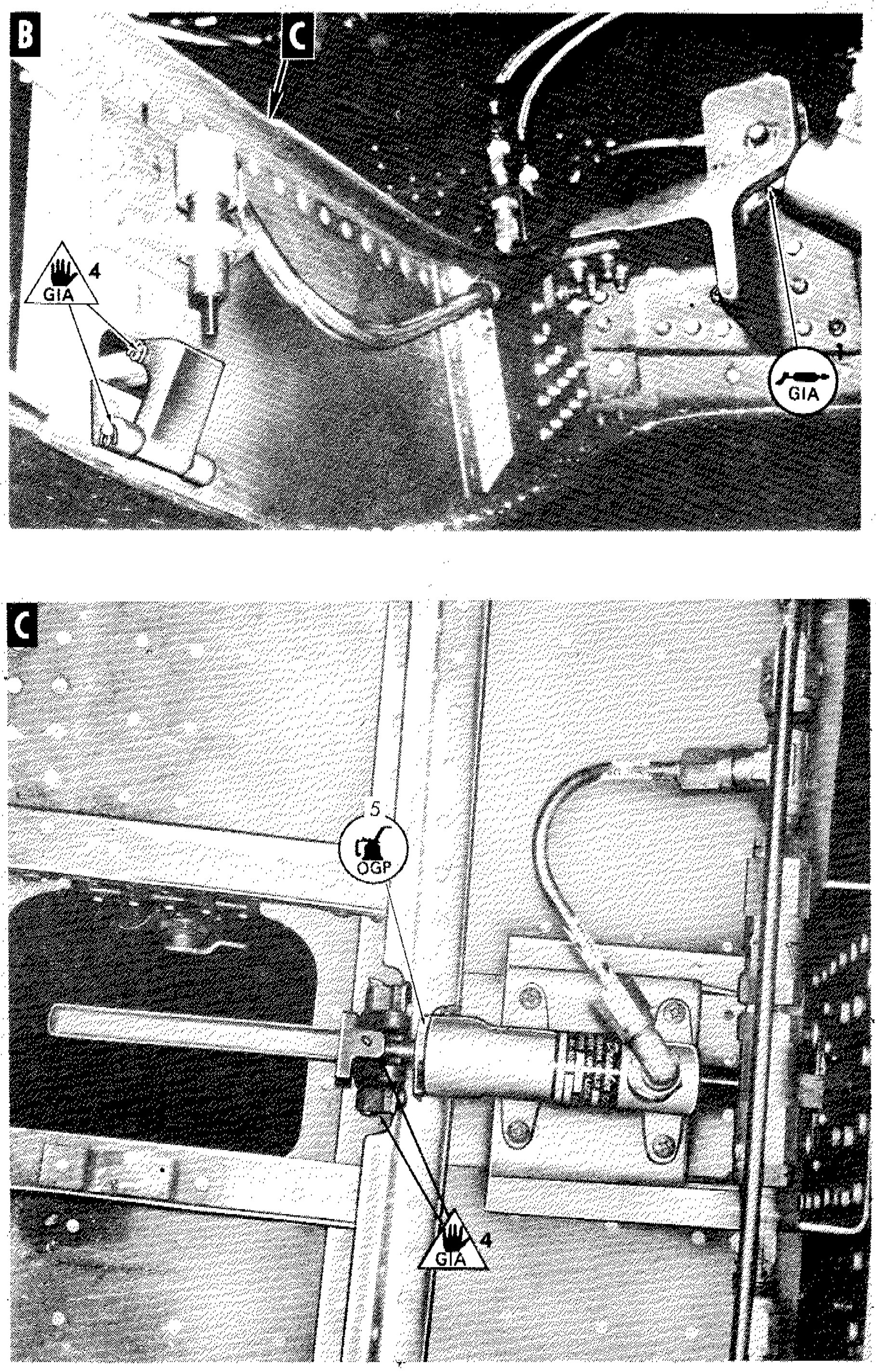
Figure 12-26. Lubrication of the Elevator



1. ACTUATOR ATTACH POINTS (4 PLACES)
2. DOOR HINGE FITTINGS (2 PLACES)
3. FELT WIPER, ACTUATOR (2 PLACES)
4. PINS, UPLOCK ASSEMBLY (4 PLACES)
5. FELT WIPER, UNLOCK CYLINDER (1 PLACE)

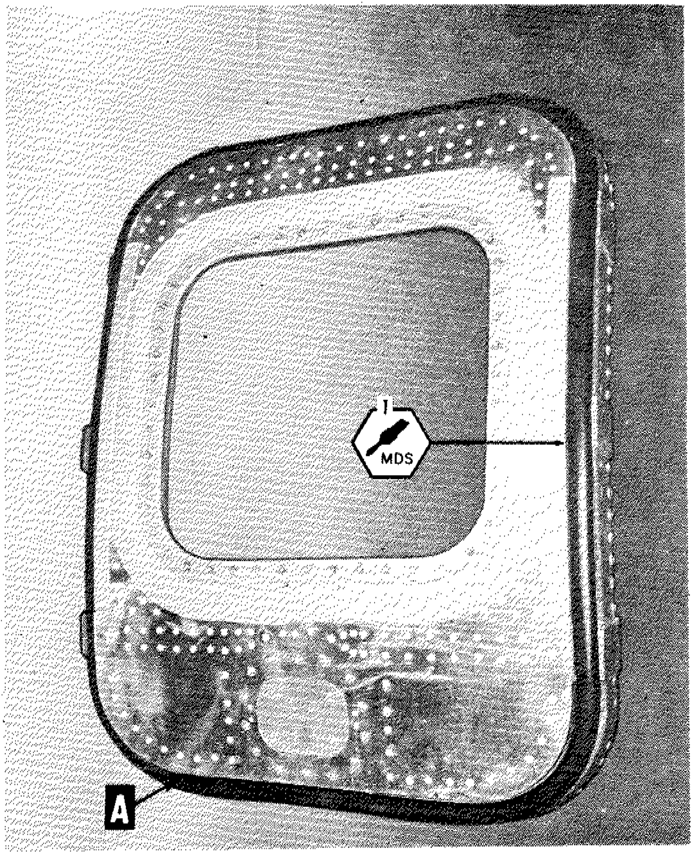
JS-1-2-X0/2-1227-1

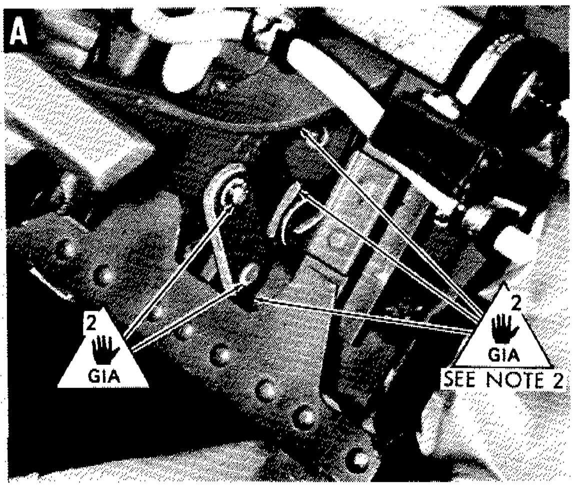
Figure 12-27. Lubrication of the Speed Brake (Sheet 1 of 2)



JS-1-2-X0/2-1227-2

Figure 12-27. Lubrication of the Speed Brake (Sheet 2 of 2)





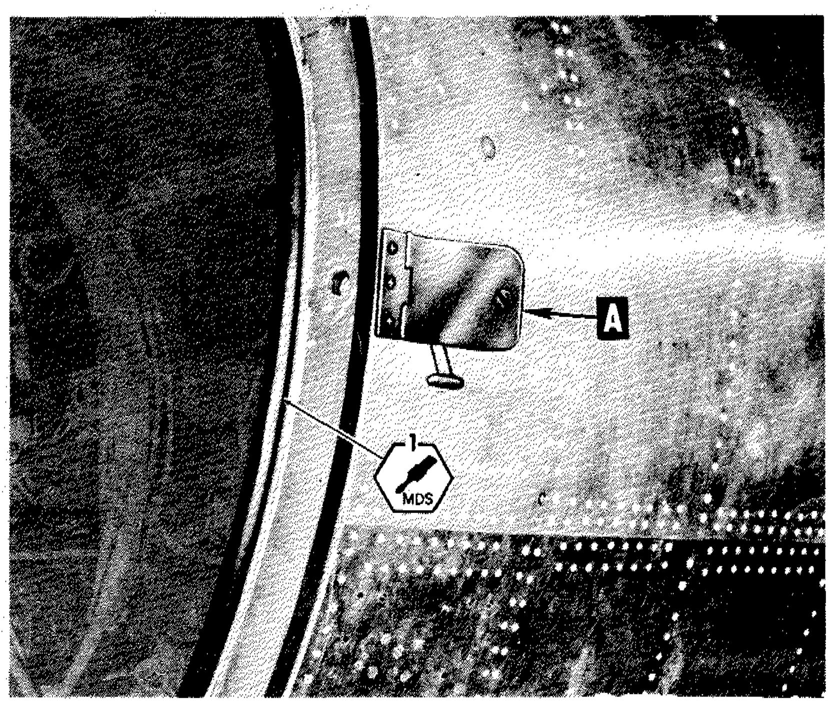
1. SEAL
2. LATCH MECHANISM PINS (5 PLACES)

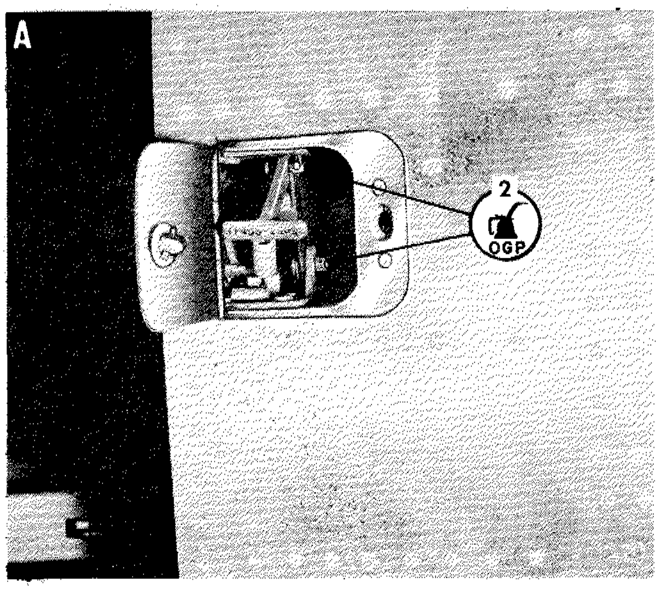
NOTE

1. EMERGENCY EXIT TYPICAL BOTH SIDES OF AIRPLANE.
2. LUBRICATE PIVOT POINTS NOT VISIBLE IN PHOTOGRAPH.

JS-1-2-X0/2-1228

Figure 12-28. Lubrication of the Emergency Exit

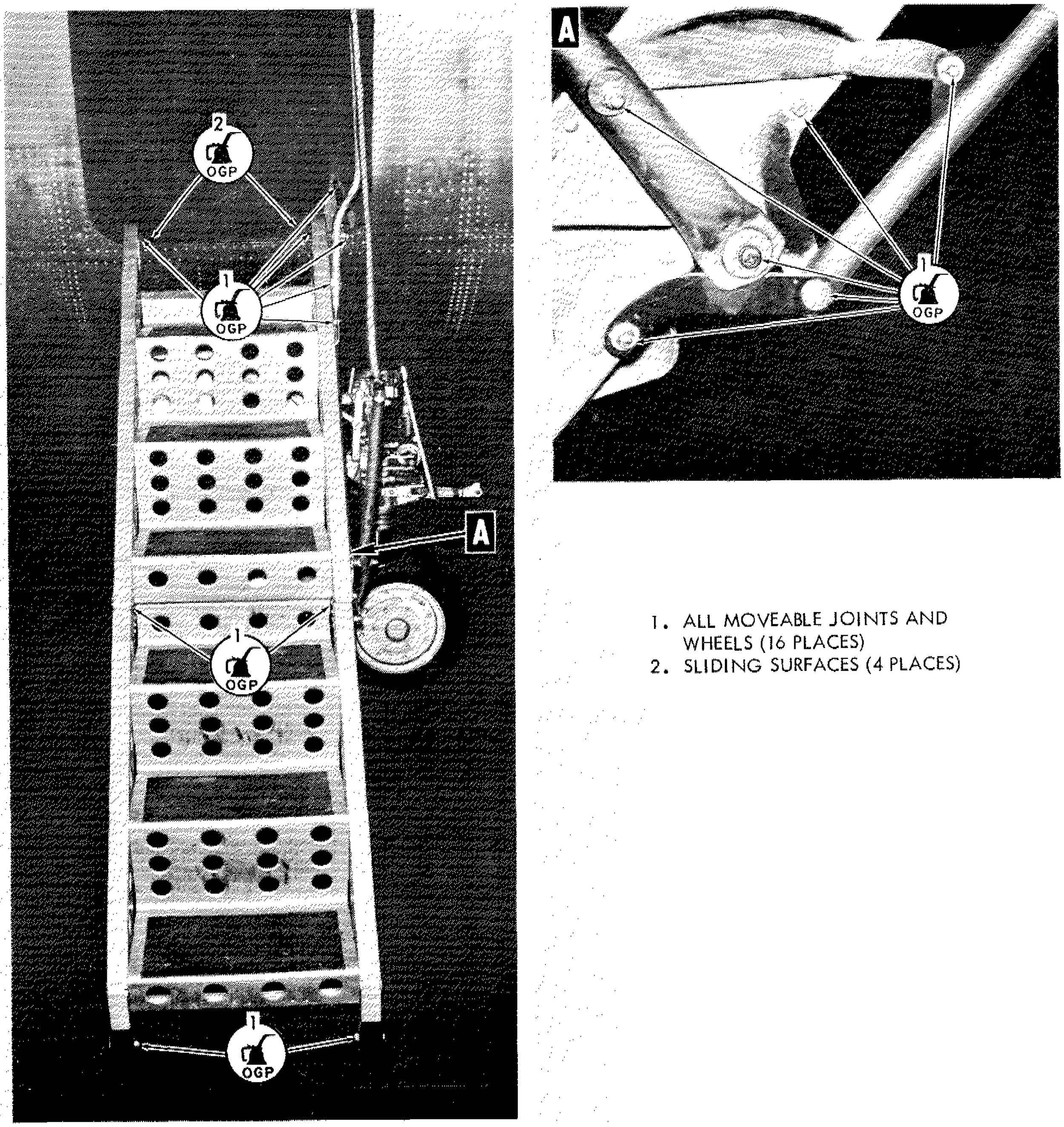




1. RUBBER SEAL (ON PRESSURENOSE ASSEMBLY)
2. LATCH MECHANISM (2 PLACES EACH LATCH)

JS-1-2-X0/2-1229

Figure 12-29. Lubrication of the Removable Nose Cone



1. ALL MOVEABLE JOINTS AND WHEELS (16 PLACES)
2. SLIDING SURFACES (4 PLACES)

Figure 12-30. Lubrication of the Airborne Ladder (Airplanes 5001 thru 5065)

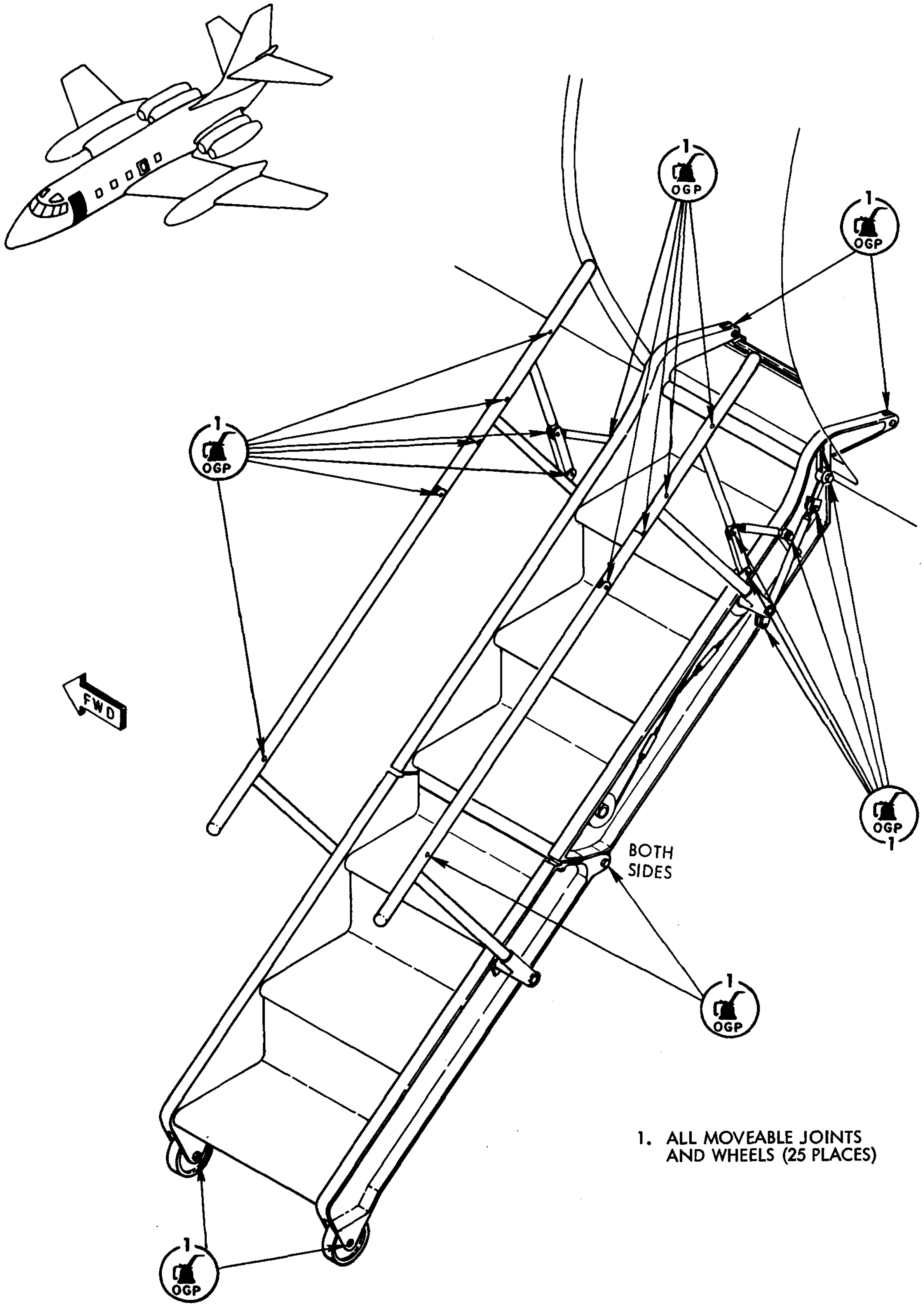
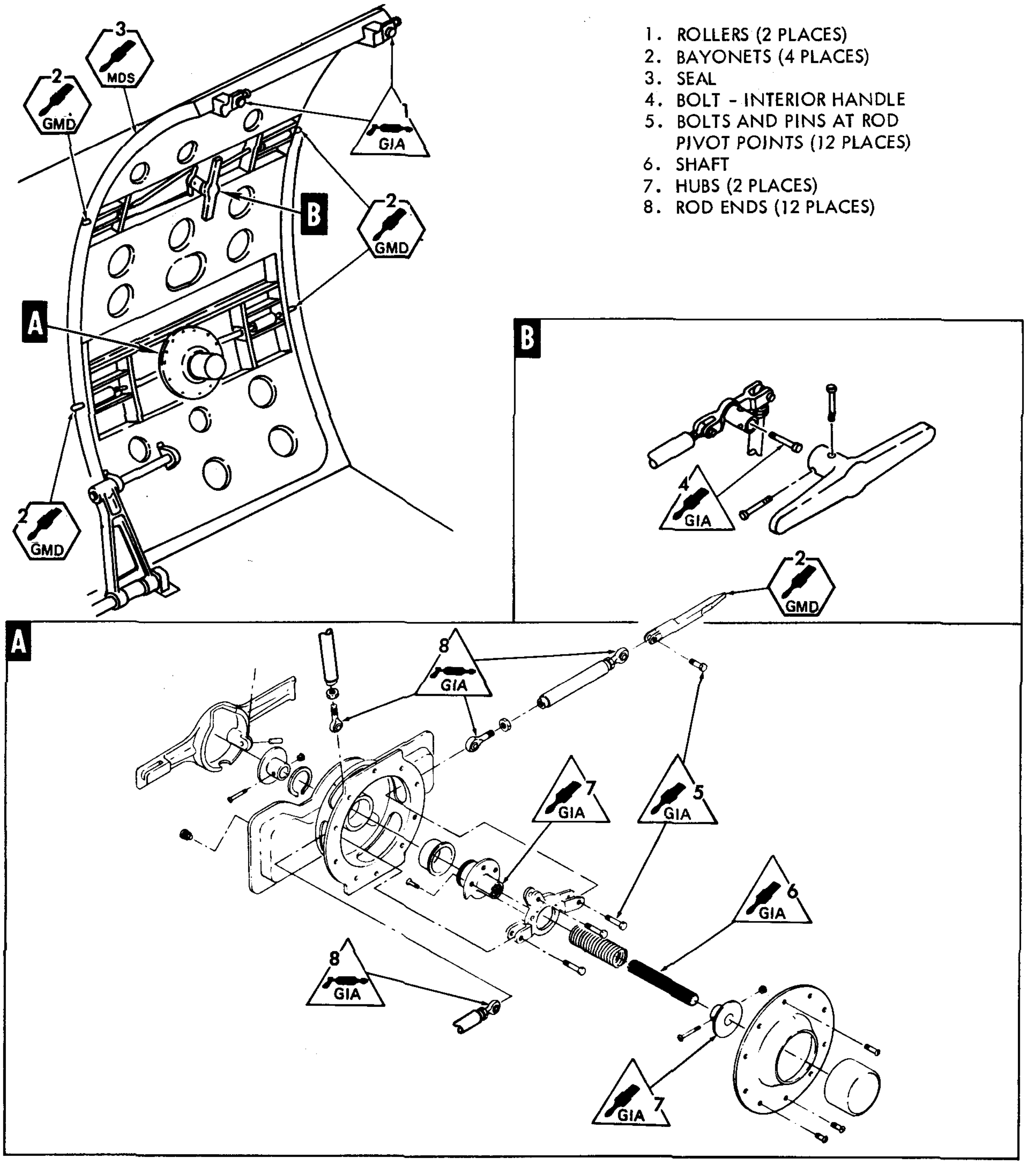


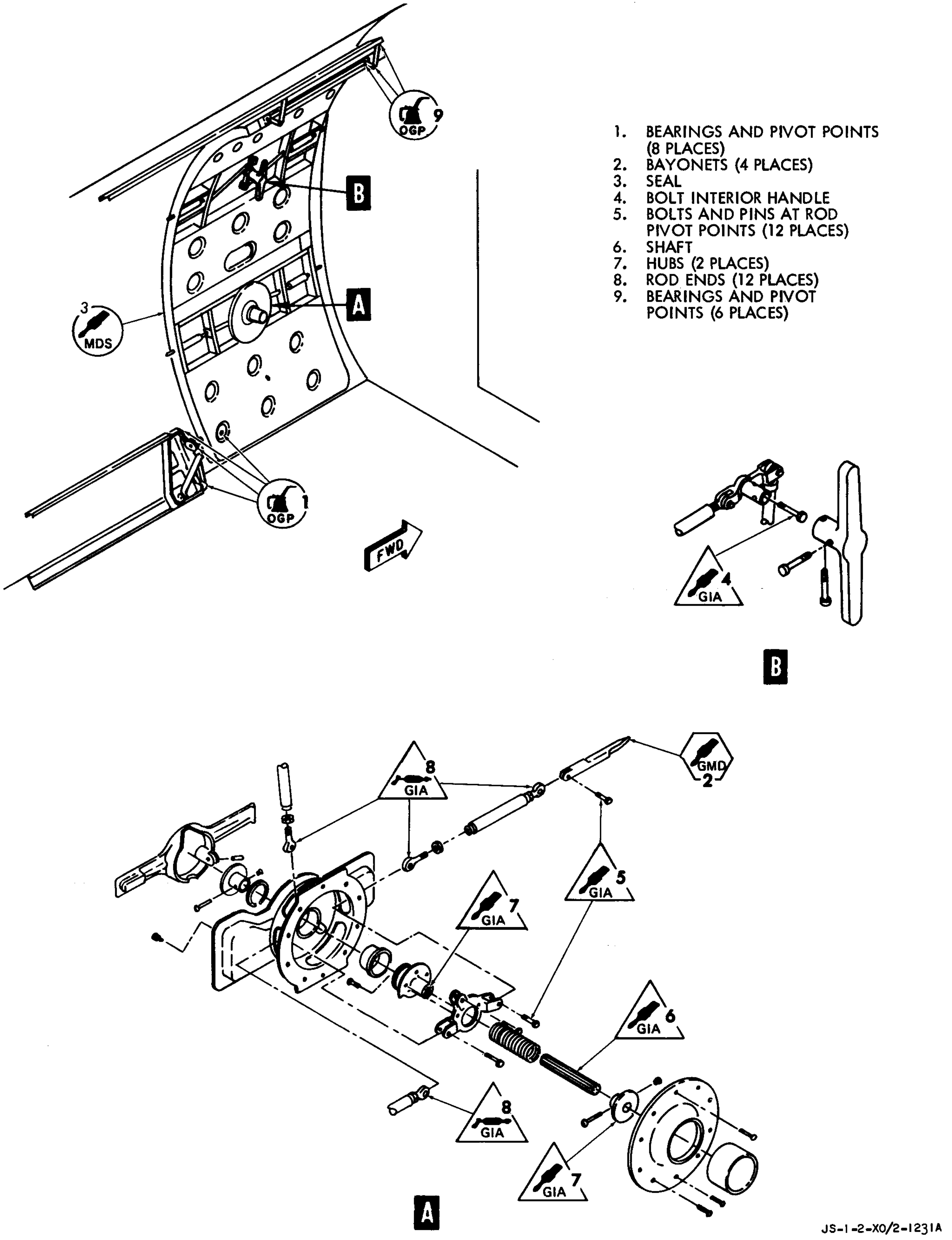
Figure 12-30A. Lubrication of the Automatic Stair Assembly (Airplanes 5066 and Up)



1. ROLLERS (2 PLACES)
2. BAYONETS (4 PLACES)
3. SEAL
4. BOLT - INTERIOR HANDLE
5. BOLTS AND PINS AT ROD PIVOT POINTS (12 PLACES)
6. SHAFT
7. HUBS (2 PLACES)
8. ROD ENDS (12 PLACES)

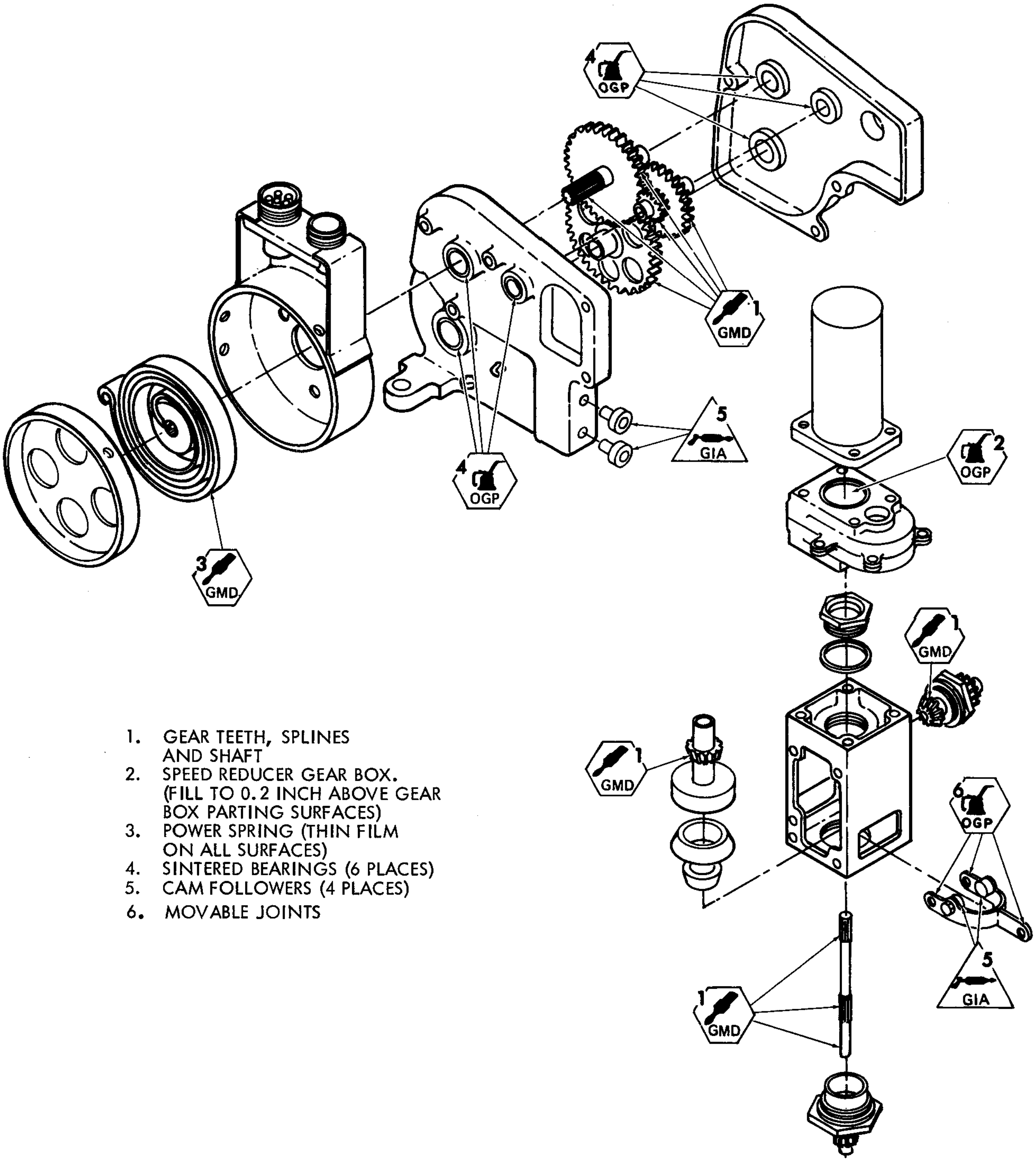
JS-1-2-X0/2-1231

Figure 12-31. Lubrication of the Main Entrance Door (Airplanes 5001 thru 5065)



1. BEARINGS AND PIVOT POINTS (8 PLACES)
2. BAYONETS (4 PLACES)
3. SEAL
4. BOLT INTERIOR HANDLE
5. BOLTS AND PINS AT ROD PIVOT POINTS (12 PLACES)
6. SHAFT
7. HUBS (2 PLACES)
8. ROD ENDS (12 PLACES)
9. BEARINGS AND PIVOT POINTS (6 PLACES)

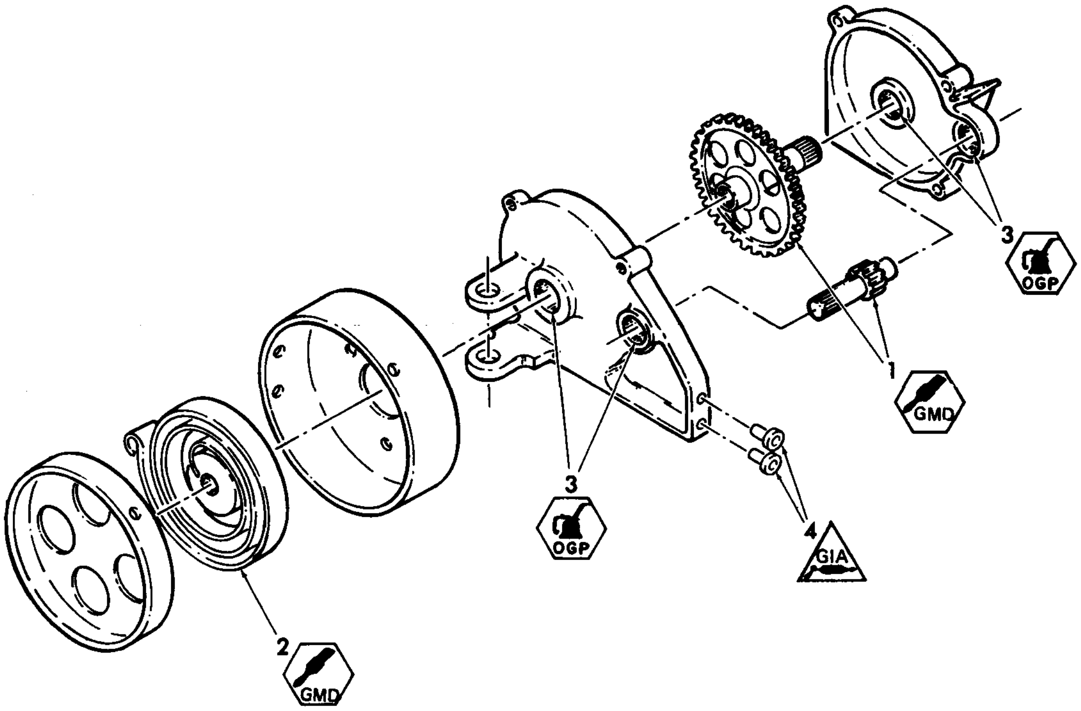
Figure 12-31A. Lubrication of the Entrance Door (Airplanes 5066 and Up)



1. GEAR TEETH, SPLINES AND SHAFT
2. SPEED REDUCER GEAR BOX. (FILL TO 0.2 INCH ABOVE GEAR BOX PARTING SURFACES)
3. POWER SPRING (THIN FILM ON ALL SURFACES)
4. SINTERED BEARINGS (6 PLACES)
5. CAM FOLLOWERS (4 PLACES)
6. MOVABLE JOINTS

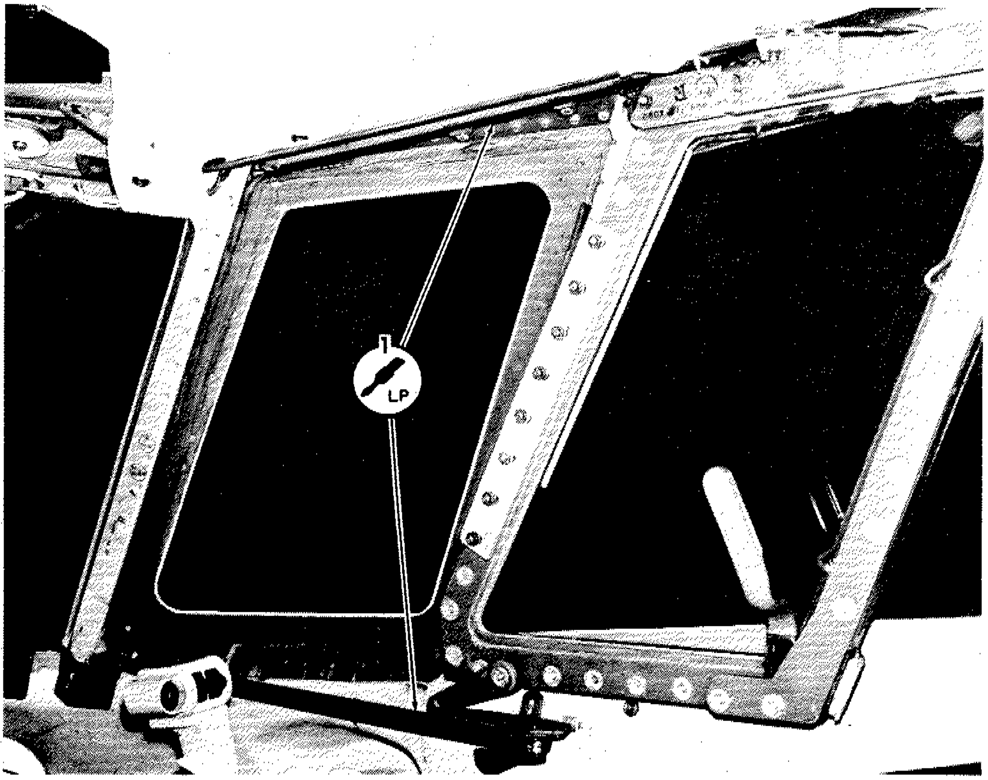
Figure 12-31B. Lubrication of the Automatic Stair Gear Box Assembly (Airplanes 5066 thru 5120)

1. GEAR TEETH, SPLINES AND SHAFT
2. POWER SPRING (THIN FILM ON ALL SURFACES)
3. SINTERED BEARINGS (6 PLACES)
4. CAM FOLLOWERS (4 PLACES)



JS-1-2-X0/2-1231C

Figure 12-31C. Lubrication of the Manual Stair Gearbox Assembly (Airplanes 5121 and Up)



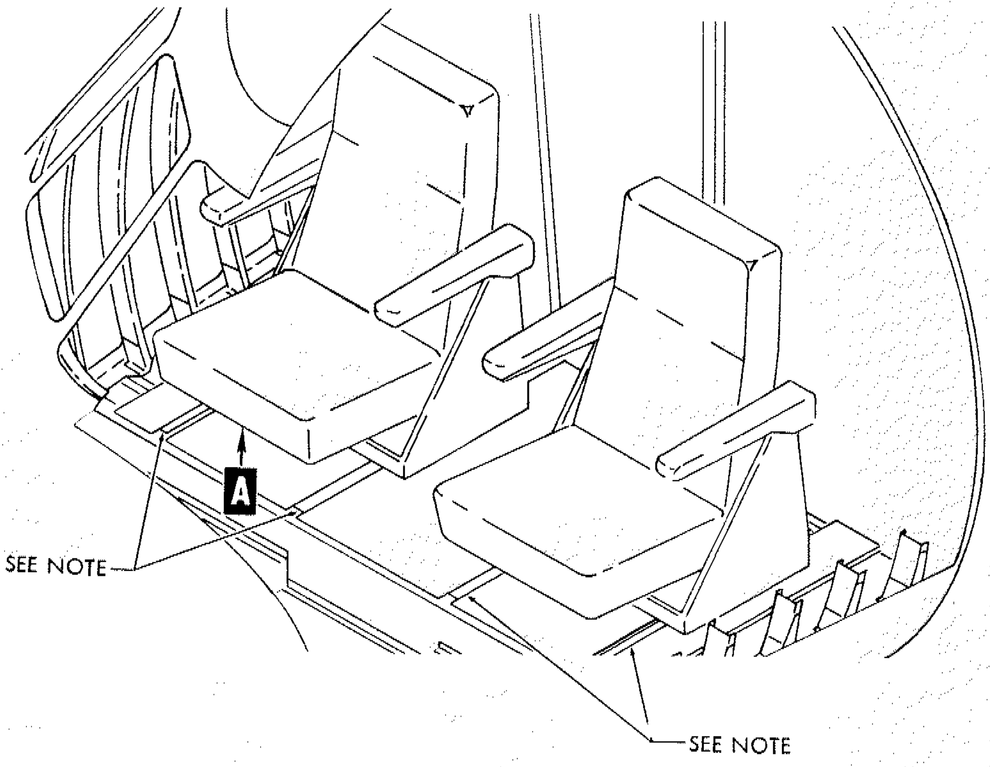
1. CLEAR VISION WINDSHIELD TRACKS (2 PLACES)

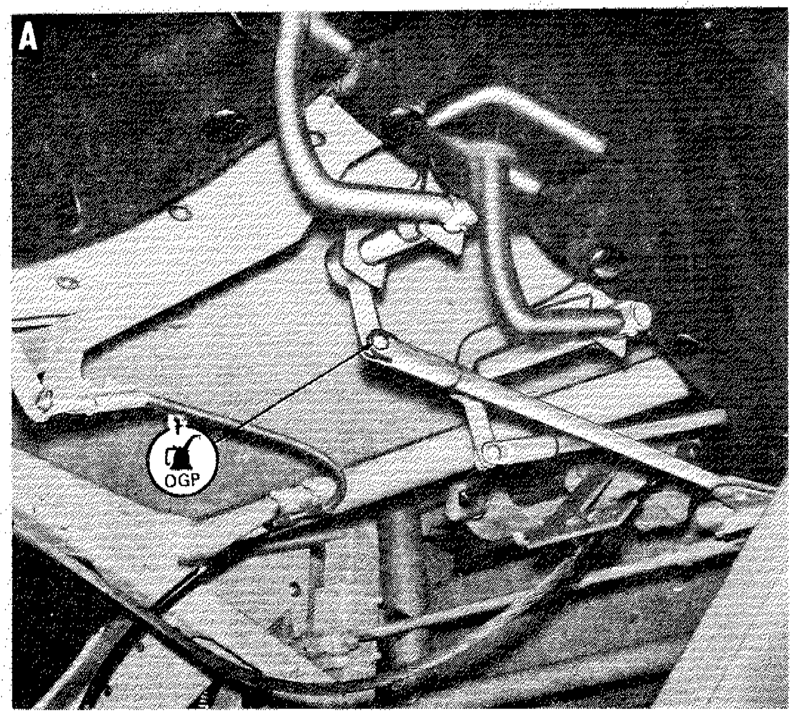
NOTE

RIGHT SIDE SHOWN
LEFT SIDE OPPOSITE

JS-2-2-1232

Figure 12-32. Lubrication of the Clear Vision Windshield





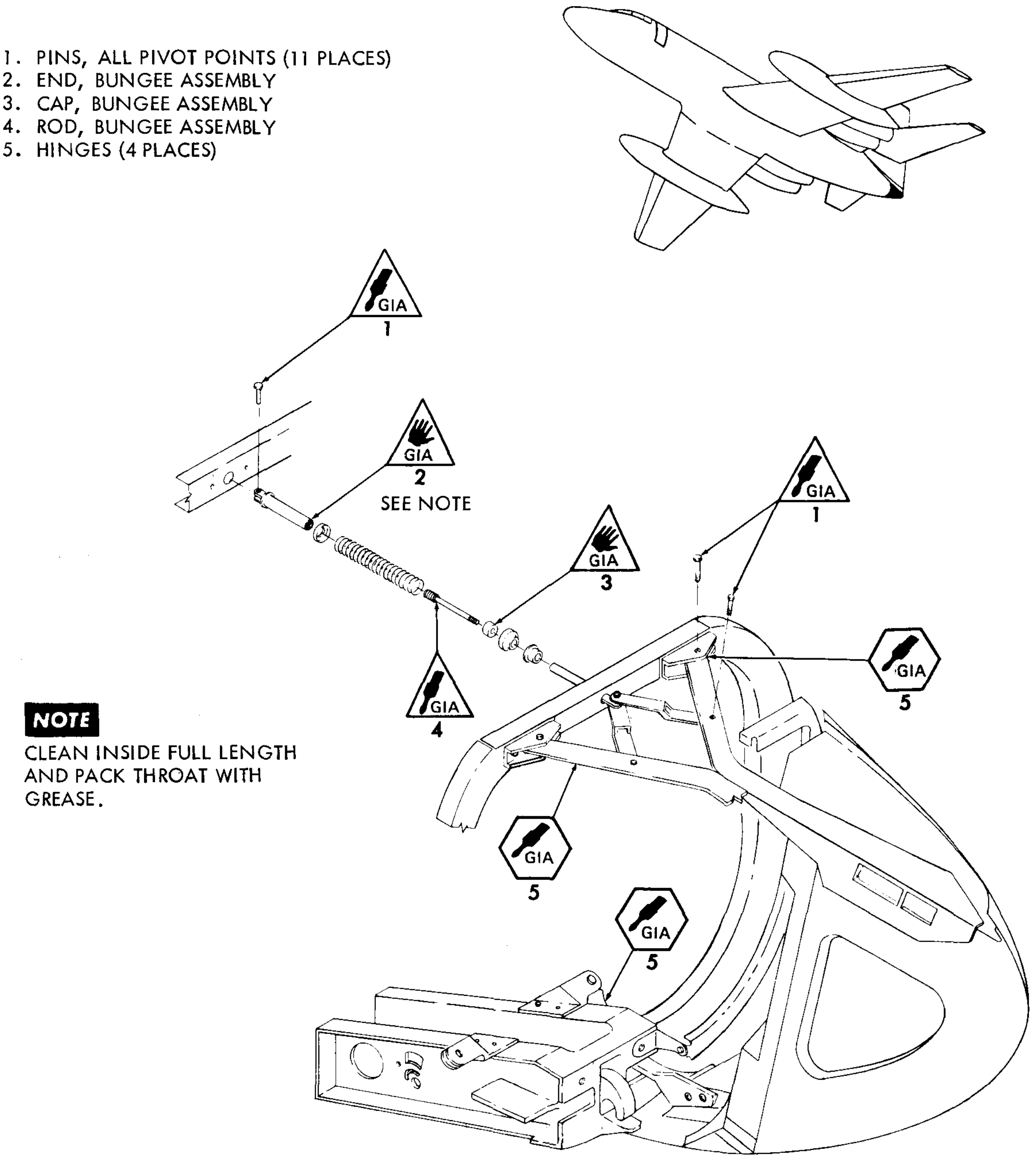
1. ALL PIVOT POINTS IN SEAT ADJUSTMENT MECHANISM, BOTH SEATS.

NOTE

1. CLEAN AND APPLY LUB (MDS) TO UPPER AND LOWER TRACK SURFACE WHENEVER SEATS ARE REMOVED.
2. CLEAN TRACK LOCKING PIN HOLES.

JS-1-2-X0/2-1233

Figure 12-33. Lubrication of the Pilots' Seats

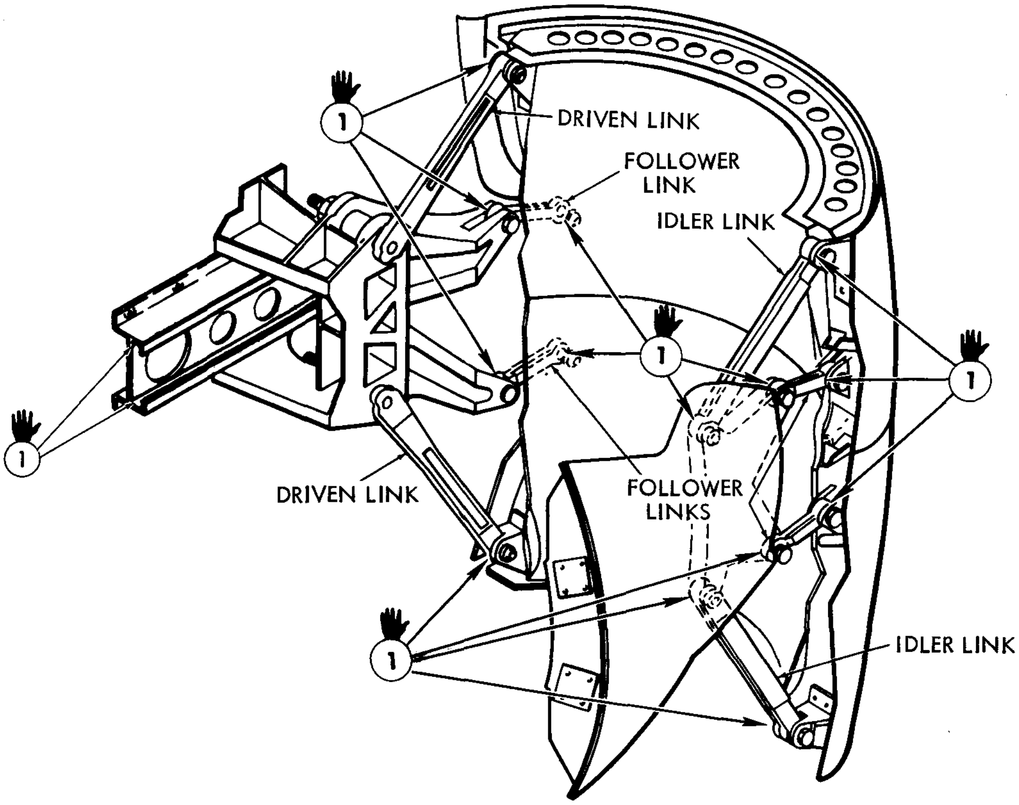


1. PINS, ALL PIVOT POINTS (11 PLACES)
2. END, BUNGEE ASSEMBLY
3. CAP, BUNGEE ASSEMBLY
4. ROD, BUNGEE ASSEMBLY
5. HINGES (4 PLACES)

NOTE

CLEAN INSIDE FULL LENGTH AND PACK THROAT WITH GREASE.

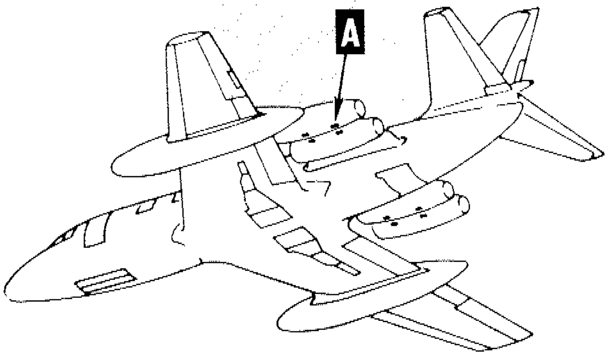
Figure 12-34. Lubrication of the Drag-Chute Mechanism (Airplanes 5001 through 5120 and/or those Airplanes Maintaining Operable Drag-Chute Systems)



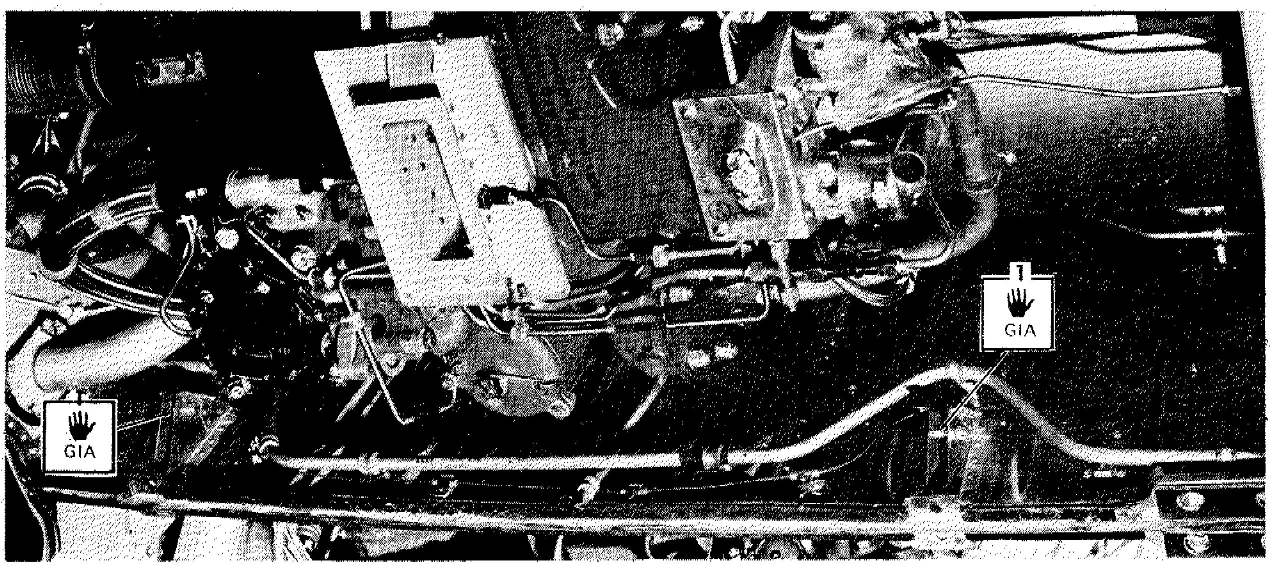
NOTE

1. EVERY 600 HOURS WITH ARMITE LED-PLATE 250
2. SEE CHAPTER 78 FOR FURTHER DETAILS

Figure 12-35. Lubrication of the Thrust Reverser



1. NACELLE ACCESS DOOR LATCH LUBRICATION (TYPICAL 8 PLACES)



VIEW LOOKING UP AT UNDERSIDE OF ENGINE NACELLE

Figure 12-36. Lubrication of Nacelle Latches

JS-1-2-X0/2-1236

All data on pages 12-81 through 12-152 deleted.

SECTION 12.14
EXTERIOR CLEANING

Contents	Page
EXTERIOR CLEANING	12-153
WHEEL WELLS	12-153
CONTROL SURFACES	12-153
GLASS WINDOWS	12-153
PLASTICS	12-153
NEOPRENE SURFACES	12-153

12.14.1. EXTERIOR CLEANING. Openings such as doors; hatches, ports, control surface wells, wheel wells, vents, and other openings, should be masked to prevent entrance of cleaning solutions. Also, surfaces such as windows and neoprene coated parts should be covered or otherwise protected. No protective covering is required for the rubber deicing boots on the wing and empennage. Use an alkaline emulsion cleaner (MIL-C-43616, Aero-Clene PF, B & B-2020 Plus, Oakite 214M, Turco 5975A, or similar) mixed in accordance with manufacturer's instructions. Alkaline emulsion cleaners should be thoroughly rinsed off with water before being allowed to dry. It is desirable that the unpainted surfaces should be neutral or slightly acid after cleaning. After thorough water rinsing, unpainted clad aluminum alloy surfaces may be rinsed with a 0.2 to 0.3 percent chromic-acid solution to obtain a neutral or slightly acid surface as checked with litmus paper applied to the wet surface.

12.14.2. WHEEL WELLS. Wheel wells should be cleaned with an alkaline emulsion cleaner followed by a cold water rinse. Hot cleaning should not be used.

12.14.3. CONTROL SURFACES. Control surfaces and wells should be cleaned with an alkaline emulsion cleaner with special care to prevent: (1) entrance of cleaner into areas that cannot be rinsed and (2) high pressure impingment that may cause control surface damage. Check condition of unpainted surfaces with litmus paper as indicated above.

12.14.4. GLASS WINDOWS. Glass windows should be cleaned with a mild detergent and rinsed immediately. Alkaline cleaner should not be used.

12.14.5 PLASTICS. Plexiglass and acrylic plastics should be cleaned only with soft, clean rags using a cleaner such as Mirror Bright Polish Co. MGH-10 mirror glaze or Plexiglas cleaner.

12.14.6. NEOPRENE SURFACES. Neoprene coated areas (black or brown covered plastic) should be cleaned with a mild detergent and rinsed immediately.

Note

See Chapter 51 for additional information on finishes and corrosion control.

CHAPTER 21
AIR CONDITIONING

TABLE OF CONTENTS

Section Page

21.0 AIR CONDITIONING SYSTEM 21-3

21.1 PRESSURIZATION SYSTEM 21-31

21.2 ELECTRONIC EQUIPMENT COOLING SYSTEM. 21-53

21.3 TOILET VENTILATION SYSTEM 21-59

SECTION 21.0
AIR CONDITIONING SYSTEM

Contents Page

- GENERAL 21-4
  - System Description 21-4
  - Normal Operation 21-5
    - Flight Operation 21-5
    - Ground Operation 21-7
  - Emergency Operation 21-7

- TEMPERATURE CONTROL 21-7
  - General 21-7
  - Cabin Temperature Control 21-7
  - Flight Station Temperature Control 21-8
  - Component Operation 21-8
    - Refrigeration Units 21-8
    - Bleed Air Flow Control Valve 21-9
    - Water Separator 21-9
    - Temperature Sensors 21-9
    - Temperature Anticipators 21-9
    - High-Limit Thermostats 21-9
    - Low-Limit Thermostat 21-10
    - Turbine Bypass Valve 21-10
    - Refrigeration Unit Bypass Valve 21-10
    - Flight Station Temperature Control Valve 21-10
    - Flight Station Air Conditioning Shutoff Valve 21-10
    - Temperature Control Box 21-11
    - Engine Bleed Air Shutoff Valve 21-11
    - Ram Air Shutoff Valve 21-11
    - Emergency Pressurization Shutoff Valve 21-11
    - Emergency Pressurization Heat Exchanger 21-11
    - Emergency Pressurization Temperature Control 21-11
    - Windshield De-Fogging 21-11
    - Anti-Icing Screens 21-11
    - Low-Limit Temperature Control Box 21-11
    - Flight Station Duct Sensor 21-11
    - Flight Station Conditioned Air Outlets 21-12

- OPERATIONAL CHECKOUT 21-12
  - General Requirements 21-12
  - Preparation for Testing 21-12
  - Operational Checkout Procedure 21-13

- TROUBLESHOOTING 21-16

- REMOVAL AND INSTALLATION 21-18
  - Removal of Air Conditioning Ducts 21-18
  - Installation of Air Conditioning Ducts 21-18
  - Removal of Water Separator 21-18A
  - Installation of Water Separator 21-18A
  - Removal of Refrigeration Unit 21-18A
  - Installation of Refrigeration Unit 21-18B
  - Removal of Shutoff Valves 21-18B
  - Installation of Shutoff Valves 21-18B
  - Removal of Anticipators 21-18B
  - Installation of Anticipators 21-18B
  - Removal of Low-Limit Thermostat 21-18B
  - Installation of Low-Limit Thermostat 21-18B
  - Removal of High-Limit Thermostat 21-18B

Contents Page

Installation of High-Limit Thermostat 21-18C
Removal of Temperature Sensor 21-18C
Installation of Temperature Sensor 21-18C
Removal of Refrigeration Unit Rotative Assembly. 21-18C
Installation of Regrigeration Unit Rotative Assembly 21-18C

SERVICING 21-18D
Adjustment 21-18D
Lubrication 21-18D
Cleaning 21-18D
Water Separator Coalescer 21-19
Engine Bleed Air Shutoff Valve Filter. 21-19
Bleed Flow Control Valve Filter. 21-19
System Preservation 21-19
Normal Operation 21-19
Airplane Storage 21-19

FIGURES

21-1. Air Conditioning and Pressurization System Components 21-20
21-2. Air Conditioning System Flow Schematic Diagram 21-21
21-3. External Power Connections and Intake Plug 21-23
21-4. Bleed Air Flow Control Valve (AiResearch Part No. 106756) 21-24
21-5. Bleed Air Flow Control Valve (Pneu-Tech Laboratories Part No. 31013) 21-25
21-6. Water Separator 21-26
21-7. Pneumatically Operated Shutoff Valves 21-27
21-8. Air Conditioning and Pressurization Systems Electrical Schematic Diagram (Airplanes 5001 through 5050) 21-29
21-8A. Air Conditioning and Pressurization Systems Electrical Schematic Diagram (Airplanes 5051, 5052, 5053) 21-30A
21-8B. Air Conditioning and Pressurization Systems Electrical Schematic Diagram (Airplanes 5054 through 5065) 21-30C
21-8C. Air Conditioning and Pressurization Systems Electrical Schematic Diagram (Airplanes 5066 through 5085) 21-30E
21-8D. Air Conditioning and Pressurization System Electrical Schematic Diagram (Airplanes 5086 through 5092 and 5094 through 5096) 21-30G
21-8E. Air Conditioning and Pressurization System Electrical Schematic Diagram (Airplanes 5093, 5097 and up) 21-30J

21.0.1 GENERAL (See figures 21-1 and 21-2.) The air conditioning system supplies temperature-conditioned air to the flight station and cabin and nose electronics compartment. The air supply is obtained from the compressor section of each engine, through a bleed air system. The air from the engine compressors is too hot for immediate use, so it is routed through refrigeration units to lower its temperature before it is supplied to the flight station and cabin outlets. The air temperature can be controlled either manually or automatically from a control panel in the flight station. The air conditioning exhaust is passed overboard through two outflow valves, which control the outflow rate to provide airplane pressurization. The distribution system is divided into a flight station and a cabin sub-system, each using the same basic air supply, but having individual temperature controls to vary the mixture of hot and cool air supplied to the outlets.

21.0.1.1 SYSTEM DESCRIPTION. (See figure 21-2.) Ninth-stage compressor bleed air is extracted from each engine at two points on the diffuser case, and flows into a single duct. The bleed air then flows through a bleed aircheck valve, which prevents reverse flow in the event of duct failure or low engine power. A bleed air shutoff valve is provided for each engine. The four bleed air lines are manifolded downstream of the shutoff valves. The bleed air supply line from No. 2 engine has a connection between the check valve and the shutoff valve, to supply bleed air to the emergency pressurization system, while a similar connection in the No. 3 engine bleed air supply line furnishes air to the wing and empennage de-icing system.

a. The hot bleed air flows from the bleed air manifold to the air conditioning system, through a bleed air flow control valve which limits the amount of air entering the system. At this point, the air supply is divided, part of it flowing to the distribution ducts through a refrigeration unit by-pass valve, and part flowing to the left and right refrigeration units for cooling. Air entering the refrigeration units is first cooled by passing through a ram air heat exchanger. The major

21.0.1.1 (Cont).

portion of this air is further cooled by expansion through a high-speed turbine, while the remainder flows to a water separator through a turbine bypass valve. The air leaving the turbine is cold, and it is mixed with the warmer bypassed air before entering the water separator in order to prevent freezing of the water in the coalescer section.

b. The water separator (see figure 21-6) removes most of the entrained moisture from the air, which then flows to the distribution ducts. Cool air for the flight station and for gasper outlet distribution is extracted from the main duct downstream of the water separator. Hot bleed air is then injected through the refrigeration bypass valve if necessary to raise the temperature of the remaining air to the desired value, and this mixed air flows to distribution ducts on each side of the airplane, for use in the cabin. The air for the flight station, previously extracted downstream of the water separator, is warmed if necessary by hot bleed air injected through the flight station temperature control valve. Then it flows to the flight station outlets, and the windshield de-fogging nozzles. Reverse flow of hot bleed air into the gasper system is prevented by a check valve in the duct upstream of the point at which hot bleed air is injected through the flight station temperature control valve. All the air conditioning supply ducts have check valves at the aft fuselage pressure bulkhead to prevent loss of cabin pressure in the event of loss of air conditioning supply air.

21.0.1.2 NORMAL OPERATION.

21.0.1.2.1 FLIGHT OPERATION. The air conditioning system is controlled from a panel above the flight station (see figure 21-1). The control panel has two TEMPERATURE CONTROL toggle switches, two TEMPERATURE CONTROL rotary selectors (one each for the flight station and cabin), and a temperature indicator. Airplanes 5086 and subsequent, and airplanes with Service Bulletin 329-213 or Service Bulletin 329-235 incorporated have two EMERG. MAN. CONTROL switches with guards. When guards are up and these switches are in any position other than OFF, all other control and switches, except EMERG. MAN. are inoperative. The TEMPERATURE CONTROL switches have "COOL," "WARM," "HOLD," and "AUTO" positions. The EMERG. MAN. CONTROL switches have "COOL," "WARM," "HOLD," and "OFF" positions. When a temperature control switch is turned to the "COOL" position, the applicable control valves move toward the position which gives maximum cold. In the "WARM" position, the control valves move toward the position giving maximum heat.

a. For manual operation, it is necessary to hold the switch in the "COOL" or "WARM" position until the desired change in temperature occurs, and then to place the switch in the "HOLD" position. This disconnects all power from the valves and they remain in the last-operated position. If the high-limit temperature is reached or, on airplanes 5001 through 5085, on airplanes 5093, 5097, and subsequent, or on airplanes 5086 through 5092 and 5094 through 5096 having Service Bulletin 329-234 incorporated, if the low limit temperature is reached in the ducts, the automatic overriding circuit adjusts the valves to bring the temperature within the proper limits, except as noted below.

b. When a TEMPERATURE CONTROL switch is placed in the "AUTO" position, the control valve positions are adjusted automatically to obtain the temperature selected by the TEMPERATURE CONTROL selector for the system. The selected temperature may be adjusted from approximately 55°F, with the selector in the "COLD" position, to approximately 95°F, with the selector in the "HOT" position. Comfortable temperature will be provided with a selector setting between the 12-o'clock position and "HOT." The exact position of the selector for a given temperature will vary slightly from airplane to airplane because of interior furnishing variations. The position for comfort must be determined by operating experience, and may then be marked on the selector dial if desired. At the higher altitudes and in low ambient humidity conditions it will be necessary to advance the selector toward the "HOT" position slightly to maintain equivalent comfort.

c. The temperature indicator on the control panel is connected to a sensor enclosed in the same box as the cabin temperature sensor. During automatic operation of the air conditioning systems, the control valves are positioned by electrical signals from the temperature control box. This box contains the electrical circuits to provide cabin temperature control, flight station temperature control, and low- and high-limit temperature control. On airplanes 5086 through 5092 and 5094 through 5096, not having Service Bulletin 329-234 incorporated, the low limit temperature control is omitted. On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, a separate control box is provided for low limit control. The cabin temperature control circuits receive input signals from a cabin temperature sensor, a cabin temperature anticipator, a cabin high-limit thermostat, and a cabin temperature selector. The signals from the sensor, anticipator, and selector are balanced on a wheatstone bridge circuit in the temperature control box when the cabin is stabilized at the temperature selected. Movement of the selector unbalances the bridge circuit, causing a signal to operate one of two relays to modulate the refrigeration unit bypass and turbine bypass valves toward the open position for warmer, or toward the closed position for

21.0.1.2.1 (Cont).

cooler air. These valves are operated in sequence to control the amount of warm air or hot air added to the previously cooled bleed air. A change in temperature from cold to hot is accomplished by moving the selector from the cold to the hot position, which unbalances the bridge circuit, causing the turbine bypass valve to be moved toward the open position. This causes more warm air (bleed air that has been routed through the heat exchanger only) to be added to the cold turbine discharge air, and at the same time causes less air to be forced through the turbine. If the turbine bypass valve opens fully and the bridge circuit has not reached a balanced condition, the refrigeration unit bypass valve will be energized toward the open position. This causes hot bleed air which has completely bypassed the refrigeration unit to be added to the mixed cold and warm air. On airplanes 5093, 5097, and subsequent, and on those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the position of the turbine bypass valve is controlled by the low limit temperature control box. When the temperature at the sensor is the same as selected temperature, the bridge circuit will be balanced and valve movement will cease. A change in temperature at the sensor will cause the bridge circuit to become unbalanced, resulting in a signal to move the valves in the direction to correct it. This cycle is reversed when the selector is moved from the full hot to full cold position. On airplanes 5086 through 5092 and 5094 through 5096, not having Service Bulletin 329-234 incorporated, the controls operate in essentially the same manner, except that the low limit system is no longer part of the control system (see paragraph 21.0.1.2.1 e.). On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, a separate low-limit control is provided which operates the turbine bypass valve to provide heat to de-ice the water separator. There is no control over the turbine bypass valve from the cabin controller. The cabin controller has control over the refrigeration bypass valve only, which adds or subtracts heat in response to signals from the cabin temperature sensor, the cabin temperature anticipator, and the cabin temperature selector. There is no electrical sequencing of the turbine bypass valve and the refrigeration bypass valve as on previous aircraft. The cabin high-limit thermostat provides protection against duct damage from overheating by completing a circuit to move the bypass valves toward the closed position when air temperature in the duct exceeds approximately 235°F. On airplanes 5086 and subsequent, and on those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, no cabin high limit control is available when the EMERG. MAN. CONTROL toggle switch is being used.

d. The flight station temperature circuits receive input signals from a flight station temperature selector, a flight station temperature anticipator (duct sensor), and on airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, a flight station temperature sensor. These signals are balanced on a wheatstone bridge circuit similar to the cabin temperature control circuits. The output signal operates a relay which controls the flight station temperature control valve in a manner similar to the cabin control system. A flight station high-limit thermostat provides duct overheat protection by automatically moving the flight station temperature control valve toward the closed position when air temperature in the duct exceeds approximately 235°F. On airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, no flight station high-limit control is available when the EMERG. MAN. CONTROL toggle switch is being used.

Note

On airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, the temperature control toggle switches should always be in the "HOLD" position when the airplane is on the ground and bleed air is off or engines are not operating. It may also be desirable in cold climates to operate the system manually until cabin temperature reaches the desired temperature, to prevent valves from automatically motoring to the full open position, resulting in a possible sudden surge of air into the cabin. On airplanes 5086 and subsequent, and those airplanes on which Service Bulletin 329-213 or Service Bulletin 329-235 has been incorporated, the EMERG. MAN. control toggle switches may be placed in the "HOLD" position when the airplane is on the ground and bleed air is off or engines are not operating. This prevents the valves and controls from having signals and power on them constantly and stops the cabin sensor fan motor.

e. On airplanes 5001 through 5085, airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-234 incorporated, a low-limit thermostat, downstream of the water separator, prevents ice formation in the water separator. When duct temperature is lowered to approximately 39°F, low-limit control circuits override the normal temperature control circuits and move the turbine bypass valve toward the open position to raise the temperature of the air passing through the water separator. On airplanes 5086 through 5092 and 5094 through 5096, not having Service Bulletin 329-234 incorporated, a screen is in-

21.0.1.2.1 (Cont).

stalled in the ducts between each turbine and the water separator. Under conditions which will tend to cause ice formation, the ice will begin to collect on the screens prior to ice formation in the water separator. Ice on the screen assemblies will restrict the area of the duct, causing the backpressure on the turbine to increase, and resulting in a higher temperature air being supplied to the water separator inlet. On airplanes 5046 and up, and on those on which Service Bulletin 329-108 has been installed, no low-limit control is available when temperature control toggle switches are in the cool position. On airplanes 5086 and subsequent, and on those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, no low-limit control is available when the cabin EMERG. MAN. CONTROL toggle switch is being used.

21.0.1.2.2 *GROUND OPERATION.*

Note

When on the ground with no engines operating or when engines are operating and bleed air is shut off, the temperature control toggle switches should be placed in the "HOLD" position to prevent unnecessary operation of the temperature control system. When electrical power is on for long periods of time, and the air conditioning system is not operating, the cabin and flight station sensors should be de-activated by disconnecting the electrical connector or opening the circuit breaker, to prevent unnecessary operation of sensor fans. Airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, do not have a flight station temperature sensor.

The airplane is equipped for two methods of ground cooling when engines are not operating:

a. Air Conditioning Ground Cart - A ground air conditioning cart may be connected through a MS 33561 type coupling located in the aft compartment (see figure 21-3). A flat cover plate is held over the opening with a Marman clamp when the ground cart is not connected. All air supplied through the coupling is routed directly to the distribution outlets in the cabin and flight station. All control of the temperature and flow rate of the cool air must be accomplished at the external unit. This connection may also be used during certain test procedures for connection of a source of air for pressurization. The pressure and flow from the unit should be limited to 10 PSI and 60 pounds (600-800 cu. ft.) per minute maximum, to prevent possible damage to ducts, equipment, etc.

b. External Bleed Air - A receptacle (see figure 21-3) is provided in the aft compartment to which a source of high-pressure air may be connected. The connection is located on the engine bleed air manifold, and consists of a flanged boss on the duct with a flanged cap retained by a Marman-type clamp installed when the external bleed air source is not connected. The connection is provided primarily for use in operational checkout of the air conditioning and pressurization systems without operating the engines. It may also be used for cooling purposes if a ground air conditioning cart is not available. Electrical power must be connected, so the temperature control system can be used to control temperature of the air supplied to the flight station and cabin outlets. The supply air pressure must be limited to 140 PSI; its temperature to 460°F; and the source should be capable of supplying 60 pounds (600-800 cu. ft.) of air per minute.

21.0.1.3 EMERGENCY OPERATION. If the cabin temperature cannot be controlled either automatically with the selector or manually by the toggle switches, the emergency pressurization system can be used. This system is selected by placing the EMER PRESS SYSTEM switch on the pressurization control panel in the "ON" position, then using the INCR TEMP - DECR TEMP switch as required for comfort. If the temperature cannot be lowered sufficiently by this method, which may occur at altitudes below approximately 14,000 feet, place the EMER PRESS SYSTEM switch in the "OFF" position; then place the DEPRESS switch in the depressurization position, followed by use of the ram air switch to open the ram air valve as required to circulate outside air through the cabin.

Note

Ground operation and flight operation of the emergency pressurization system below a 10,000-foot altitude should be avoided due to possible overheating of ducts and system components.

21.0.2 TEMPERATURE CONTROL.

21.0.2.1 GENERAL. The bleed air extracted from the engines is extremely hot due to compression, therefore reduction of its temperature before it enters the cabin and flight station is the only requirement of the temperature control system.

21.0.2.2 CABIN TEMPERATURE CONTROL. Cabin air temperature control is accomplished by positioning of the refrigeration unit bypass valve and the turbine bypass valve. These valves are controlled by electrical signals from a temperature control box, or from the manual controls on the air conditioning control panel. The positions of these two bypass valves determine the proportions of hot air, warm air, and cold air to be supplied to the cabin. For maximum heating, the turbine bypass valve is moved to the open position; when this valve is fully open, the refrigeration unit bypass valve also moves to the fully open position. On aircraft 5093, 5097, and subsequent, and

21.0.2.2 (Cont).

on those airplanes with Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the movement of the refrigeration unit bypass valve is independent of the movement of the turbine bypass valve. With both bypass valves fully open, the duct pressure upstream is only slightly higher than cabin pressure. This causes the bleed air flow control valve to move towards the closed position, to maintain correct flow schedule, and the low pressure causes only a small amount of air to pass through the cooling turbines. The conditioned air supplied to the cabin is a mixture of these hot, warm and cold flows.

a. When the temperature control box sends signals for colder air, the refrigeration unit bypass valve moves to the closed position, and the turbine bypass valve then moves to the closed position. Closing these paths of low resistance to flow causes the pressure upstream of the bypass valves to increase, which causes the bleed air flow control valve to open to maintain flow. When both of the bypass valves are closed, the upstream pressure reaches a peak, causing all of the air supplied to the cabin to pass through the cooling turbines.

b. Operation of the turbine bypass and refrigeration unit bypass valves is controlled by a four-position ("COOL", "WARM", "HOLD", and "AUTO") CABIN TEMPERATURE CONTROL switch. This is located on the air conditioning control panel above the flight station. The valves can be positioned by manual selection by placing the switch in the "WARM" or "COOL" position; or the valves can be controlled automatically by placing the switch in the "AUTO" position. When the switch is in "AUTO", the desired cabin temperature is selected by use of a CABIN TEMPERATURE CONTROL selector on the air conditioning control panel. The cabin temperature control box transmits the controlling signals to maintain the desired temperature.

c. On airplanes 5086 and subsequent, operation of the turbine bypass and refrigeration unit bypass valves is also controlled by a guarded four-position ("COOL", "WARM", "HOLD", and "OFF") CABIN EMERG. MAN. CONTROL switch. This is located on the air conditioning control panel above the flight station. This switch may be used to control cabin temperature if a failure occurs that prevents use of the CABIN TEMPERATURE CONTROL switch. The EMERG. MAN. CONTROL switch is normally in the "OFF" position. The valves can be positioned by manual selection by raising the guard and placing the switch in the WARM" or "COOL" position. When the switch is in the "HOLD" position, the valves will remain in the last position selected. On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the CABIN TEMPERATURE CONTROL switch modulates the refrigeration bypass valve but does not affect the position of the turbine bypass valve. The EMERG. MAN. CONTROL switch modulates the turbine bypass valve but does not move the refrigeration bypass valve except toward the closed position.

21.0.2.3 FLIGHT STATION TEMPERATURE CONTROL. Operation of this system is similar to that of the cabin temperature control system. The same temperature control box is used, and the air supply is the same mixture of cold air from the heat exchangers, diverted into the flight station air conditioning ducts before the hot bleed air for the cabin is injected. Hot bleed air for the flight station system is obtained from a separate bleed air duct. This duct is teed into the bleed air manifold upstream of the flow control valve and connected to the flight station air conditioning duct. Bleed air flow is regulated by a flight station control valve, which is controlled by the temperature control box or by a four-position ("COOL", "WARM", "HOLD", and "AUTO") FLIGHT STATION TEMPERATURE CONTROL switch. The control switch is located on the air conditioning control panel. This switch has the same functions as the cabin temperature control switch, but affects only flight station temperature. On airplanes 5086 and subsequent, and those having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, the flight station temperature control valve may also be controlled by a guarded four-position ("COOL," "WARM," "HOLD," and "OFF") FLIGHT STA. EMERG. MAN. CONTROL switch on the air conditioning control panel. This switch has the same function as the cabin emergency manual control switch, but affects only flight station temperature.

a. A flight station air conditioning shutoff valve, installed in the bleed air duct to the flight station air conditioning duct, is controlled by a FLT STA AIR COND SHUTOFF switch on the bleed air control panel. The switch energizes the valve to the closed position to prevent hot bleed air flow to the flight station duct when the switch is placed in the "SHUTOFF" position.

Note

To shut off all air supply to the flight station, both FLT STA AIR COND and CABIN AIR COND must be shut off. (See figure 21-2.)

21.0.2.4 COMPONENT OPERATION.

21.0.2.4.1 REFRIGERATION UNITS. The air conditioning system includes two refrigeration units (see figure 21-1). Each unit consists of a heat exchanger, a cooling air fan, and an expansion turbine. Hot bleed air is passed through the heat exchanger and is cooled by ambient air flowing across the exchanger. Ram air is obtained from a scoop in the leading edge of each engine pylon. The cooled bleed air is divided downstream of the heat exchanger. During normal oper-

21.0.2.4.1 (Cont).

ation, part of the air goes to the air conditioning system for distribution, and the remainder to the high-speed turbine for further cooling. The turbine is driven at approximately 60,000 RPM by this air when maximum cooling is demanded. In driving the turbine, the air expands and additional heat is extracted from it. This cold air then flows to the air conditioning ducts. The ram air fan in each refrigeration unit is driven by the corresponding expansion turbine. The fan absorbs power from the turbine to accomplish cooling, and it moves air through the ram air ducts when ram air pressure is too low, such as during ground operation of the system. Enough air is drawn through the ducts to permit normal system operation on the ground without overheating the ducts. Since obstruction of the ram air scoops can cause the unit to fail through overheating, spring-loaded doors in the ducts, immediately upstream of the heat exchanger, open automatically. This permits the affected fan to draw in enough cooling air from the aft compartment if an inlet obstruction occurs. Each fan is "single-ended," with both bearings on one end of the shaft, so the bearing housing is completely surrounded by the cold air discharge from the corresponding high-speed turbine. The ram air is exhausted overboard.

21.0.2.4.2 BLEED AIR FLOW CONTROL VALVE. (See figures 21-4 and 21-5.) The amount of air flowing through the air conditioning system is limited by a venturi-type air flow regulator incorporating a modulating butterfly valve. The butterfly shutoff valve is normally open, but the amount of valve opening is varied to maintain a scheduled flow of inlet air. A duct pressure of 10 PSI, as well as electrical power at the solenoid, is necessary to close the shutoff valve. The solenoid is energized when the CABIN AIR COND SHUTOFF switch on the bleed air control panel is placed in the "SHUTOFF" position. Airplanes 5001 through 5045 are equipped with an AiResearch Part Number 106756 valve (see figure 21-4). Airplanes 5046 and up are equipped with a Pneu-Tech Part Number 31013 valve (see figure 21-5). This part is also supplied as spares for airplanes 5001 through 5045.

21.0.2.4.3 WATER SEPARATOR. (See figure 21-6.) Moisture is removed from the air conditioning air supply by a water separator. The water separator consists of a coalescer on a conical surface, an impingement surface, a filtered drain, and an internal bypass valve. The coalescer causes free moisture particles to collect into large water drops. The conical surface is louvered to give the air a swirling motion and to cause the drops to collect on the impingement surface. From here they drain overboard through a drain tube system. If the coalescer becomes clogged with dirt or ice to the extent that the differential pressure across the coalescer increases to 4.50 (±0.50) inches of mercury, the bypass valve opens and permits free air flow through the separator. On airplanes 5001 through 5085, 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-234 incorporated, icing in the water separator is normally prevented by the action of the low-limit thermostat downstream of the separator. On airplanes 5086 through 5092 and 5094 through 5096 not having Service Bulletin 329-234 incorporated, separator icing is normally prevented by function of the screens upstream of the separator. The water drain fitting has a filter to prevent clogging of the drain tube. (See paragraph 21.0.6.3 for filter maintenance instructions.)

21.0.2.4.4 TEMPERATURE SENSORS. A temperature sensor is provided for the cabin. Airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, have a temperature sensor in the flight station. Each sensor continually transmits an electrical signal to the temperature control box, according to the temperature at the sensor location. An electric-motor-driven blower circulates air over the sensing element to ensure accurate sampling. The flight station sensor is on the trim panel aft of the right clear view window, adjacent to the smoke goggles stowage compartment. The cabin sensor location varies with the custom interior installation. Airplanes 5086 and subsequent, and those having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, have a duct sensor installed in the flight station air supply duct (see paragraph 21.0.2.4.21).

21.0.2.4.5 TEMPERATURE ANTICIPATORS. The temperature of the air in the cabin air supply duct is sensed by a temperature anticipator in the duct. (See figure 21-1.) On airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, a temperature anticipator is located in the flight station air supply duct. Each unit varies an electrical signal according to duct temperature variations, to prevent overshooting or undershooting the selected compartment temperature. During temperature variations, the air in the supply ducts reaches the desired temperature before the compartment air does. The resulting electrical signal from the anticipator is transmitted to the temperature control box to stop movement of the temperature control or bypass valves before the compartment air reaches the desired temperature. The flight station anticipator is in the main flight station supply duct, forward of the rear pressure bulkhead check valve, at approximately fuselage station 510. The cabin anticipator is in the aft compartment at approximately fuselage station 580, in the right branch of the cabin distribution duct.

21.0.2.4.6 HIGH-LIMIT THERMOSTATS. Because the flight station and cabin air temperatures can be controlled independently of each other, a high-limit thermostat is installed in the distribution duct (see figure 21-1) of each system. The thermostats are mounted with the temperature-sensitive probe inside the duct, to measure air temperature in the duct. When the calibrated temperature is reached, a switch in the unit is actuated from one position to another, to complete circuits which supply cooler air by ad-

21.0.2.4.6 (Cont).

justing the corresponding temperature control valves. Both thermostats are calibrated to permit a maximum duct air temperature of 235°F. The high-limit thermostats are adjacent to the corresponding temperature anticipators. The flight station thermostat is in the flight station supply duct, at approximately fuselage station 503, and the cabin thermostat is in the aft compartment, in the right branch of the cabin distribution duct, at approximately fuselage station 580. For airplanes 5046 and up, and on airplanes having Service Bulletin 329-106 incorporated, the cabin high-limit thermostat is located in the left hand branch of the cabin distribution duct. On airplanes 5054 thru 5085 the high-limit thermostats are connected to thermostat bypass relays (external to the thermostat) and to the temperature controller. Each relay coil is connected to 28-volts DC in series with the normally closed contacts of its system thermostat. The relay uses one set of contacts to open the hot circuit and a completely isolated set of contacts to actuate the cold circuit. In addition, the manual control switches are connected directly to their respective power sources. On airplanes 5051 thru 5085 the cabin station thermostat circuit has a bypass relay. The EMERG. MAN. CONTROLS, in lieu of relays, bypass their respective high-limit thermostats on airplanes 5086 and subsequent, and those having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated.

21.0.2.4.7 LOW-LIMIT THERMOSTAT. On airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, when the air temperature in the supply duct, immediately downstream (forward) of the water separator, reaches 39 (±5)°F, the low-limit thermostat (see figure 21-1) is actuated to initiate a valve operation cycle. This increases air temperature, and prevents ice formation in the water separator. The low-limit signal always overrides any conflicting signal from the automatic control, and positions the turbine bypass valve for warmer air. A similar function exists in the manual control mode of operation on airplanes 5001 through 5045, except those on which Service Bulletin 329-108A has been installed. The low-limit thermostat has no function at temperatures above the calibrated control range. The temperature-sensitive probe of the thermostat extends inside the air duct, in direct contact with the air leaving the water separator. The duct in which the thermostat is mounted is in the aft compartment forward of the water separator. On airplanes 5086 through 5092 and 5094 through 5096 not having Service Bulletin 329-234 incorporated, ice formation in the water separator is retarded by screens upstream of the separator (see paragraph 21.0.2.4.19). On airplanes 5093, 5097, and subsequent, and those having Service Bulletin 329-213, Service Bulletin 329-234 or Service Bulletin 329-235 incorporated, the low limit thermostat modulates the turbine bypass valve to control the water separator discharge temperature to 39 (±5)°F through an automatic control system which receives only low limit control signals.

21.0.2.4.8 TURBINE BYPASS VALVE. This is an electric motor-driven butterfly valve. An external pointer indicates the butterfly position, as an aid to system troubleshooting. The valve is moved by a modulated signal originating in the cabin or low-limit bridge circuit, and controlled by the temperature control box or by a constant signal from the manual control toggle switch. On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234 or Service Bulletin 329-235 incorporated, the valve is moved by a modulating signal from the low-limit bridge only, and has no interconnection to the cabin bridge circuit. On airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, the valve may also be moved by a constant signal from the cabin emergency manual control toggle switch. The valve is in the aft compartment, in the duct routed aft into the turbine discharge duct, to the right of the water separator. (See figure 21-1.)

21.0.2.4.9 REFRIGERATION UNIT BYPASS VALVE. This is a motor-driven, butterfly valve. An external pointer indicates the butterfly position, as an aid to troubleshooting. The valve is moved by a modulated signal originating in the cabin temperature control bridge circuit, and controlled by the temperature control box or by a constant signal from the manual control toggle switch. On airplanes 5086 through 5092 and 5094 through 5096, not having Service Bulletin 329-234 incorporated, the valve is also moved by a constant signal from the cabin emergency manual control toggle switch. On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the valve is also moved to the closed position only by a constant signal from the cabin emergency manual control toggle switch. For these airplanes, the cabin emergency manual control toggle switch will not open the valve. The valve is in the aft compartment, in the duct routed forward, which tees into the water separator exit duct just forward of the water separator. (See figure 21-1.)

21.0.2.4.10 FLIGHT STATION TEMPERATURE CONTROL VALVE. This is an electric motor-driven, butterfly valve. An external pointer indicates the valve position. The valve is moved by a modulating signal originating in the flight station temperature control bridge circuit and controlled by the temperature control box or by a constant signal from the flight station manual control toggle switch. On airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, the valve is also moved by a constant signal from the flight station emergency manual control toggle switch. Increased opening of this valve increases flight station air temperature; decreased opening decreases air temperature. The valve is in the aft compartment, in the duct upstream of the aft pressure bulkhead flight station supply line check valve. (See figure 21-1.)

21.0.2.4.11 FLIGHT STATION AIR CONDITIONING SHUTOFF VALVE. (See figure 21-7). This is a nor-

21.0.2.4.11 (Cont).

mally closed, pneumatically actuated, solenoid controlled poppet valve. The valve is in the aft compartment, upstream of the flight station temperature control valve (see figure 21-1). This valve is closed when the FLT STA AIR COND SHUTOFF-switch is in the "SHUTOFF" position and upstream duct pressure of 10 PSI exists.

21.0.2.4.12 TEMPERATURE CONTROL BOX. The combined electrical signal outputs of all temperature sensors and controls are coordinated in the temperature control box, resulting in electrical signals for positioning the system valves to obtain the temperatures selected on the air conditioning control panel. On airplanes 5093, 5097, and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the turbine bypass valve is not controlled by this box. Both flight station and cabin temperatures are controlled through this unit. See figure 21-8 for the internal components and circuitry. The control box is in the aft compartment, on the generator control panel rack (see figure 21-1).

21.0.2.4.13 ENGINE BLEED AIR SHUTOFF VALVE. (See figure 21-7.) This is normally closed, pneumatically actuated, solenoid controlled butterfly valve. The four valves are in the aft compartment, one in each engine bleed air supply line (see figure 21-1). These valves are closed when the bleed air shutoff switches are in the "SHUTOFF" position and upstream pressure of 4 PSI is available.

21.0.2.4.14 RAM AIR SHUTOFF VALVE. This is an electric motor-operated butterfly valve, closed during normal system operation. It is on the right side of the aft compartment (see figure 21-1.) This valve can be operated only when the NORMAL-DEPRESS switch on the pressurization control panel is in the DEPRESS position (see figure 21-8).

21.0.2.4.15 EMERGENCY PRESSURIZATION SHUTOFF VALVE. This is an electric-motor-operated shutoff valve (see figure 21-1). It is operated to the fully open position when the EMER PRESS SYSTEM switch is placed in the "ON" position.

21.0.2.4.16 EMERGENCY PRESSURIZATION HEAT EXCHANGER. The temperature of emergency pressurization bleed air can be controlled by varying the amount of ram air routed across the emergency pressurization heat exchanger. The unit is on the left side of the fuselage aft compartment. It contains a heat exchanger through which bleed air from the No. 2 engine is routed. Ram air from the left-hand scoop is passed through the emergency pressurization ram air control valve and across the heat exchanger to cool the bleed air. The degree of cooling depends on the position of the ram air control valve, which regulates the amount of ram air flowing across the heat exchanger.

21.0.2.4.17 EMERGENCY PRESSURIZATION TEMPERATURE CONTROL. Ram air for cooling the emergency pressurization heat exchanger passes through the ram air control valve. This valve, in the form of a flapper door (see figure 21-1), is positioned by an electric-motor-driven actuator controlled by the emergency pressurization temperature switch on the pressurization control panel above the flight station. Electrical power for the ram air control actuator is obtained from the 28-volt, DC essential bus, through a circuit which is armed when the EMER PRESS SYSTEM SWITCH is on.

21.0.2.4.18 WINDSHIELD DE-FOGGING SYSTEM. Air from the flight station air conditioning system is used for windshield de-fogging. A discharge nozzle for each forward windshield directs a stream of air over the inner surface of the windshield glass. The system functions when air is flowing to the flight station through the supply ducts during normal system operation or when ram air ventilation is being used, provided the manual shutoff slides, one for the pilot and one for the copilot, are open.

21.0.2.4.19 ANTI-ICING SCREENS. Airplanes 5086 through 5092, and 5094 through 5096, not having Service Bulletin 329-234 incorporated, have a screen assembly installed in each duct between each refrigeration unit and the water separator. Under conditions which will tend to permit ice formation, the ice will begin to collect on the screen assemblies prior to icing of the water separator. Ice on the screen assemblies will restrict the area of the duct, causing the back pressure on the refrigeration unit turbine to increase, resulting in higher temperature (approximately 35° F) air being supplied to the water separator inlet. The screens are in the aft compartment, inside the duct joints on each side of the "Y" aft of the water separator inlet (see figure 21-1).

21.0.2.4.20 LOW-LIMIT TEMPERATURE CONTROL BOX. On airplanes 5093, 5097 and subsequent, and those airplanes having Service Bulletin 329-213, Service Bulletin 329-234 or Service Bulletin 329-235 incorporated, a signal from the low-limit thermostat is input to the low-limit temperature control box, resulting in an electrical signal for positioning the turbine bypass valve. The valve's position is controlled to maintain a 39 (±5)°F temperature at the discharge side of the water separator. See figure 21-8 for the internal components and circuitry. The low-limit control box is in the aft compartment, on the generator control panel rack (see figure 21-1).

21.0.2.4.21 FLIGHT STATION DUCT SENSOR. Airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, have a duct sensor in the flight station air supply duct. The sensor continually transmits an electrical signal to the temperature control box, according to the temperature at the sensor loca-

21.0.2.4.21 (Cont).

tion. The flight station duct sensor is in the main flight station supply duct, forward of the rear pressure bulkhead check valve, at approximately Fuselage Station 510 (see figure 21-1).

21.0.2.4.22 FLIGHT STATION CONDITIONED AIR OUTLETS.

a. Gasper outlets are provided for the pilot and copilot, and are located on the pilot's and copilot's inclined panels (see figure 21-1). The air flow quantity and direction may be regulated by adjusting the gaspers.

b. Body warmer outlets are provided for the pilot and copilot, and are located at the forward ends of the pilot's and copilot's side console panels (see figure 21-1). The air flow quantity and direction may be regulated by adjusting the body warmers.

c. Lower outlets are provided for the pilot and copilot, and are located to discharge above the pilot's and copilot's rudder pedals. The air flow quantity may be regulated by the "FOOT WARMER PULL" handles located on the main instrument panel (see figure 21-1).

The body warmers and the lower outlets supply air from the same conditioned air distribution duct. For maximum system efficiency, whether "HOT" or "COLD" has been selected, both body warmers and both lower outlets should be in the full open position.

21.0.3 OPERATIONAL CHECKOUT.

21.0.3.1 GENERAL REQUIREMENTS. Two men are required to perform the operational checkout of the air conditioning system. Man A will be stationed in the flight station to operate the controls. Man B will check the airflow from the flight station and cabin air conditioning outlets as required. This test is performed by operating any two of the engines as a source of bleed air. It is preferable to operate engines No. 2 and No. 3. To avoid pressurizing the fuselage, open the personnel entrance door before commensing the checkout.

21.0.3.2 PREPARATION FOR TESTING.

a. Start the selected engines in accordance with the instructions in Chapter 72. Run the engines above approximately 80 percent RPM to assure that sufficient bleed air pressure is available to close bleed air shutoff valves. All four engines may be operated if it is also desired to check for duct leakage or bleed air shutoff valve operation at the same time.

b. Energize the airplane AC and DC electrical systems.

c. Position the following switches on the pressurization control panel as noted:

SWITCH	POSITION
NORMAL-DEPRESS	"NORMAL" (Guarded)
RAM AIR	"CLOSE"
EMER PRESS control	"OFF" (Guarded)
EMER PRESS temperature	Centered

d. Position the following controls on the air conditioning control panel as noted:

CONTROL	POSITION
FLIGHT STATION TEMPERATURE switch	"HOLD"
CABIN TEMPERATURE switch	"HOLD"
FLIGHT STATION TEMPERATURE selector	MID-RANGE (12 O'CLOCK POSITION)
CABIN TEMPERATURE selector	MID-RANGE (12 O'CLOCK POSITION)
FLIGHT STATION EMERGENCY MANUAL CONTROL switch (Airplane 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated)	"OFF" (Guarded)
CABIN EMERGENCY MANUAL CONTROL switch (Airplanes 5086 and subsequent, and those airplanes having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated)	"OFF" (Guarded)

e. Position the following switches on the bleed air control panel as noted:

SWITCH	POSITION
BLEED AIR(four switches)	"SHUTOFF"
FLT STA AIR COND SHUTOFF	"NORMAL"
CABIN AIR COND SHUTOFF	"NORMAL"

21.0.3.2 (Cont).

f. Check that the following circuit breakers, on the pilot's wainscot circuit breaker panel, are engaged:

EMER PRESS
EMER DEPRESS
CABIN TEMP CONT
FLT STA TEMP CONT
TEMP CONT IND
TURB & AIR COND VALVES

21.0.3.3 OPERATIONAL CHECKOUT PROCEDURE.
*Denotes steps applicable only to airplanes 5001 thru 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated.
**Denotes steps applicable only to airplanes 5086 and subsequent, and those having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated.
Check the operation of the air conditioning system as follows:

Note

Troubleshoot in accordance with section 21.0.4 if system fails to function as indicated.

a. Man A: Place the ENG BLEED AIR switch for each engine operated in the "NORMAL" position.

b. Man A: Hold the FLIGHT STATION and CABIN temperature control toggle switches in the "COOL" position for approximately one (1) minute; then release them to the "HOLD" position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Cool air should flow from all outlets in the airplane.

c. Man A: Turn the FLIGHT STATION and CABIN temperature selectors to "HOT".
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Cool air should continue to flow from all outlets in the airplane.

d. Man A: Turn the FLIGHT STATION and CABIN temperature selectors to "COLD".
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Cool air should continue to flow from all outlets in the airplane.

e. Man A: Turn the FLIGHT STATION and CABIN temperature selectors to the mid-range (12 o'clock) position. Hold the flight station temperature control switch in the "WARM" position for approximately one (1) minute; then release to the "HOLD" position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become warmer. The airflow temperature from the cabin outlets should remain cool.

f. Man A: Turn the FLIGHT STATION and CABIN temperature selectors to "COLD" and then to "HOT".
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The temperature of the air coming from all outlets should not change.

g. Man A: Turn the FLIGHT STATION and CABIN temperature selectors to the mid-range (12 o'clock) control switch in "COOL" position for approximately one (1) minute; then release to "HOLD" position.
Man B: Check the flight station and cabin outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become cooler. The airflow temperature from the cabin outlets should not change.

h. **Man A: Hold the FLIGHT STATION emergency manual temperature control switch in the "WARM" position long enough to determine a change in air temperature; then release to the "HOLD" position.

CAUTION

The flight station high limit thermostat is inoperative during this portion of checkout. Do not permit ducts to overheat.

Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become warmer. The airflow temperature from the cabin outlets should not change.

i. **Man A: Hold the FLIGHT STATION emergency manual temperature control switch in the "COOL" position for approximately one (1) minute; then release to the "HOLD" position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become cooler. The airflow

21.0.3.3 (Cont).

temperature from the cabin outlets should not change.

j. **Man A: Place the FLIGHT STATION emergency manual temperature control switch in the "OFF" position.

k. Man A: Hold the CABIN temperature control switch in "WARM".
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become warm and the airflow from the cabin outlets should become hot. On Aircraft 5093, 5097 and subsequent and those having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the temperature of the flight station air should not change.

l. Man A: With the CABIN temperature control switch still held in "WARM", turn the FLIGHT STATION and CABIN selectors to "HOT", then to "COLD", and then to the mid-range (12 o'clock) position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Air temperature at the flight station and cabin outlets should not change at any setting of the selectors.

m. Man A: With the CABIN temperature control switch still held in "WARM", hold the FLIGHT STATION temperature control switch in "WARM".
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Air temperature at the flight station outlets should become hot.

n. Man A: With the CABIN and FLIGHT STATION temperature control switches still held in "WARM", turn the FLIGHT STATION and CABIN temperature selectors to "HOT", then to "COLD", and then to the mid-range (12 o'clock) position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: The temperature of the air at the flight station and cabin outlets should not change at any setting of the selectors.

Note

The next step requires exposing the cabin temperature sensor, and, for airplanes 5001 through 5085 not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, the flight station temperature sensor to the temperature of air entering these compartments through the ducts. This may be done by permitting the temperature of each compartment to adjust to the duct temperature, which can take several minutes; or by detaching each temperature sensor from its mounting and reconnecting it electrically to the control system with a suitable patch cable harness, so that it can be held in the airstream from the outlet. If the first method is used, and the sensors are left installed, adjust the pressurization controls in accordance with section 21.1.2.1. This is to assure that the airplane will not be pressurized and close the entrance and flight station doors. This assures a true temperature sampling by the sensors, with no dilution of compartment air by outside air or by air from the other compartment of the airplane.

o. Man A: Hold both the CABIN and the FLIGHT STATION temperature control switches in the "COOL" position for approximately one (1) minute and permit flight station and cabin air to become cool. When the air is cool, place the FLIGHT STATION temperature control switch in the "AUTO" position and the CABIN temperature control switch in "HOLD".

p. *Man B: Expose the flight station sensor to the airstream from a flight station outlet. Check the changing temperature of air from the flight station outlets.
RESULT: The air from the flight station outlets should become warmer.

q. *Man B: Remove the flight station sensor from the airstream.

r. Man A: Turn the FLIGHT STATION temperature selector to "HOT".
Man B: Check the temperature of the air being supplied by the flight station outlets.
RESULT: The airflow from the flight station outlets should become warmer.

s. *Man B: While warm air is flowing from the flight station outlets, expose the flight station temperature sensor to the airstream from a flight station outlet.

t. Man A: Turn the FLIGHT STATION selector to the mid-range (12 o'clock) position.
Man B: Check the flight station air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become cooler.

u. *Man B: Remove the flight station sensor from the outlet airstream.

v. Man A: Turn the FLIGHT STATION temperature selector to the "COLD" position.
Man B: Check the flight station air outlets for the temperature of the air being supplied.
RESULT: The airflow from the flight station outlets should become cooler.

21.0.3.3 (Cont).

Note

The airflow temperature in the cabin should not change during operation of the flight station temperature controls.

w. **Man A: Place the FLIGHT STATION temperature control switch in the "HOLD" position. Hold the CABIN emergency manual temperature control switch in the "WARM" position long enough to determine a change in air temperature; then release to the "HOLD" position.

CAUTION

The cabin high limit thermostat is inoperative during this portion of checkout. Do not permit ducts to overheat.

Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.

RESULT: Airflow from all outlets should become warmer. On Aircraft 5093, 5097 and subsequent and those having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the temperature of the flight station air should not change.

x. **Man A: Hold the CABIN emergency manual temperature control switch in the "COOL" position for approximately one (1) minute; then release to the "HOLD" position.
Man B: Check the flight station and cabin air outlets for the temperature of the air being supplied.
RESULT: Air flow from all outlets should become cooler. On Aircraft 5093, 5097 and subsequent and those having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the temperature of the flight station air should not change.

y. **Man A: Place the CABIN emergency manual temperature control switch in the "OFF" position.

z. Man A: Place the CABIN temperature control switch in "AUTO", and turn the CABIN temperature selector to the mid-range (12 o'clock position.
Man B: Expose the cabin sensor to the airstream from a cabin outlet and check the temperature of the air coming from the flight station and cabin outlets.
RESULT: Air flow from all outlets should become warmer. The cabin temperature indicator on the air conditioning control panel should show an increase in temperature.

aa. Man B: Remove the cabin sensor from the outlet airstream, check the temperature of the airflow at both the cabin and flight station outlets.
RESULT: The airflow from all outlets should become warmer. On Aircraft 5093, 5097 and subsequent and those having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the temperature of the flight station air should not change.

ab. Man A: Turn the CABIN temperature selector to "HOT".
Man B: Check the temperature of the airflow at the flight station and cabin outlets.
RESULT: The airflow from the flight station and cabin outlets should become warmer. On Aircraft 5093, 5097 and subsequent and those having Service Bulletin 329-213, Service Bulletin 329-234, or Service Bulletin 329-235 incorporated, the temperature of the flight station air should not change.

ac. Man A: Place both the FLIGHT STATION and CABIN temperature control switches in "AUTO", temperature selectors to the mid-range (12 o'clock) position.
Man B: Check the temperature of the airflow at the flight station and cabin outlets.
RESULT: The airflow from the flight station and cabin outlets should become cooler.

ad. Man A: Adjust the pilot's and copilot's foot warmer valves, and check the outflow from them.
RESULT: Adjustment of the valves should vary the airflow from the foot warmers.

ae. Man A: Check for free flow of air from both crew gasper valves and both crew body warmer valves.
Man B: Check for free flow of air from all cabin gasper valves, overhead outlets, and floor outlets.
RESULT: Air should flow freely from all gasper valves and outlets.

af. Man A: With air flowing freely from all outlets, place the FLT STA AIR COND SHUT-OFF switch, on the bleed air control panel, in the "SHUTOFF" position.
Man B: Check the amount of airflow from the flight station outlets.
RESULT: Airflow from the flight station outlets should decrease, but should not cease.

ag. Man A: Return the FLT STA AIR COND SHUT-OFF switch to the "NORMAL" position.
Man B: Check the amount of airflow from the flight station outlets.
RESULT: Airflow from the flight station outlets should increase.

ah. Man A: Place the CABIN AIR COND SHUT-OFF switch on the bleed air control panel, in the "SHUTOFF" position.
Man B: Check the amount of airflow from the cabin outlets, and the temperature of the air coming from the flight station outlets.
RESULT: Airflow from the cabin outlets should cease. Airflow from the flight station outlets should decrease and become hot or cease completely.

ai. Man A: Place the CABIN AIR COND. SHUT-OFF switch in the "NORMAL" position.
Man B: Check the airflow from the flight station and cabin outlets.
RESULT: Airflow should resume from all outlets.

aj. Man A: Place the NORMAL-DEPRESS switch in the "DEPRESS" condition.
Man B: Check airflow from the flight station and cabin outlets.
RESULT: All airflow should stop.

ak. Man A: Return the NORMAL-DEPRESS switch to the "NORMAL" position.
Man B: Check airflow from the flight station and cabin outlets.
RESULT: Airflow should resume from all outlets.

al. Shut down the air conditioning system and the operating engines. Reinstall the flight station and cabin temperature sensors, if they were removed from their mountings for the checkout.

21.0.4 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO AIR FLOWS FROM OUTLETS WHEN SYSTEM IS TURNED ON.		
If engine is being used to operate system, bleed air valve defective or receiving power.	Check bleed air valve input voltage and valve position. If bleed air valve is functioning satisfactorily, check flow control valve position and input voltage.	If valve is closed and not receiving power, replace valve. If valve is receiving power, repair circuit.
If ground air supply is being used to operate system, flow control valve defective or receiving power.	Check flow control valve position and input voltage.	If valve is closed and not receiving power, replace valve. If valve is receiving power, repair circuit.
WARM AIR FLOWS FROM OUTLETS WHEN COOLING HAS BEEN SELECTED.		
Temperature control box defective or not receiving power.	Check temperature control box input voltage and power supply, including circuit breakers.	Replace temperature control box if electrical power supply to box is satisfactory. If box is not receiving power, repair circuit as required.
Refrigeration unit bypass valve stuck in open position.	Check valve position (external pointer).	Replace refrigeration unit bypass valve.
Sensor for affected compartment defective or not receiving power.	Check sensor power supply.	Replace sensor if power supply is satisfactory.
Duct high-limit thermostat defective.	Perform continuity check of thermostat and control switch circuits.	Repair wiring or replace units as required.
Seized refrigeration unit rotative assembly.	Check air flow at fan exhaust ports both sides of airplane.	Replace rotative assembly.
Temperature control selector defective.	Perform continuity check of temperature selector.	Replace temperature selector if found defective. Replace temperature control box if selector is satisfactory.
If warm air flows only from flight station outlets, flight station temperature control valve or temperature control box is defective	Check input voltage to flight station temperature control valve.	Replace valve if 28 volts, DC available at "cool" winding. Replace temperature control box if 28 volts, DC not available at "cool" winding.
If warm air flows only from cabin outlets, refrigeration unit bypass valve or temperature control box defective.	Check input voltage to refrigeration unit bypass valve.	Replace valve if 28 volts DC available at "cool" winding. Replace temperature control box if 28 volts DC not available at "cool" winding.
If warm air flows only from cabin outlets on left hand side, emergency pressurization shutoff valve leaking.	Check position of butterfly and input voltage.	Close butterfly if open. Replace valve if leaking. If receiving power, repair circuit as required.

21.0.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
OUTLET AIR TEMPERATURE FLUCTUATES WITH CONTROL IN "AUTO".		
Temperature anticipator or duct sensor defective or installed incorrectly.	Check that anticipator or duct sensor is installed with arrow on mounting flange pointing in direction specified in Section 21.0.5.10.	Reinstall anticipator or duct sensor in correct position if necessary. Replace anticipator or sensor if it was installed correctly.
Temperature control box defective.	Check control box input voltage and output voltage.	Replace box if defective.
Air flow control valve malfunctioning (fluctuates).	Check that butterfly position changes with variation in inlet pressure to valve.	Replace valve if defective.
REDUCTION IN AIRFLOW DURING OPERATION.		
Icing of water separator.	Check condition of low-limit thermostat and circuits and operation of turbine bypass valve.	Replace low-limit thermostat, repair circuit or replace turbine bypass valve, as necessary.
Damaged ducts or leaking duct connections.	Check ducts and connections, beginning with readily accessible ducts in aft compartment.	Repair or replace duct or reinstall connections as necessary.
AIR TEMPERATURE DOES NOT INCREASE WHEN "AUTO" IS SELECTED, AFTER PROLONGED OPERATION IN "COOL".		
Temperature control switch or temperature control box defective.	Perform continuity check of temperature control switch.	Replace temperature control switch if defective. If switch is not defective, replace temperature control box.
AIR TEMPERATURE DOES NOT CHANGE AS DESIRED WHEN TEMPERATURE CONTROL SELECTOR IS TURNED TO "COLD" OR "HOT" POSITION.		
Temperature control selector or temperature control box defective.	Perform continuity check of temperature control selector.	Replace selector if it is found defective. If selector is not defective, replace temperature control box.
CABIN TEMPERATURE INDICATOR READINGS NOT INCREASING OR DECREASING WITH ADJUSTMENTS TO CABIN TEMPERATURE.		
Indicator power supply interrupted, or indicator or sensor defective.	Check power supply (circuit breaker).	If power supply is satisfactory, replace indicator or sensor as necessary.

21.0.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FOG AND/OR WATER DROPLETS DISCHARGED FROM AIR OUTLETS IN CABIN.		
Icing of water separator.	Check condition of low-limit thermostat and circuits and turbine bypass valve.	Replace low-limit thermostat, repair circuit or replace turbine bypass valve.
Obstruction in water separator drain line or drain filter clogged.	Remove drain line fitting and check line and filter.	Remove obstruction, replace tube or clean filter.
Water separator condenser section damaged.	Remove water separator, remove band coupling, and check condenser.	Replace water separator.

21.0.5 REMOVAL AND INSTALLATION.

21.0.5.1 REMOVAL OF AIR CONDITIONING DUCTS.

Removal of the ducts, after access has been gained by removing trim or furnishings as applicable, consists of disassembly of joint clamps, sleeves, and mounting provisions, and subsequent removal of the affected ducts. It may sometimes be necessary to loosen the duct mountings on each end of the one to be removed to obtain clearance for removal without damaging duct ends. Use caution when removing ducts between which check valves are installed to prevent damage to the valve. Never use force to separate a duct from adjacent ducts; damage to the ends may create leaks causing system malfunctions or, in the case of some ducts, escape of hot bleed air, with resultant overheating on surrounding parts. Replace any cracked ducts with like serviceable parts. Stainless steel or aluminum ducts which are dented and do not have a sharp bend or crease at the dent may be straightened by using water pressure to return them to normal configuration. Any duct showing a crease or deformation after straightening should be replaced. Insulation should be inspected and repaired or replaced, if damaged.

21.0.5.2 INSTALLATION OF AIR CONDITIONING DUCTS.

a. Align the duct ends as accurately as possible with those of adjacent connecting ducts or flanges of units to which the duct is to be connected, before installing the joint sleeves or clamps. Where check valves are installed between ends of ducts, use caution to align check valves so adequate clearance exists for free swing of the flappers, and observe that installation of the check valve with respect to direction of flow is correct. Position check valves so the alignment lugs on the valve flange align with the slots in the end of the ducts where applicable. Installation of each bleed air check valve, located in the engine pylon at B.L. 47.85, requires replacement of two gaskets (Vendor P/N 24096-200-C).

b. Check that at any duct joint the clearance between the flanges at any point is not more than 0.015 inch.

c. Check that the force required to align a duct with the connecting duct or unit flange does not exceed 3 pounds, not including duct weight.

CAUTION

Be particularly careful not to exceed the above duct preload limit when connecting ducts to the refrigeration units.

d. If the conditions in steps (b) and (c) have been satisfied, install the connecting sleeves and couplings, observing the torque values given in table of AIR CONDITIONING SYSTEM COUPLING TORQUE VALUES. If the conditions cannot be satisfied, adjust the mounting provisions until satisfactory alignment has been obtained; then install the connecting sleeves and couplings.

e. For all JQ-461-series couplings, tighten the coupling screws alternately until hand-tight; then tighten each screw an additional half-turn.

f. Tighten all other couplings to the torque value specified below, applied at the T-bolt nut; lightly tap around the outside of the coupling with a soft-faced mallet to seat it on the flanges. Reapply the same torque to the T-bolt nut.

AIR CONDITIONING SYSTEM COUPLING TORQUE VALUES

COUPLING P/N	APPLICATION	TORQUE VALUE (INCH-POUNDS)
LS 1561-46 or 21693C287R	When used with LS 1611A11 hose, LS35031A11 hose, STSPH200B11 hose, or AD777B11 hose	35
18276-100 or 18276A100	All locations	45
18276-150 or 18276A150	On aluminum ducts	45
18276-200 or 18276A200	On fiberglass and aluminum ducts	45
18276-200 or 18276A200	On steel ducts	85
18276-250 or 18276A250	All locations	110
18276-300 or 18276A300	All locations	130
4480-356-SH	On fiberglass ducts	30
4480-356-SH	On steel ducts	85
43907-532-S	All locations	40
4483-581-SH	On aluminum ducts	50
54722-200	On steel ducts	125

21.0.5.3 REMOVAL OF WATER SEPARATOR.

a. Disconnect the ducts from the front and rear ends of the water separator by loosening the couplings and sliding the hose clear of the bosses on the separator.

b. Disconnect the water drain tube from the separator.

c. Disassemble the hardware that attaches the separator to the fuselage intercostal member, and lower the separator. Remove the separator from the airplane.

21.0.5.4 INSTALLATION OF WATER SEPARATOR.

a. Position the separator below the supporting intercostal member. Attach it to the member with the hardware previously removed.

b. Connect the water drain line to the separator.

c. Connect the ducts at each end of the water separator with hose and couplings. Tighten the coupling screws alternately until hand-tight; then tighten each screw another half-turn.

21.0.5.5 REMOVAL OF REFRIGERATION UNIT.

a. Disconnect the turbine bypass, cold air, and ram air overboard ducts from the aft end of the unit by loosening the attaching clamps and moving the sleeves or gaskets as required.

b. Disassemble the hardware securing the forward flange of the unit to the ram air duct.

c. With the unit temporarily supported, disassemble the screws and nuts that secure it to the supporting structure. Carefully disengage the unit from the ducts, avoiding any forcing. Loosen duct support clamps as necessary to accomplish this. Remove the unit from the airplane.

Note

If the unit is to be shipped or stored it should first be drained by tilting the unit as required to drain as much oil as possible by gravity flow. Turn unit until remainder of the oil collects in one corner of sump and remove with a suitable syringe.

21.0.5.5 (Cont).

CAUTION

Do not attempt to remove oil by disassembling the sump from the unit and pouring the oil from the sump. When the sump is reassembled to the unit, it is possible the lubrication wicks may be cut, causing subsequent oil starvation of the bearings.

Note

See paragraph 21.0.5.17 for removal instructions for rotative assembly only.

21.0.5.6 INSTALLATION OF REFRIGERATION UNIT.

a. Position the unit on the supporting structure, and secure it with the screws and nuts previously removed.

b. Align the turbine bypass, cold air, and ram air overboard ducts with the respective ports on the unit.

c. Install gaskets and a sleeve, as applicable, and secure the ducts to the unit with couplings and clamps, according to the locations from which these were removed. Tighten the clamps and couplings in accordance with paragraph 21.0.5.2 as applicable.

d. Install a serviceable gasket between the forward (square) flange of the refrigeration unit and the mating flange of the transition duct. Connect the two flanges with the hardware previously removed.

e. Check the refrigeration unit for correct servicing in accordance with Chapter 12.

Note

See paragraph 21.0.5.18 for installation instructions for rotative assembly only.

21.0.5.7 REMOVAL OF SHUTOFF VALVES.

a. Disconnect the electrical harness at the connector on the valve.

b. Disassemble the hardware that secures the valve flange to the adjacent parts, or the couplings that connect the valve to adjacent ducts, whichever is applicable.

c. Remove the valve from the airplane.

21.0.5.8 INSTALLATION OF SHUTOFF VALVES.

a. Position the valve between the duct flanges, or between the duct flange and the port flange, as applicable, with the valve facing in the correct direction. Note flow arrow on valve body.

b. Connect the valve to the mating flanges with the couplings and/or hardware previously removed. Tighten the couplings in accordance with paragraph 21.0.5.2. Connect the electrical harness to the valve.

21.0.5.9 REMOVAL OF ANTICIPATORS. Disconnect electrical connector and remove four mounting screws. Retain hardware for reuse if serviceable. Discard gasket.

21.0.5.10 INSTALLATION OF ANTICIPATORS. The new anticipator should be equipped with a new gasket. Position the anticipator with the arrow on mounting base pointing in the direction of air flow. Flow is forward for the anticipator in the flight station supply duct at approximately fuselage station 510 right-hand side in the cabin. Flow is toward the right-hand side of the airplane in the duct which contains the cabin anticipator located at approximately fuselage station 580 in the aft compartment.

21.0.5.11 REMOVAL OF LOW-LIMIT THERMOSTAT. Remove the low-limit thermostat located at approximately fuselage station 620, water line 128, in the aft compartment by disconnecting the electrical connector and removing the four attaching screws. Discard the gasket but retain the hardware for reuse if serviceable.

21.0.5.12 INSTALLATION OF LOW-LIMIT THERMOSTAT. The new thermostat should be equipped with a new gasket. Install the low-limit thermostat by inserting the sensing element in the hole provided in the duct (see paragraph 21.0.5.11) and attach with four screws. Make sure that gasket is installed. Install the electrical connector.

21.0.5.13 REMOVAL OF HIGH-LIMIT THERMOSTAT. The flight station high-limit thermostat is located in the flight station supply duct on right hand side of the airplane at approximately fuselage station 503, water line 95. The cabin high-limit thermostat is located in the right hand branch of the cabin supply duct at approximately fuselage station 579, water line 105 on airplanes 5001 thru 5045. On airplanes 5046 and up and on previous airplanes on which Service Bulletin 329-106 has been incorporated, it is in a similar location in the left-hand branch of the supply duct. The high-limit thermostats are retained in the bosses on ducts by means of a pipe thread on the thermostat housing. Removal is accomplished by disconnecting the electrical leads, removing the safety wire, and then rotating the thermostat until the threads are disengaged.

CAUTION

Use wrenches on the hex portion only to prevent damage to internal switch and housing.

21.0.5.14 INSTALLATION OF HIGH-LIMIT THERMOSTAT. Install the high-limit thermostats by rotating the threaded portion into the threaded bosses provided on the ducts. Prior to installing the thermostat, brush all threads on the thermostat, except the first two, with petrolatum conforming to Specification VV-P-236, or equivalent. Then screw thermostat into boss by hand or light wrench torque. Back thermostat out of boss and apply anti-seize compound conforming to Specification TT-S-1732, or equivalent, to all threads except the first two, and reinstall and tighten to 75 to 100 inch pounds. Use wrench on hex portion only to prevent damage to internal switch and housing. Install electrical leads and safety wire. Use 0.030-inch diameter, corrosion resistant steel safety wire.

Note

On airplanes 5001 thru 5085 high limit thermostat Part No. FYLD 7111-2 is the preferred replacement for Part No. FYLD 7111-1. The terminals are numbered differently on these units. To eliminate confusion, the correct wiring relationship is as follows:

FLYD 7111-1	FLYD 7111-2
A	3
B	2
C	1

21.0.5.15 REMOVAL OF TEMPERATURE SENSOR.

a. The flight station sensor on airplanes 5001 through 5085, not having Service Bulletin 329-213 or Service Bulletin 329-235 incorporated, is located at approximately fuselage station 228, on the right-hand side, behind the copilot's head. The cabin sensor location is a function of the custom interior installation. It varies from airplane to airplane, but is generally in the area immediately forward of fuselage station 490, on the right-hand side of cabin. The flight station sensor is installed with five screws, grommets and gaskets. It is necessary to remove the sensor cover to gain access to the screws. Disconnect the electrical connector from the sensor, then remove the five attaching screws. Retain the gaskets and hardware for reuse if serviceable.

b. The cabin sensor installation is similar to the flight station installation in that five screws are used to retain the sensor to the mounting bracket. The gasket and grommet configuration may, however, be different.

Note

It will be necessary to note the arrangement of gaskets and/or grommets on this installation for use on reinstallation.

Retain hardware, grommets, gaskets, etc., for reuse if serviceable.

21.0.5.16 INSTALLATION OF TEMPERATURE SENSOR. Remove cover from unit for access to attaching holes. Prior to installing the flight station unit, ascertain that the MS 35489-3X grommets are installed in the screw holes in the trim panel. Install sensor with the gasket between the mounting surface and base of sensor. Install screws and tighten. The gasket thickness after it is compressed by tightening the screws should not be less than .18 inch. Installation of the cabin sensor is similar. Gasket and grommet configuration should be restored to the configuration which existed prior to removal.

21.0.5.17 REMOVAL OF REFRIGERATION UNIT ROTATIVE ASSEMBLY (TURBINE AND FAN ASSEMBLY).

a. Remove the two (2) bolts and washers at the turbine inlet-bleed air duct junction. Discard the preformed packing.

b. Disconnect the cold air and the ram air overboard ducts from the aft end of the unit by loosening the attaching clamps and moving the sleeves or gaskets as required.

c. With the unit temporarily supported, decouple the clamp between the turbine fan assembly and the heat exchanger. Discard the preformed packing.

d. Carefully disengage the unit from the ducts, avoiding any forcing. Loosen duct support clamps as necessary to accomplish this. Remove the unit from the airplane, being careful not to invert the assembly.

Note

If the unit is to be shipped or stored, it should first be drained. Tilt the unit as required to drain as much oil as possible by gravity flow. Turn the unit until the remainder of the oil collects in one corner of the sump, and remove with a suitable syringe.

CAUTION

Do not attempt to remove oil by disassembling the sump from the unit and pouring the oil from the sump. When the sump is reassembled to the unit, it is possible that the lubrication wicks may be cut, causing subsequent oil starvation of the bearings.

21.0.5.18 INSTALLATION OF REFRIGERATION UNIT ROTATIVE ASSEMBLY (TURBINE AND FAN ASSEMBLY).

a. Position unit by aligning with heat exchanger flange, cold air and ram air overboard ducts, and bleed air duct.

21.0.5.18 (Cont).

b. Install new preformed packing (vendor P/N 69490B247) between the turbine fan and the heat exchanger. Install the clamp and tighten to 40 inch pounds torque value using the general instructions in paragraph 21.0.5.2.

c. Install gaskets and a sleeve, as applicable, and secure the ducts to the unit with couplings and clamps, according to the locations from which these were removed. Tighten the clamps and coupling in accordance with paragraph 21.0.5.2 as applicable.

d. Install a new preformed packing (vendor P/N 69490B218) at the junction between the turbine inlet and the bleed air duct. Install two (2) screws and washers in the positions from which they were previously removed.

e. Check the rotative assembly for correct servicing in accordance with Chapter 12.

21.0.6 SERVICING.

21.0.6.1 ADJUSTMENT. The units comprising the air conditioning system should not be adjusted as a function of line maintenance. Those units incorporating adjustment provisions are calibrated during manufacture or overhaul using specialized test equipment. A unit causing system malfunction should be isolated, using the troubleshooting procedure in paragraph 21.0.4, and replaced with a serviceable unit.

21.0.6.2 LUBRICATION. The refrigeration unit expansion turbine is the only component in the air conditioning system that requires periodic lubrication. The oil level in the oil reservoir should be checked prior to each flight, but need not be checked more than once a day. The oil supply should normally provide for 500 hours of operation without replenishment; however, oil should be added if necessary to maintain level within the limits marked on the reservoir. Refer to Chapter 12 for additional servicing instructions.

21.0.6.3 CLEANING.

WARNING

Provide adequate ventilation when using cleaning solvent (Federal Specification P-D-680, Type II). Avoid prolonged breathing of vapors and minimize skin contact.

The drain line filter in the water separator (see figure 21-6) should be inspected and cleaned periodically. The required minimum operating time interval between inspections will vary according to the moisture content and cleanliness of the air passing through the water separator. The interval should therefore be established by the operator to be compatible with the conditions under which the airplane is operated. It is recommended that the filter be inspected upon completion of 25 hours of operation after delivery and at each 50-hour interval thereafter until it is determined that some other interval is acceptable. If required, clean the filter screen with dry cleaning solvent (Federal Specification P-D-680, Type II or equivalent) and dry with compressed air. Inspect drain fitting orifice for foreign matter. Replace the MS28778-12 gasket with a new one on reassembly. The bleed air and conditioned air supply ducts should require no cleaning for normal operation. If, however, the air supplied to the cabin and flight station appears to be excessively dirty, the left and right turbine cold air discharge ducts should be removed, inspected and cleaned if necessary. No further inspection of ducts is warranted if these ducts are not excessively dirty.

21.0.6.3.1 WATER SEPARATOR COALESCER. The coalescer in the water separator should be cleaned every 100 hours or as needed. The coalescer may be removed and hand washed in a mild soap solution, thoroughly rinsed, and reinstalled wet to ease installation. As an alternative method, the coalescer may be cleaned without removing it from the support assembly by cleaning with dry cleaning solvent per Federal Specification P-D-680, Type II. A new coalescer should be installed if the old one is damaged or frayed.

21.0.6.3.2 ENGINE BLEED AIR SHUTOFF VALVE FILTER. Inspect and clean the filter in each bleed air shutoff valve every 500 hours.

Note

On airplanes modified by 731 Mod with Garrett engines installed, filters should be cleaned every 250 hours.

a. Remove and inspect filter vent plug for stripped threads or other damage.

WARNING

Provide adequate ventilation when using cleaning solvent. Avoid prolonged breathing of vapor and minimize skin contact.

b. Remove and ultrasonically clean filter using instructions of ultrasonic equipment manufacturer. The pressure drop for a new filter is 25 inches of water at a flow rate of 0.60 SCFM (venting to atmosphere). The maximum pressure drop allowable for a cleaned filter is 50 inches of water. If the pressure drop exceeds 50 inches of water, the filter should be replaced.

c. Alternate cleaning method: Wash in cleaning solvent, Federal Specification P-D-680 using a soft-bristle brush. Dry thoroughly with filtered compressed air, and cover or wrap with clean paper until reinstallation.

d. Reinstall filter and plug using new O-rings.

21.0.6.3.3 BLEED FLOW CONTROL VALVE FILTER. Inspect and clean the bleed flow control valve filter every 500 hours.

Note

If a Pneu-Tech Laboratories flow control valve is installed, it will be necessary to disconnect the sense line from the plug prior to plug removal. If an AiResearch flow control valve is installed, it will be necessary to remove the regulator assembly to gain access to the filter.

a. Remove and inspect filter vent plug for stripped threads or other damage.

WARNING

Provide adequate ventilation when using cleaning solvent. Avoid prolonged breathing of vapor and minimize skin contact.

b. Remove and clean filter using cleaning solvent, Federal Specification P-D-680, Type II, and a soft bristle brush.

c. Dry filter thoroughly with filtered compressed air and cover with clean paper or acetate until reinstallation.

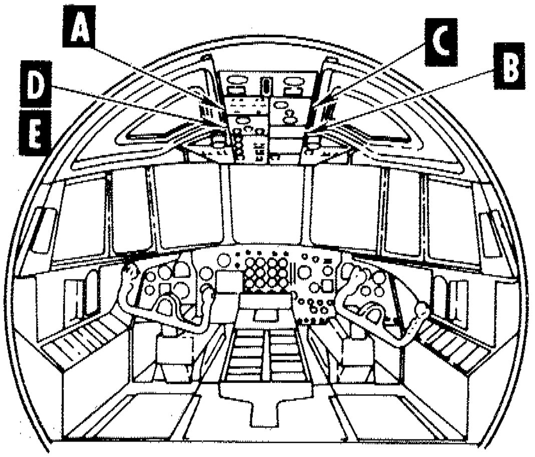
d. Reinstall filter and plug using new O-ring. Reconnect sense line on Pneu-Tech Laboratories units.

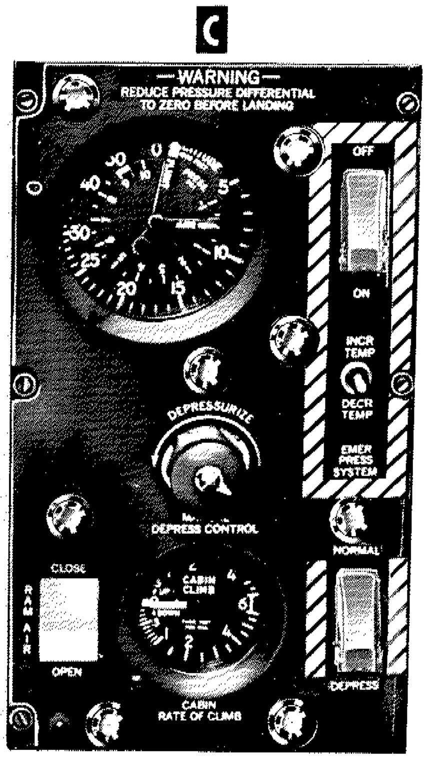
21.0.6.4 SYSTEM PRESERVATION.

21.0.6.4.1 NORMAL OPERATION. During normal flight operation and preflight and post-flight ground operation, no special considerations are required to obtain maximum service life from the components or the system. To prevent unnecessary operation of the system valves during periods of time when electrical power is available on the airplane and engines are not to be operated, or bleed air is off, the temperature control toggle switches should be placed in the "HOLD" position to prevent electrical power from reaching the temperature control box. The cabin and flight station sensors should also be deactivated by disconnecting the electrical connector or by opening the air conditioning system circuit breakers. No special considerations are required for operation in extreme environments (temperature, humidity and contaminated air) except as noted in this section.

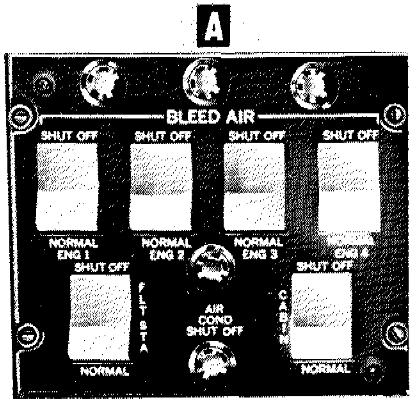
21.0.6.4.2 AIRPLANE STORAGE. When the airplane is to be stored outside for periods in excess of 30 days the ram air inlets in the leading edge of each engine nacelle support pylon and the turbine fan exhaust openings on each side of the fuselage at approximately fuselage station 654, water line 94, should be covered with standard covers, if available, or with waterproof paper to prevent entry of water, dust, birds, etc. (See figure 21-3).

21.0.6.5 (Deleted.)

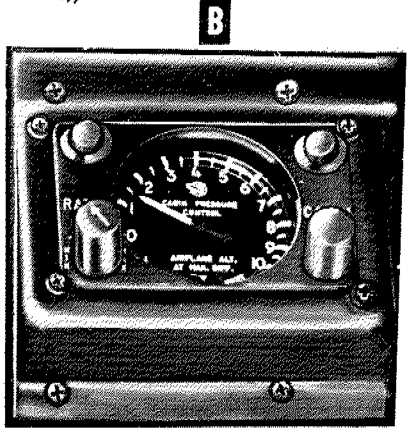




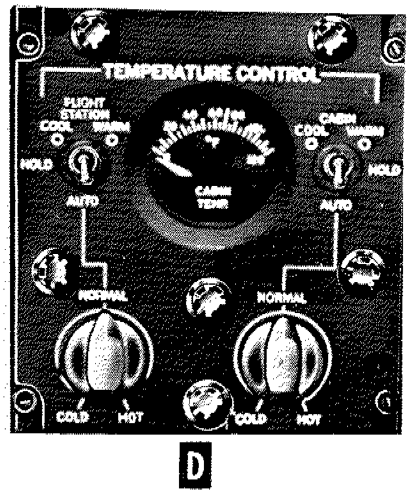
PRESSURIZATION CONTROL PANEL



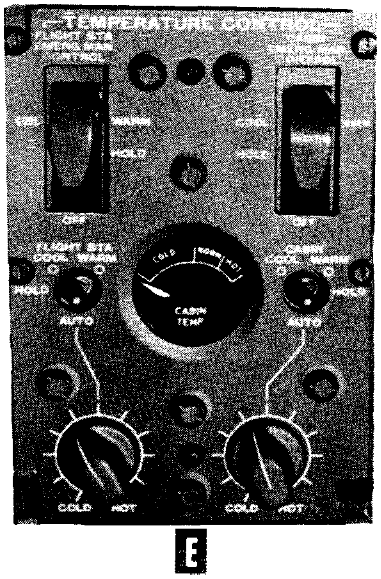
BLEED AIR CONTROL PANEL



CABIN PRESSURE CONTROLLER



AIR CONDITIONING CONTROL PANEL
(AIRPLANES 5001 THRU 5085, NOT HAVING SERVICE BULLETIN 329–213 OR SERVICE BULLETIN 329–235 INCORPORATED)



AIR CONDITIONING CONTROL PANEL
(AIRPLANES 5086 AND UP, AND THOSE HAVING SERVICE BULLETIN 329–213 OR SERVICE BULLETIN 329–235 INCORPORATED)

Figure 21-1. Air Conditioning and Pressurization Systems Components (Sheet 1 of 2)

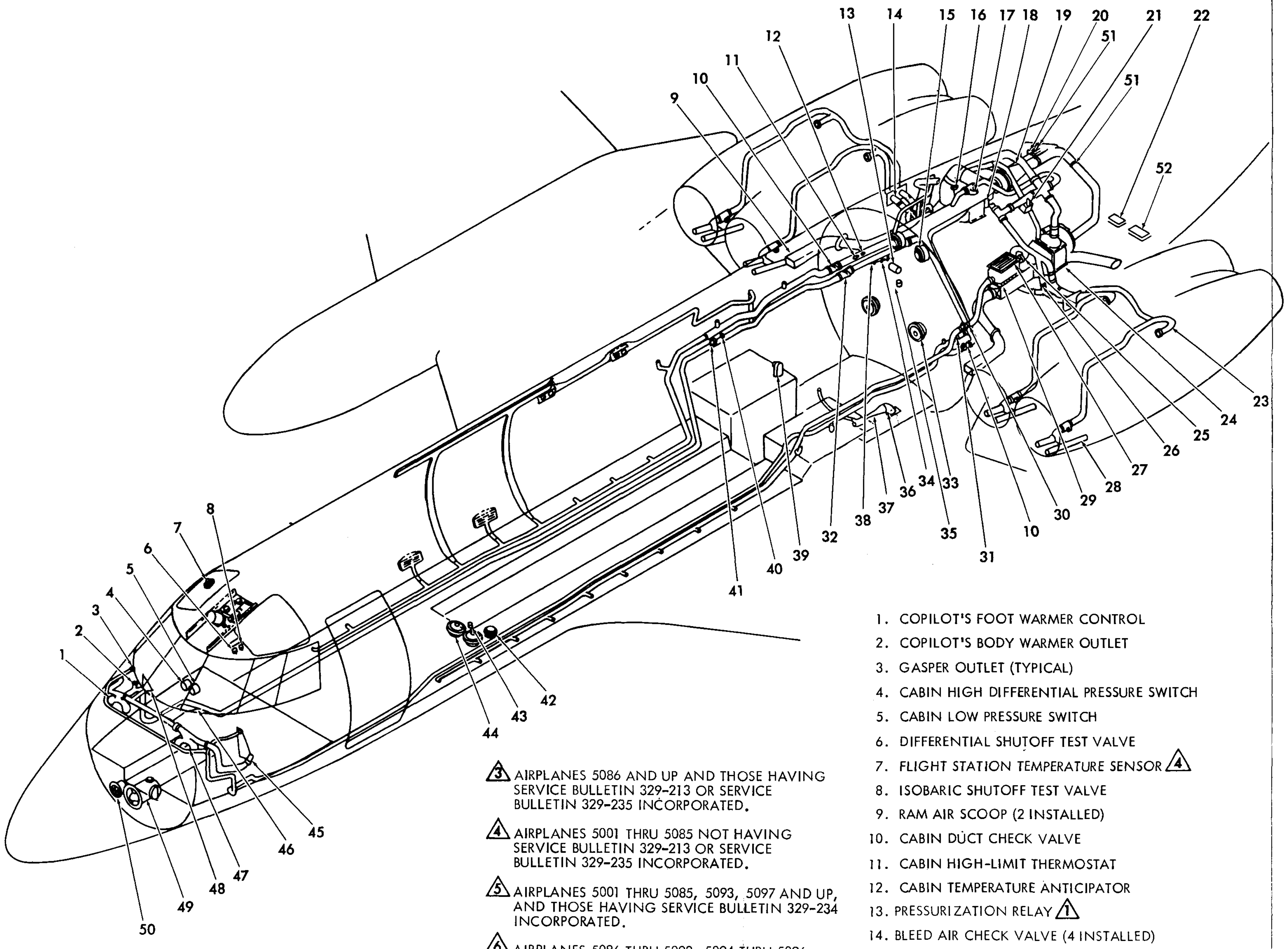


Figure 21-1. Air Conditioning and Pressurization Systems Components (Sheet 2 of 2)

NOTE

△1 AIRPLANES 5046 AND UP AND THOSE HAVING SERVICE BULLETIN 329-61 INCORPORATED.

△2 AIRPLANES 5001 THRU 5045 NOT HAVING SERVICE BULLETIN 329-61 INCORPORATED.

△3 AIRPLANES 5086 AND UP AND THOSE HAVING SERVICE BULLETIN 329-213 OR SERVICE BULLETIN 329-235 INCORPORATED.

△4 AIRPLANES 5001 THRU 5085 NOT HAVING SERVICE BULLETIN 329-213 OR SERVICE BULLETIN 329-235 INCORPORATED.

△5 AIRPLANES 5001 THRU 5085, 5093, 5097 AND UP, AND THOSE HAVING SERVICE BULLETIN 329-234 INCORPORATED.

△6 AIRPLANES 5086 THRU 5092, 5094 THRU 5096 NOT HAVING SERVICE BULLETIN 329-234 INCORPORATED.

△7 AIRPLANES 5093, 5097 AND UP, AND THOSE HAVING SERVICE BULLETIN 329-213, SERVICE BULLETIN 329-234 OR SERVICE BULLETIN 329-235 INCORPORATED.

1. COPILOT'S FOOT WARMER CONTROL
2. COPILOT'S BODY WARMER OUTLET
3. GASPER OUTLET (TYPICAL)
4. CABIN HIGH DIFFERENTIAL PRESSURE SWITCH
5. CABIN LOW PRESSURE SWITCH
6. DIFFERENTIAL SHUTOFF TEST VALVE
7. FLIGHT STATION TEMPERATURE SENSOR △4
8. ISOBARIC SHUTOFF TEST VALVE
9. RAM AIR SCOOP (2 INSTALLED)
10. CABIN DUCT CHECK VALVE
11. CABIN HIGH-LIMIT THERMOSTAT
12. CABIN TEMPERATURE ANTICIPATOR
13. PRESSURIZATION RELAY △1
14. BLEED AIR CHECK VALVE (4 INSTALLED)
15. SAFETY VALVE (2 INSTALLED)
16. RAM AIR SHUTOFF VALVE
17. REFRIGERATION UNIT BYPASS VALVE
18. LOW LIMIT SENSOR △5
19. WATER SEPARATOR
20. TURBINE BYPASS VALVE
21. BLEED AIR FLOW CONTROL VALVE
22. TEMPERATURE CONTROL BOX
23. BLEED AIR DUCT
24. REFRIGERATION UNIT (2 INSTALLED)
25. EMERGENCY PRESSURIZATION SHUTOFF VALVE
26. ENGINE BLEED AIR SHUTOFF VALVE (4 INSTALLED)
27. EMERGENCY PRESSURIZATION HEAT EXCHANGER
28. ENGINE INLET ANTI-ICING DUCT
29. EMERGENCY PRESSURIZATION RAM AIR CONTROL
30. EMERGENCY PRESSURIZATION DUCT CHECK VALVE
31. GASPER DUCT CHECK VALVE
32. FLIGHT STATION DUCT CHECK VALVE
33. OUTFLOW VALVE (2 INSTALLED) △1
34. FLIGHT STATION AIR CONDITIONING SHUTOFF VALVE
35. JET PUMP △1
36. CHECK VALVE
37. TOILET EXHAUST VENTURI
38. FLIGHT STATION TEMPERATURE CONTROL VALVE
39. CABIN TEMPERATURE SENSOR
40. FLIGHT STATION TEMPERATURE ANTICIPATOR △4 FLIGHT STATION DUCT SENSOR △3
41. FLIGHT STATION HIGH-LIMIT THERMOSTAT
42. PRESSURIZATION RELAY △2
43. JET PUMP △2
44. OUTFLOW VALVE (2 INSTALLED) △2
45. PILOT'S BODY WARMER OUTLET
46. PILOT'S WINDSHIELD DE-FOGGER
47. PILOT'S FOOT WARMER CONTROL
48. COPILOT'S WINDSHIELD DE-FOGGER
49. FAN AND FLOW REGULATOR, ELECTRONIC COMPARTMENT
50. FLOW LIMITER VALVE
51. ANTI ICING SCREEN ASSEMBLY (IN DUCT) △6
52. LOW-LIMIT TEMPERATURE CONTROL BOX △7

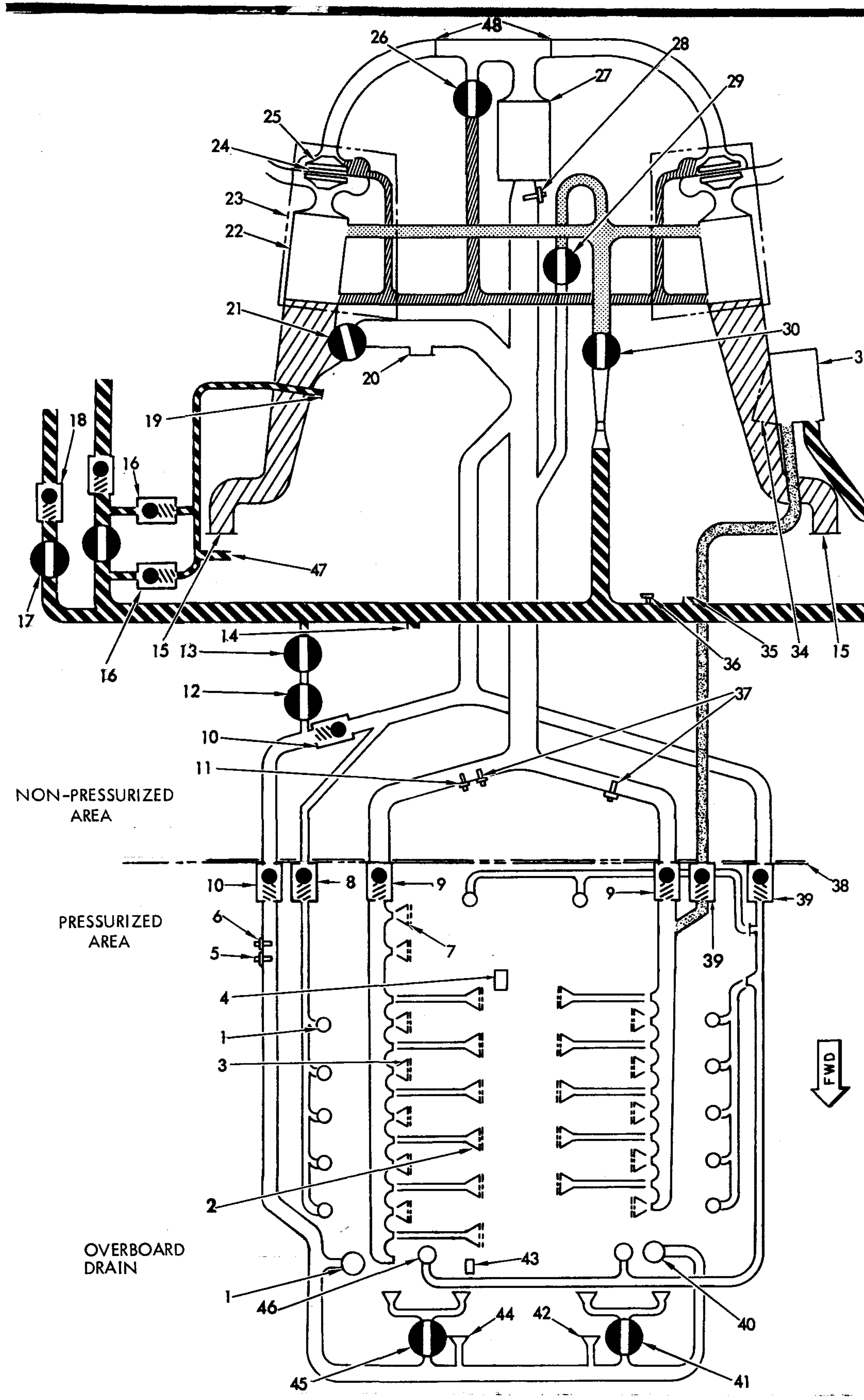


Figure 21-2 Air Conditioning System Flow Schematic Diagram

Changed 1 October 1985

1. COPILOT'S BODY WARMER OUTLET
2. OVERHEAD OUTLET (TYPICAL)
3. FLOOR OUTLET (TYPICAL)
4. CABIN TEMPERATURE SENSOR
5. FLIGHT STATION HIGH-LIMIT THERMOSTAT △5
FLIGHT STATION DUCT SENSOR △4
6. FLIGHT STATION TEMPERATURE ANTICIPATC
7. AFT CABIN OUTLET
8. GASPER DUCT CHECK VALVE
9. CABIN DUCT CHECK VALVE
10. FLIGHT STATION DUCT CHECK VALVE
11. CABIN TEMPERATURE ANTICIPATOR
12. FLIGHT STATION TEMPERATURE CONTROL VALVE
13. FLIGHT STATION AIR CONDITIONING SHUTOFF VALVE
14. CABIN PRESSURIZATION JET PUMP SUPPLY △2
15. RAM AIR SCOOP
16. WING AND EMPENNAGE DE-ICING CHECK VALVE
17. ENGINE BLEED AIR SHUTOFF VALVE (4 INSTALLED)
18. BLEED AIR CHECK VALVE (4 INSTALLED)
19. WING AND EMPENNAGE DE-ICING SUPPLY
20. GROUND AIR CONDITIONING CONNECTION
21. RAM AIR SHUTOFF VALVE
22. HEAT EXCHANGER (2 INSTALLED)
23. REFRIGERATION UNIT (2 INSTALLED)
24. RAM AIR FAN (2 INSTALLED)
25. HIGH-SPEED TURBINE (2 INSTALLED)
26. TURBINE BYPASS VALVE
27. WATER SEPARATOR
28. LOW-LIMIT SENSOR △6
29. REFRIGERATION UNIT BYPASS VALVE
30. BLEED AIR FLOW CONTROL VALVE
31. EMERGENCY PRESSURIZATION HEAT EXCHANGER
32. EMERGENCY PRESSURIZATION SHUTOFF VALVE
33. ENGINE BLEED AIR INLET (4 INSTALLED)
34. EMERGENCY PRESSURIZATION RAM AIR CONTROL
35. HYDRAULIC RESERVOIR AIR PRESSURE SUPPLY
36. GROUND HIGH PRESSURE CONNECTION
37. CABIN HIGH-LIMIT THERMOSTAT △1
38. AFT PRESSURE BULKHEAD
39. GASPER DUCT CHECK VALVE
40. PILOT'S BODY WARMER OUTLET
41. PILOT'S FOOT WARMER CONTROL
42. PILOT'S WINDSHIELD DE-FOGGER
43. FLIGHT STATION TEMPERATURE SENSOR △5
44. COPILOT'S WINDSHIELD DE-FOGGER
45. COPILOT'S FOOT WARMER CONTROL
46. GASPER OUTLET (TYPICAL)
47. CABIN PRESSURIZATION JET PUMP SUPPLY △3
48. ANTI ICING SCREEN ASSEMBLY △7

CODE	DESIGN CONDITIONS							
	NORMAL MAXIMUM				ABSOLUTE MAXIMUM			
	HOT DAY		COLD DAY		HOT DAY		COLD DAY	
	PRESS PSIG	TEMP F°	PRESS PSIG	TEMP F°	PRESS PSIG	TEMP F°	PRESS PSIG	TEMP F°
▨	87	600	130	450				
					91	615	139	460
░	87	600	101	540				
					91	615	139	460
▧	87	190	101	110				
					91	200	139	10
▤						146	4.2	
═							10.6	
▓							20	

NOTE

△1 HIGH LIMIT THERMOSTAT IS ON RIGHT SIDE ON AIRPLANES 5001 THRU 5045 AND ON LEFT SIDE ON AIRPLANES 5046 AND UP AND ON 5001 THRU 5045 ON WHICH SERVICE BULLETIN 329-106 HAS BEEN ACCOMPLISHED.

△2 ON AIRPLANES 5001 THRU 5045 NOT HAVING SERVICE BULLETIN 329-61 INCORPORATED.

△3 ON AIRPLANES 5046 AND UP AND ON AIRPLANES ON WHICH SERVICE BULLETIN 329-61 HAS BEEN INCORPORATED.

△4 ON AIRPLANES 5086 AND UP AND ON AIRPLANES ON WHICH SERVICE BULLETIN 329-213 OR SERVICE BULLETIN 329-235 HAS BEEN INCORPORATED.

△5 ON AIRPLANES 5001 THRU 5085 NOT HAVING SERVICE BULLETIN 329-213 OR SERVICE BULLETIN 329-235 INCORPORATED.

△6 ON AIRPLANES 5001 THRU 5085, 5093, 5097 AND UP, AND THOSE WITH SERVICE BULLETIN 329-234 INCORPORATED.

△7 ON AIRPLANES 5086 THRU 5092, 5094 THRU 5096 NOT HAVING SERVICE BULLETIN 329-234 INCORPORATED.

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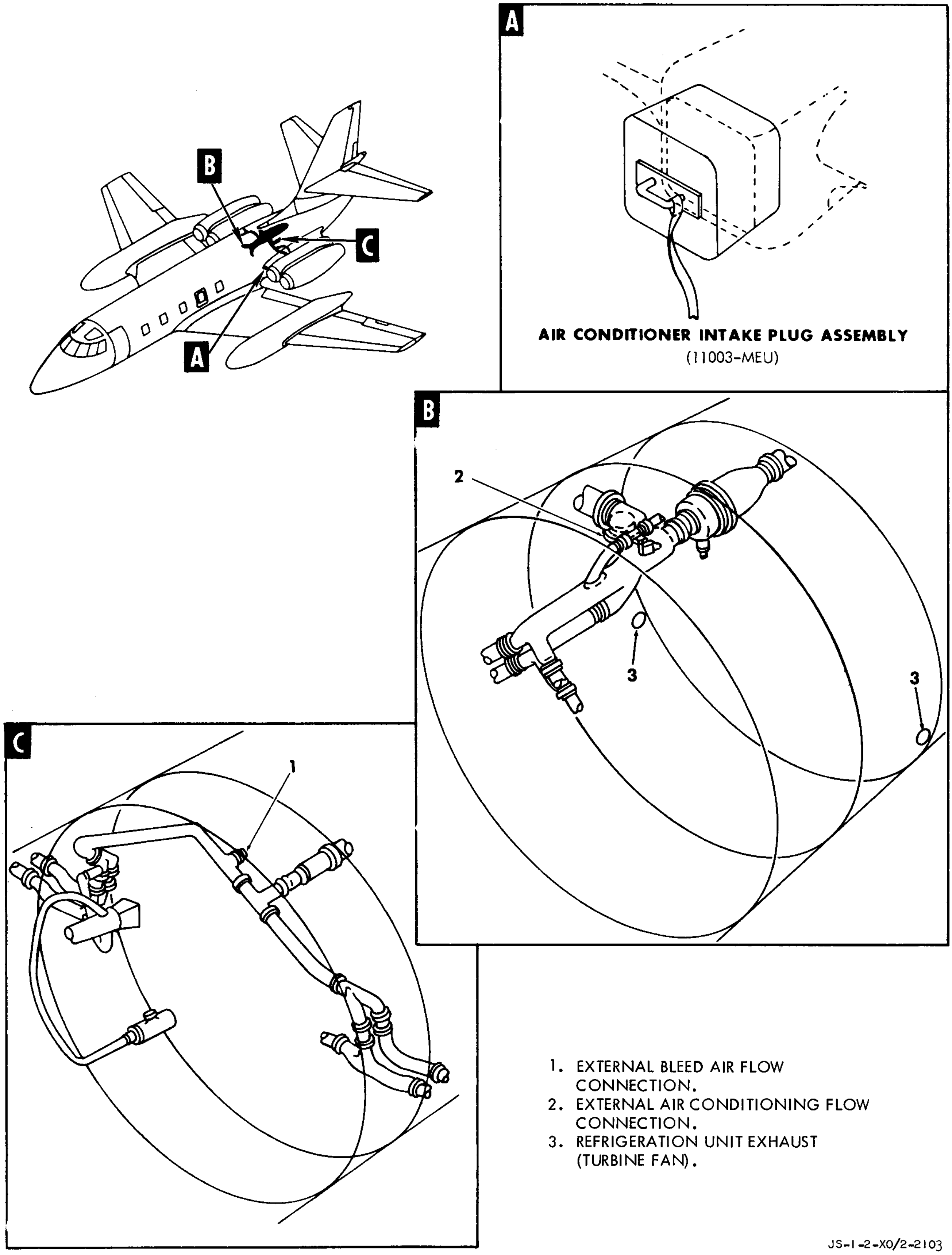
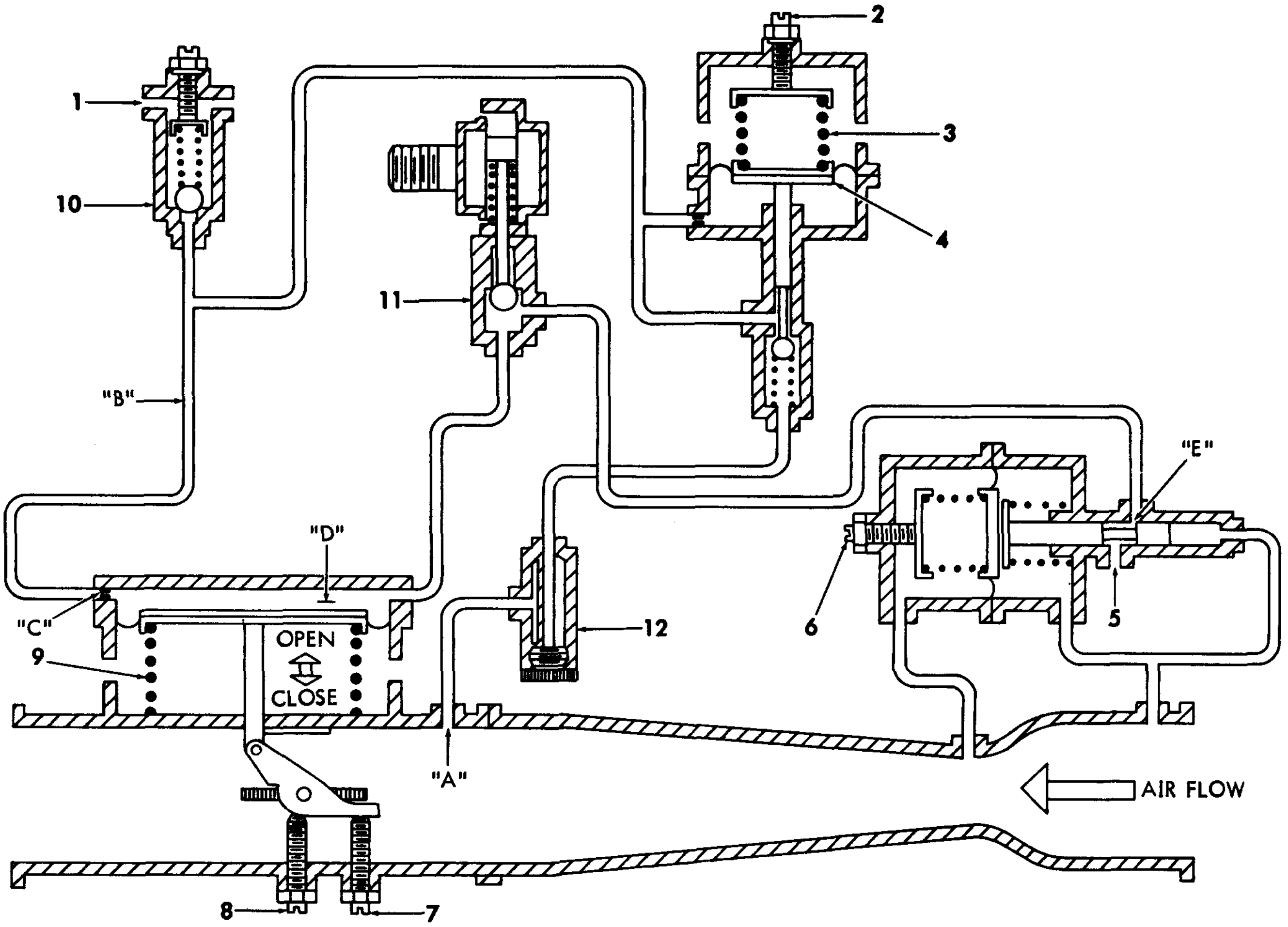


1. EXTERNAL BLEED AIR FLOW CONNECTION.
2. EXTERNAL AIR CONDITIONING FLOW CONNECTION.
3. REFRIGERATION UNIT EXHAUST (TURBINE FAN).

Figure 21-3. External Power Connections and Intake Plug



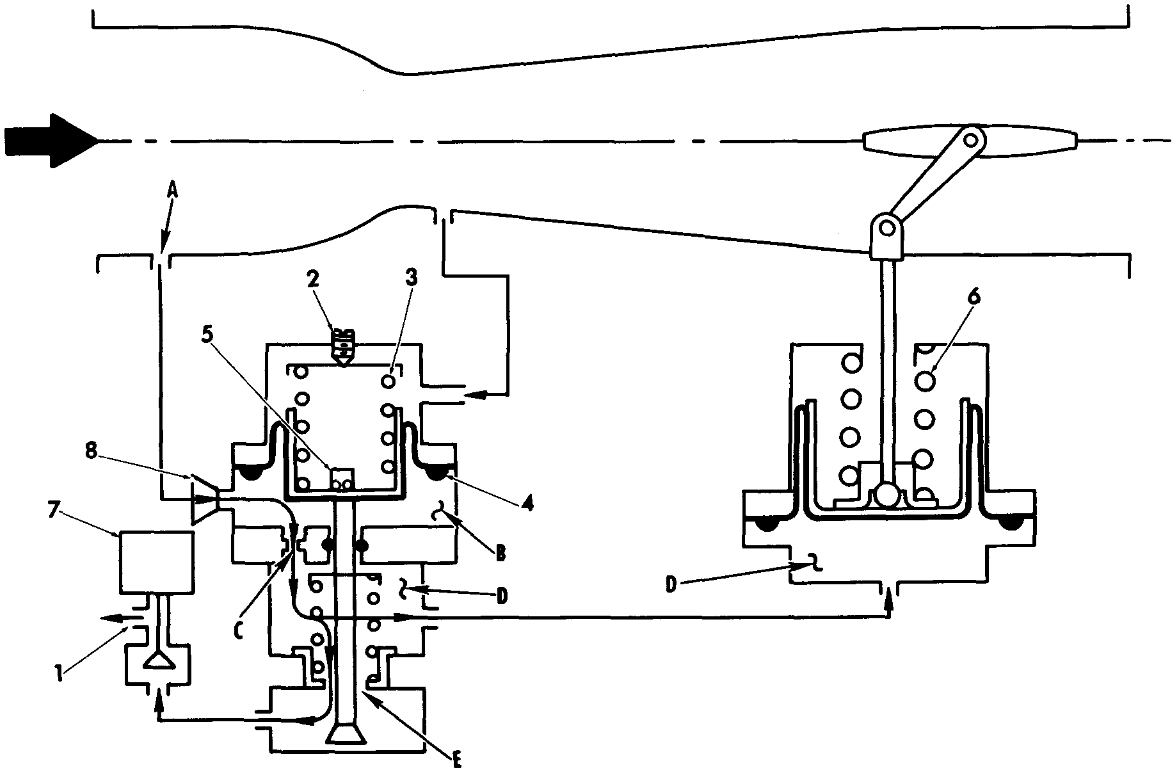
1. AMBIENT VENT
2. REGULATOR CALIBRATION ADJUSTMENT
3. REGULATOR SPRING
4. REGULATOR DIAPHRAGM
5. AMBIENT VENT
6. FLOW CONTROL CALIBRATION ADJUSTMENT
7. OPEN-STOP ADJUSTING SCREW
8. CLOSE-STOP ADJUSTING SCREW
9. BUTTERFLY ACTUATOR SPRING
10. ACTUATOR PRESSURE RELIEF VALVE
11. SOLENOID
12. FILTER

OPERATION:
AIR IS ADMITTED TO THE PNEUMATIC CONTROL SYSTEM AT "A" AND PASSES THROUGH A FILTER TO A SMALL REGULATOR WHICH HOLDS A CONSTANT PRESSURE IN LINE "B". THIS LINE CONNECTS TO THE VALVE ACTUATOR CHAMBER "D" THROUGH ORIFICE "C", TO BUILD UP A PRESSURE TENDING TO CLOSE THE VALVE (FAILSAFE OPEN). THE BUTTERFLY POSITION IS CONTROLLED BY THE RATE AT WHICH AIR IN CHAMBER "D" IS BLED OFF THROUGH A SERVO METERING ORIFICE "E". THE SIZE OF THIS METERING ORIFICE DECREASES WITH AN INCREASE OF DIFFERENTIAL PRESSURE (INCREASE OF FLOW) IN THE VENTURI. THIS BUILDS UP THE VALVE ACTUATOR PRESSURE CAUSING THE BUTTERFLY TO MOVE TOWARD THE CLOSED POSITION. HENCE, ANY CHANGE IN UPSTREAM PRESSURE TENDING TO INCREASE THE FLOW THROUGH THE VENTURI IS COMPENSATED BY A PROPORTIONATE REDUCTION OF THE VALVE OPENING WHICH MAINTAINS THE DESIRED SCHEDULE OF WEIGHT FLOW.

A NORMALLY OPEN SOLENOID IN THE SERVO BLEED LINE WILL CLOSE THE VALVE WHEN IT IS ENERGIZED PROVIDED THERE IS PRESSURE IN THE UNIT. UNDER THESE CONDITIONS IT IS POSSIBLE FOR SEEPAGE PAST THE REGULATOR TO BUILD UP PRESSURE IN LINE "B" WHICH MIGHT RUPTURE THE DIAPHRAGMS. A SMALL RELIEF VALVE IS PLACED IN THIS LINE AS A SAFETY FEATURE.

JS-2-2-2104

Figure 21-4. Bleed Air Flow Control Valve (AiResearch Part No. 106756)

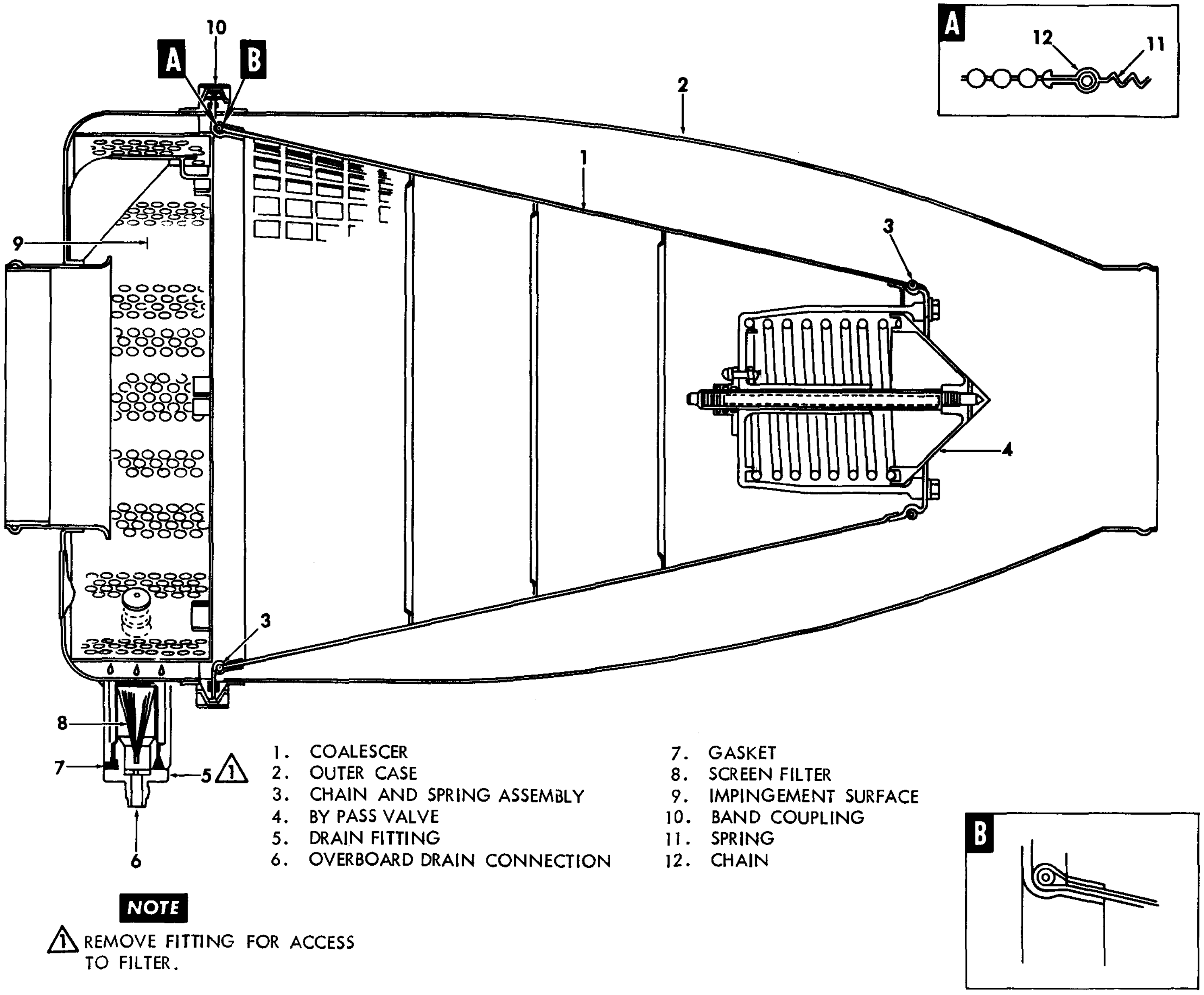


1. AMBIENT VENT
2. REGULATOR CALIBRATION ADJUSTMENT
3. REGULATOR SPRING
4. REGULATOR DIAPHRAGM
5. FLOW CONTROL CALIBRATION ADJUSTMENT
6. BUTTERFLY ACTUATOR SPRING
7. SOLENOID
8. FILTER

OPERATION:
AIR IS ADMITTED TO THE PNEUMATIC CONTROL SYSTEM AT "A" AND PASSES THROUGH A FILTER TO A SMALL REGULATOR WHICH HOLDS A CONSTANT PRESSURE IN CHAMBER "B". THIS CHAMBER CONNECTS TO THE VALVE ACTUATOR CHAMBER "D" THROUGH ORIFICE "C", TO BUILD UP A PRESSURE TENDING TO CLOSE THE VALVE (FAILSAFE OPEN). THE BUTTERFLY POSITION IS CONTROLLED BY THE RATE AT WHICH AIR IN CHAMBER "D" IS BLED OFF THROUGH A METERING ORIFICE "E". THE SIZE OF THIS METERING ORIFICE DECREASES WITH AN INCREASE OF DIFFERENTIAL PRESSURE (INCREASE OF FLOW) IN THE VENTURI. THIS BUILDS UP THE VALVE ACTUATOR PRESSURE CAUSING THE BUTTERFLY TO MOVE TOWARD THE CLOSED POSITION. HENCE, ANY CHANGE IN UPSTREAM PRESSURE TENDING TO INCREASE THE FLOW THROUGH THE VENTURI IS COMPENSATED BY A PROPORTIONATE REDUCTION OF THE VALVE OPENING WHICH MAINTAINS THE DESIRED SCHEDULE OF WEIGHT FLOW.

A NORMALLY OPEN SOLENOID IN THE SERVO BLEED LINE WILL CLOSE THE VALVE WHEN IT IS ENERGIZED PROVIDED THERE IS PRESSURE IN THE UNIT.

Figure 21-5. Bleed Air Flow Control Valve (Pneu-Tech Laboratories Part No. 31013)

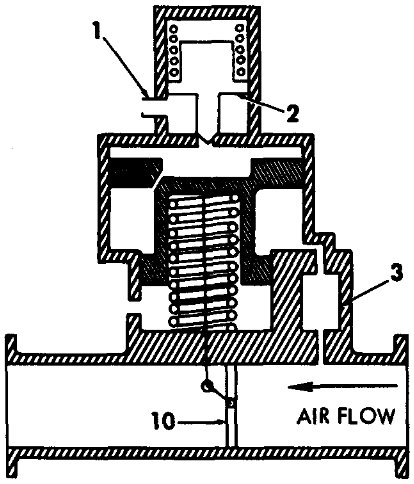


1. COALESCER
2. OUTER CASE
3. CHAIN AND SPRING ASSEMBLY
4. BY PASS VALVE
5. DRAIN FITTING
6. OVERBOARD DRAIN CONNECTION
7. GASKET
8. SCREEN FILTER
9. IMPINGEMENT SURFACE
10. BAND COUPLING
11. SPRING
12. CHAIN

NOTE

⚠1 REMOVE FITTING FOR ACCESS TO FILTER.

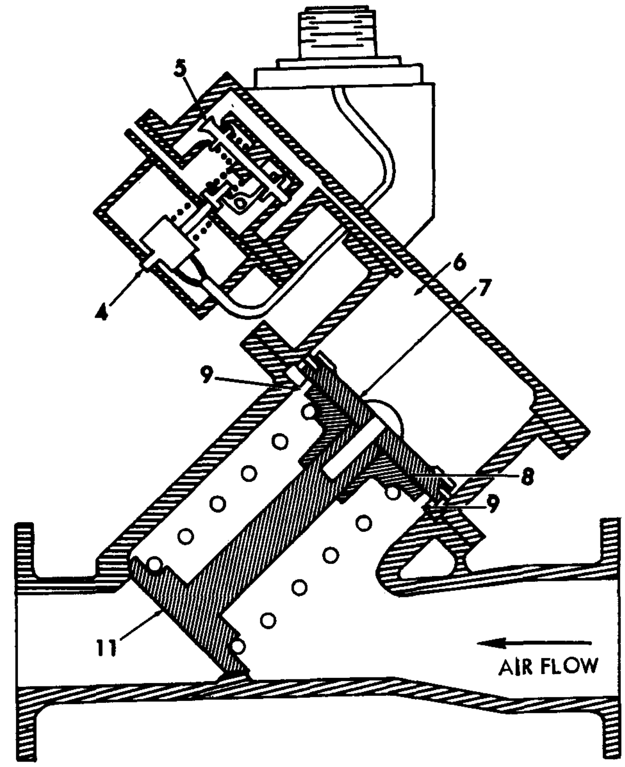
Figure 21-6. Water Separator



ENGINE BLEED AIR SHUTOFF VALVE

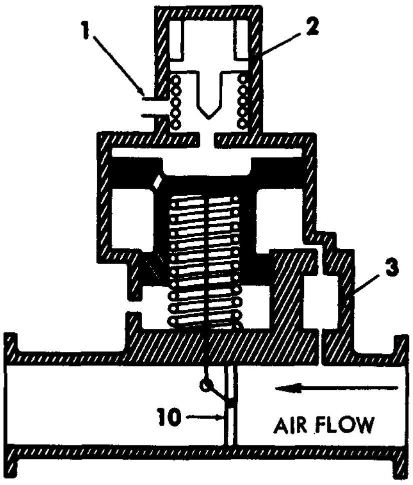
OPERATION: VALVE (10) IS SPRING-LOADED TO THE CLOSED POSITION REQUIRING UPSTREAM PRESSURE TO OPEN. ELECTRICAL POWER IS REQUIRED TO CLOSE VALVE. IN THE EVENT OF AN ELECTRICAL POWER FAILURE, THE VALVE WILL FAIL IN THE OPEN POSITION AND REMAIN OPEN WITH BLEED AIR PRESSURE.

1. VENT
2. SOLENOID
3. FILTER
4. SOLENOID (NORMALLY DE-ENERGIZED)
5. ACTUATOR POPPET VALVE
6. ACTUATOR CHAMBER
7. PISTON AND POPPET ASSEMBLY
8. STATIONARY GUIDE
9. CONTROL PORTS
10. BUTTERFLY
11. POPPET



FLIGHT STATION SHUTOFF VALVE

OPERATION: UPSTREAM AIR PASSES THROUGH THE CONTROL PORTS AND FORCES THE PISTON UP, COMPRESSING SPRING AND OPENING POPPET. LEAKAGE PAST THE PISTON RING INTO THE ACTUATOR CHAMBER IS BLED OFF BY THE ACTUATOR POPPET VALVE AND FLOWS TO ATMOSPHERE. WHEN THE SOLENOID IS ENERGIZED, IT CLOSES THE ACTUATOR POPPET VALVE. THIS ALLOWS THE PISTON LEAKAGE TO EQUALIZE THE PRESSURE IN THE ACTUATOR CHAMBER WITH UPSTREAM PRESSURE. THE SPRING THEN CLOSES THE POPPET.



SOLENOID-OPERATED EMERGENCY PRESSURIZATION SHUTOFF VALVE

OPERATION: VALVE (10) IS SPRING-LOADED IN THE CLOSED POSITION REQUIRING UPSTREAM PRESSURE AND ELECTRICAL POWER TO OPEN. IN THE EVENT OF AN ELECTRICAL POWER FAILURE, VALVE FAILS IN THE CLOSED POSITION AND REMAINS CLOSED WITH BLEED AIR PRESSURE.

THIS VALVE INSTALLED ON AIRPLANE SERIALS 5001 THRU 5025, 5027, 5029 & 5031. AIRPLANE SERIALS 5026, 5028, 5030, 5032 & UP AND THOSE ON WHICH SERVICE BULLETIN 329-8 HAS BEEN INCORPORATED ARE EQUIPPED WITH ELECTRIC MOTOR OPERATED VALVE.

Figure 21-7. Pneumatically Operated Shutoff Valves

Figure 21-8. Air Conditioning and Pressurization Systems Electrical and Schematic Diagram (Airplanes 5001 thru 5050)

Figure 21-8A. Air Conditioning and Pressurization Systems Electrical Schematic Diagram (Airplanes 5051, 5052, 5053)

PRESSURIZATION CONTROL
WARNING LIGHTS
SEE JR128
CABIN PRESS. LOW
CABIN PRESS. HIGH
CABIN PRESS.
SEALED PRESS.
CABIN PRESS. LOW (ABSOLUTE) JQ1011-1
CABIN PRESS. HIGH (DIFF) JQ1010-1
AMBIENT PRESS.
BLEED AIR SHUTOFF VALVES
ENG NO. 1
ENG NO. 2
ENG NO. 3
ENG NO. 4
BLEED AIR SHUTOFF
BLEED AIR SHUTOFF CONT SW
FIRE EMERG CONT SW
EMERG PRESS
EMERG DEPRESSURIZATION AND AIR-CONDITIONING
RAM AIR CONT
CLOSE
OPEN
NORM
EMERG DEPRESS
EMERG DEPRESS. RELAY
RAM AIR SHUTOFF VALVE JQ1008-1
ESSENTIAL DC BUS
EMERG PRESS CONT SWITCH
OFF
ON
TEMP CONT SWITCH
DECR
INCR
EMERG PRESS. RAM AIR CONT RELAY
EMERG PRESS. BLEED AIR SHUTOFF RELAY
FILTER
EMERG PRESS. RAM AIR CONTROL JQ1023-1
TURBINE AND AIR-CONDITIONING VALVES
AIR-CONDITIONING SHUTOFF
CABIN CONTROL
FLT STA CONTROL
CABIN BLEED AIR FLOW CONT SHUTOFF JQ1001-1
FLT DECK EMERG SHUTOFF JQ1022-1
ELECTRONIC COOLING FAN
NOISE FILTER
BLOWER
DIFF PRESS.
ELECTRONICS COOLING FAN JQ1006-1
EMERG PRESS. CONT RELAY
EMERG PRESS. SHUTOFF VALVE RELAY
EMERG PRESS. SHUTOFF VALVE JQ1025-1
CABIN TEMP CONT
TEMP IND
FLT STA TEMP CONT
MAIN DC BUS
CABIN TEMP CONTROL
LOW LIMIT TEMP CONTROL
FLIGHT STATION TEMP CONTROL
CONTROL BOX JQ1030-18
CABIN BRIDGE
(UNLAG)
(LAG)
AYLF700-1
MP2 AYLZ7276-400
K2 (COOL)
K3 (HEAT)
MP1 AYLZ5809-400
LOW LIMIT BRIDGE
FLT STA BRIDGE
MP3 AYLZ7276-400
K4 (COOL)
K5 (HEAT)
CABIN TEMP IND 147B35
TEMP CONTROL CABIN
AUTO
WARM
COOL
HOLD
MANUAL SELECT RELAY
TEMP CONTROL FLT STA
CABIN SENSOR JQ1030-6
CABIN DUCT SENSING ELEMENT (ANTICIPATOR) JQ1030-19
CABIN TEMP SELECTOR JQ1030-13
CABIN DUCT THERMOSTAT (HIGH LIMIT) JQ1030-16
TURBINE BYPASS VALVE (TEMP CONTROL) JQ1030-15
REFRIG BYPASS VALVE (TEMP CONT) JQ1030-4
CABIN DUCT SENSING ELEM (LOW LIMIT) JQ1030-10
FLIGHT STATION SENSOR JQ1030-6
FLT STA DUCT SENSING ELEM (ANTICIPATOR) JQ1030-19
FLT STA TEMP SELECTOR JQ1030-13
FLT STA DUCT (THERMOSTAT) JQ1030-16
FLT STA VALVE (TEMP CONTROL) JQ1030-5

Figure 21-8B. Air Conditioning and Pressurization System Electrical Schematic Diagram (Airplanes 5054 thru 5065)

Figure 21-8C *Air Conditioning and Pressurization System Electrical Schematic Diagram (Airplanes 5066 thru 5085)*

Changed 1 December 1965

Figure 21-8D. Air-Conditioning and Pressurization Systems Electrical and Schematic Diagram (Airplanes 5086 thru 5092, 5094 thru 5096 Without Service Bulletin 329-234)

Figure 21-8E. *Air Conditioning and Pressurization Systems Electrical and Schematic Diagram (Airplanes 5093, 5097 and Up and on Airplanes 5086 through 5092, 5094, 5095, 5096 with Service Bulletin 329-234).*

SECTION 21.1
PRESSURIZATION SYSTEM

Contents Page

GENERAL 21-32

SYSTEM OPERATION 21-33
  Normal Ground Operation 21-33
    General 21-33
    Ground Control Procedure 21-33
    System Functional Description for Ground Operation 21-33
  Normal Flight Control Procedure 21-34
  General 21-34
    Flight Control Procedure 21-34
    System Functional Description for Flight Operation 21-34
  Manual Depressurization 21-35
  Emergency Operation 21-35

COMPONENT OPERATION 21-36
  Pressurization Control Panel 21-36
  Cabin Pressure Controller 21-36
  Outflow Valves 21-37
  Pneumatic Relay 21-37
  Manual Depressurization Control Valve 21-37
  Jet Pump 21-37
  Safety Valve 21-37
  Air Filters 21-38
  Isobaric Shutoff Test Valve 21-38
  Differential Shutoff Test Valve 21-38
  Jet Pump Pressure Regulator 21-38
  Warning Lights 21-38
  Emergency Pressurization Shutoff Valve 21-38
  Ram Air Shutoff Valve 21-38

OPERATIONAL CHECKOUT 21-38
  General Requirements 21-38
  Preparation for Testing 21-39
  Operational Checkout Procedure 21-39
  Cabin Uncontrolled Leakage 21-41
  Leak Testing Control Tubing 21-41

TROUBLESHOOTING 21-42
  General 21-42
  Air Inflow Check 21-42
    Bleed Air Shutoff Valve Operation 21-42
    Bleed Air Flow Control Valve Operation 21-42
    Bleed Air Duct Leakage 21-43
  Air Outflow Check 21-43
    Electronics Cooling Fan and Flow Regulator Operation 21-43
    Cabin Uncontrolled Leakage 21-43
  Pressurization Control System Check 21-43
    General 21-43
    Tubing Leak Check 21-43
    Component Malfunction Isolation 21-43
    Verification of Component Malfunction 21-44
      Pressure Controller 21-44
      Pneumatic Relay 21-45
      Outflow Valves 21-45

Contents Page

Jet Pump 21-45
Jet Pump Pressure Regulator 21-45

REMOVAL AND INSTALLATION 21-45
General 21-45
Removal of Tubing 21-45
Installation of Tubing 21-45
Removal of Outflow Valves 21-45
Installation of Outflow Valves 21-45
Removal of Safety Valves 21-46
Installation of Safety Valves 21-46
Removal of Pressure Controller 21-46
Installation of Pressure Controller 21-46
Removal of Pneumatic Relay 21-46
Installation of Pneumatic Relay 21-46
Removal of Manual Depressurization Control Valve 21-47
Installation of Manual Depressurization Control Valve 21-47
Removal of Air Filters 21-47
Removal of Controller External Filter 21-47
Removal of Pneumatic Relay Filter 21-47
Removal of Safety Valve Filter 21-47
Installation of Air Filters 21-47
Installation of Controller External Filter 21-47
Installation of Pneumatic Relay Filter 21-47
Installation of Safety Valve Filter 21-47
Removal of Jet Pump 21-47
Installation of Jet Pump 21-48
Removal of Test Valves 21-48
Installation of Test Valves 21-48
Removal of Jet Pump Pressure Regulator 21-48
Installation of Jet Pump Pressure Regulator 21-48

SERVICING 21-48
Adjustment 21-48
Water Drainage 21-48
Filter Cleaning and Element Replacement 21-48
Controller External Filter 21-48
Controller Internal Filters 21-49
Pneumatic Relay Filter 21-49
Safety Valve Filters 21-49
Cleaning Valve Seats 21-49
Cleaning Static Ports 21-49
Overhaul 21-49
System Preservation. 21-49
Normal Operation 21-49
Storage of the Airplane 21-49

FIGURES

21-9. Pressurization System Flow Schematic Diagram 21-51
(Airplanes 5001 Thru 5045 Not Modified by Service Bulletin 329-61)
21-10. Pressurization System Flow Schematic Diagram 21-52
(Airplanes 5046 thru 5125 and Those Modified by Service Bulletin 329-61)
21-10A. Pressurization System Flow Schematic Diagram 21-52A
(Airplanes 5126 and up and Airplanes Modified by Service Bulletin 329-259)

21.1.1 GENERAL. The pressurization system uses the temperature-conditioned air supplied by the air conditioning system to pressurize the fuselage for crew and passenger comfort in flight. The degree of pressurization is controlled by regulating the flow of air overboard. The system also controls the rate of change of fuselage pressure as the airplane climbs or descends, to avoid discomfort which would be caused by rapid pressure changes. Overriding devices are provided by which pressure can be dumped manually or automatically under certain flight conditions. Emergency pressurization or ventilation also can be

21.1.1 (Cont).

selected, if the normal system is inoperative. In addition to the air exhausted for normal pressure control, a small amount of cabin air is exhausted overboard to ventilate the toilet area, and an additional amount is exhausted overboard through the nose electronic equipment compartment, through a flow regulator valve and fan assembly, to provide cooling for the equipment. Refer to Section 21.2 for description of the electronic equipment cooling system, and to Section 21.3 for description of the toilet ventilation system.

21.1.2 SYSTEM OPERATION. The system is controlled by a pressurization control panel and a cabin pressure controller on the overhead control panel in the flight station. The controlling medium is air pressure acting on pneumatic components in response to signals from the pressure controller. The control pressures result from bleed air flow through a jet pump. The bleed air is tapped from the bleed air manifold in the aft equipment compartment and, due to the unavailability of bleed air with engines inoperative, at least one engine must be operating or an auxiliary source of high-pressure air must be available to the engine bleed air manifold for the control system to operate. Electrical power is used only to operate cabin low- and high-pressure warning lights, ram air ventilation, and the emergency pressurization system. Other than these four circuits, the pressurization system is completely independent of the airplane electrical power supply system. A means of overriding the automatic pressurization control system is provided for manual depressurization of the airplane. An emergency pressurization system is provided by which bleed air can be supplied for pressurization in the event of loss of pressure due to failure of the primary bleed air supply system. Because of the limited cooling capacity when this system is used at low altitude on a hot day, a means of ventilation by use of ram air is provided. If the difference between cabin and atmospheric pressure tends to become excessive, safety valves open to release air from the fuselage or to admit air into the fuselage. The safety valve operation is completely automatic, depending only upon differential pressure between cabin and atmosphere.

21.1.2.1 NORMAL GROUND OPERATION.

21.1.2.1.1 GENERAL. The airplane is normally operated during takeoff and landing with the cabin completely unpressurized. It is necessary for the outflow valves to be open if windows and doors are closed, in order to establish outflow and prevent the inflow of bleed air from pressurizing the fuselage.

The normal function of some components under certain conditions affect the cabin pressure when the pressure control system is inoperative. For instance, the electronics cooling fan will be operated when electrical power is on, even though engines are not operating. If the windows and doors are closed, the cabin pressure will decrease due to the exhaust of air, with no corresponding inflow, to about one-half inch of mercury. If an engine is then started, the pressurization control system will become operative, the outflow valves will be opened and the cabin pressure will suddenly increase to atmospheric, approximately. These sudden decreases and increases in cabin pressure will be uncomfortable for most passengers. Extremely rapid engine power changes can also cause similar pressure surges under certain conditions.

21.1.2.1.2 GROUND CONTROL PROCEDURE. The following procedure should be followed during all ground operation and during takeoff until the cruise condition is set on the controller (see paragraph 21.1.2.2). By following this procedure, the uncomfortable "pressure bumps" described in paragraph 21.1.2.1 will be reduced to an acceptable level or eliminated.

a. Before electrical power is turned on, open a clear vision window, or, if weather does not permit, open the ram air ventilation valve (see paragraph 21.1.3.14).

b. Set the correct field elevation on the airplane altimeter, and set the cabin altitude selector for a cabin altitude of 100 to 300 feet above field elevation.

Note

Before setting, rotate the cabin altitude selector knob counterclockwise until the internal stop is contacted, then rotate clockwise to the desired setting (see paragraph 21.1.3.2).

Set the rate of climb knob at "MIN".

c. Start the first engine.

d. Close ram air ventilation valve or window.

e. Start remaining engines and do not change controller settings until after takeoff (see section 21.1.2.2).

f. It may be necessary on airplanes 5001 through 5045 except those on which Service Bulletin 329-61 has been incorporated, to taxi with one engine at 60% RPM or higher to keep the system in complete control.

21.1.2.1.3 SYSTEM FUNCTIONAL DESCRIPTION FOR GROUND OPERATION. Normal ground operation is unpressurized. This is accomplished by setting the cabin altitude selector for a cabin altitude slightly higher than the ground elevation (100 to 300 feet). This setting will cause the isobaric metering valve in the pressure controller to remain open. The jet pump will then induce a reduced pressure in the head chamber of the pneumatic relay (figure 21-9 or

21.1.2.1.3 (Cont).

21-10) and the metering pin will be opened. The jet pump will then induce a reduced pressure in the head chamber of the outflow valves, and they will be opened to permit cabin air to flow out.

21.1.2.2 NORMAL FLIGHT CONTROL PROCEDURE.

21.1.2.2.1 GENERAL. On airplanes 5001 through 5045, except those on which Service Bulletin 329-61 has been incorporated, cabin pressure surges (pressure bumps) may be experienced when engine power is changed rapidly from idle, or to idle, due to low bleed air manifold pressure at idle power. It may, therefore, be necessary to keep at least one engine power at 60% RPM or above during taxi on the ground and during flight descents, to keep the pressure control system completely operative.

21.1.2.2.2 FLIGHT CONTROL PROCEDURE. (See paragraph 21.1.2.1.2 for ground operating and takeoff procedure). After takeoff and the climb is established, the controller settings for cruise may be made. Normally, for maximum comfort, it is desirable to cruise with the lowest cabin altitude practicable. The window in the controller dial, at the 6 o'clock position, which displays the flight altitude for maximum differential pressure, may be used for this purpose. The cabin altitude selector should be rotated until the desired airplane cruise altitude is displayed in the window (maximum is 45,000 at which cabin altitude at maximum differential will be 8,000 feet). The rate of climb knob should be set approximately vertical (500 feet per minute). The ''MIN'' position is 200 feet per minute and the ''MAX'' position is 2000 feet per minute rate of change. The controller needs no further attention during the climb and cruise phase of the flight.

The controller will require attention during descent, and the type of descent determines the method of operation. Timing is the important factor in successful cabin pressure control in the descent, the object being to have the airplane altitude and cabin altitude coincide when landing traffic altitude is reached. The following procedure is recommended:

a. Uniform Descent. If a steady descent is to be made, prior to starting the descent, estimate the average airplane rate of descent and determine total time required. Determine what rate of cabin altitude change will cause the cabin to arrive at desired altitude in the estimated time interval, and set the rate knob accordingly.

b. Step Descent. The objective is the same, i.e., to have the airplane altitude and cabin altitude reach landing traffic altitude at the same time. Due to the usually unknown time of loiter at each step, the cabin altitude and rate settings will require monitoring during each step. The descent for each step may be accomplished as a uniform descent with controller settings changed as required.

21.1.2.2.3 SYSTEM FUNCTIONAL DESCRIPTION FOR FLIGHT OPERATION.

a. The pressure controller is divided into two chambers: the control chamber, which produces primary control pressure, and a rate chamber, which modulates primary control pressure in response to settings of the controls. (See figure 21-9 or 21-10.) These two chambers are separated by a diaphragm, which is the medium by which the primary control pressure is normally varied, and are connected by an air passage having an internal filter and adjustable needle valve. Flow through this passage tends to equalize pressure in the two chambers. It is adjustable by varying the adjustable valve with a rate control knob. This restriction of flow prevents immediate equalization of pressure in the rate and control chambers and a corresponding immediate change of cabin altitude to that set on the controller.

b. As the airplane takes off and begins to climb, controlled pressurized operation begins with air flowing from the outflow valves. (See paragraph 21.1.2.1.3.) The cabin altitude, selected on the controller by the cabin altitude selector knob, represents the pressure altitude to which the fuselage is to be pressurized when the airplane has reached the flight altitude. The altitude selector knob loads a cabin altitude selector spring through a screw mechanism, and it drives a selected altitude indicator, which shows the cabin altitude which has been selected. The cabin altitude selector spring applies force, through a mechanical linkage, to the diaphragm between the control chamber and the rate chamber. An isobaric metering valve, moved by the diaphragm, affects primary control pressure.

c. The rate of cabin pressure change, as the airplane climbs or descends, is adjustable by means of the rate control knob, which varies the flow of air between the control chamber and the rate chamber of the controller. When the airplane climbs or descends at a rate in excess of that which has been selected on the rate control, the flow of air from one chamber of the controller to the other, caused by air entering or leaving the control chamber through the filter and orifice, is not rapid enough for immediate balancing of the diaphragm between the two chambers by the airflow through the internal filter and air passage, and past the metering pin of the rate control. The pressures on the diaphragm separating the two controller chambers are thus out of balance, which either compresses a rate control spring, if the airplane is descending, or allows it to expand, if the airplane is climbing. The isobaric metering valve opening varies in accordance with this, closing more in a climb and opening more in a descent. This changes the primary control pressure faster than an evacuated bellows can sense the change. The

21.1.2.2.3 (Cont).

evacuated bellows forms part of the linkage controlling the isobaric metering valve opening, being a variable-length link, modulating the isobaric valve opening according to rate control chamber pressure. A change in airplane altitude at a rate equal to, or less than, the selected rate permits this bellows to perform its normal function by expanding or contracting according to rate chamber pressure. This, in turn, changes primary control pressure by altering the isobaric metering valve setting. In a climb or descent at a rate greater than that selected for cabin pressure change, the evacuated bellows no longer senses the change rapidly enough, in effect acting as a brake on it. The resultant change in primary control pressure is slower, delaying the decrease or increase in cabin pressure by its effect on secondary control pressure, which controls the outflow valve position. As the airplane climbs past the selected cabin altitude, the resultant reduction in cabin pressure reduces primary control pressure, the jet pump partial vacuum input remaining constant, due to the action of the pressure regulator. The reduction in primary control pressure causes the diaphragm between the control chamber and the rate chamber to move toward the control chamber, carrying the isobaric metering valve with it. This reduces the valve opening through which the jet pump is drawing air from the control chamber, in turn reducing the differential between primary control pressure and cabin pressure. This, in turn, reduces the differential pressure in the pneumatic relay, causing the relay metering pin to start restricting flow to the jet pump. Secondary control pressure then increases in the relay actuator chamber and is transmitted to the head chambers of the outflow valves, causing the poppets to close slightly to maintain a constant, or isobaric, cabin pressure. Continued increase in airplane altitude modulates the outflow valve openings according to this schedule, so that they move gradually toward the closed position to retain pressurization at the selected level.

d. A selected cabin altitude cannot, however, be maintained regardless of airplane altitude, due to the need for limiting the differential pressure between the cabin and atmosphere to a safe value in terms of structural strength. A differential metering valve in the controller opens under these conditions, to relieve primary control pressure to atmosphere. The valve is spring-loaded against a seat, the assembly moving as part of a diaphragm until the valve stem makes contact with the end of a high differential adjustment screw. The diaphragm is also spring-loaded, by a differential calibration spring, against a disk attached to the adjustment screw. Both these springs tend to hold the valve closed. One side of this diaphragm is exposed to atmospheric pressure, and the other side to control chamber pressure (cabin pressure). When the differential between these two pressures becomes high enough, 8.8 PSI, the diaphragm moves, under control chamber pressure, to compress the differential calibration spring and to carry the valve stem toward the end of the adjustment screw. Further diaphragm movement in this direction causes the valve stem to butt against the adjustment screw, while the valve seat continues to move with the diaphragm, compressing the valve spring and opening the valve. Control chamber pressure then bleeds to atmosphere. The resulting reduction in control chamber pressure reduces primary control pressure, causing the pneumatic relay to open the outflow valves. Cabin pressure drops until a new balance is obtained between control chamber pressure and atmospheric pressure. This causes the differential metering valve to close, assisted by spring pressure, and the control system to continue to modulate outflow valve position.

21.1.2.3 MANUAL DEPRESSURIZATION. Two methods may be used to depressurize the cabin. When time permits, the air conditioning and pressurization system may be turned off to allow normal leakage to reduce cabin pressure. If a more rapid and controllable rate of depressurization is required, the MANUAL DEPRESS CONTROL on the pressurization control panel is turned in the "DEPRESSURIZE" direction. This manually opens a valve in the primary control pressure system, bleeding primary control pressure to the partial vacuum produced by the jet pump. The reduction in primary control pressure opens the outflow valves, through action of the pneumatic relay on secondary control pressure. The rate of depressurization is controlled by the amount the manual valve is opened. Before landing, the valve is normally opened fully to ensure complete depressurization.

21.1.2.4 EMERGENCY OPERATION. If the normal air conditioning system should fail to provide cabin pressure, an emergency pressurization system can be used. This system obtains bleed air from the No. 2 engine, through an emergency pressurization valve which is tapped into the No. 2 engine bleed air supply line upstream of the bleed air shutoff valve. The emergency pressurization bleed air is routed from the emergency pressurization valve through a heat exchanger in the fuselage aft compartment. The heat exchanger uses ram air from the intake in the left engine nacelle to cool the bleed air. The amount of cooling is determined by the position of an emergency pressurization ram air flapper door and by the temperature and amount of ram air available which is, in turn, dependent on airplane altitude and airspeed. The door diverts air from the ram air duct and causes it to pass across the heat exchanger and out into the aft equipment area. The cooled bleed air is routed through the heat exchanger to the cabin, through an individual duct separate from the normal

21.1.2.4 (cont).

system ducts, through a check valve at the aft pressure bulkhead, and into the cabin left side outlet ducts. The emergency pressurization duct joins the normal system left side duct downstream of the normal system aft pressure bulkhead check valve. Control of the emergency pressurization system is by an EMER PRESS SYSTEM control switch and an EMER PRESS SYSTEM temperature switch on the pressurization control panel. The EMER PRESS SYSTEM control toggle switch is guarded to the "OFF" position. Raising the guard permits placing the switch in the "ON" position; this energizes all four engine bleed air shutoff valves to the closed position, opens the emergency pressurization system ram air control to divert ram air to the heat exchanger, and arms the EMER PRESS SYSTEM TEMP switch. The temperature of the emergency pressurization air supply is controlled by positioning the ram air flapper door, which diverts more air through the heat exchanger for decreased supply air temperature, or less air for increased temperature. The EMER PRESS SYSTEM TEMP switch has three positions: "DECR TEMP," in which power is applied to the extension winding of the ram air flapper door actuator to divert more ram air to the heat exchanger for increased cooling of the bleed air; "INCR TEMP," in which power is applied to the retraction winding of the actuator, to divert less ram air to the heat exchanger for decreased cooling of the bleed air: and a central, neutral position, which removes power from the actuator to leave it in the last-operated position. Control of cabin air temperature, in a limited range, is thus possible, by holding the temperature control switch in the "INCR" or "DECR" position until the air temperature reaches the desired level, and then releasing it to the neutral position, to leave the flapper door in the position which has provided this air temperature. At low airplane altitudes, operation of the emergency pressurization system may cause excessively high air temperature. Ground checkout of the system should therefore be as brief as possible to avoid heat damage to ducts and components. When the cabin temperature cannot be controlled sufficiently low for comfort when using the emergency pressurization system, which may occur at altitudes below approximately 14,000 feet, ram air ventilation should be used.

Air for ventilation is obtained from the ram air intake in the right pylon, through a ram air shutoff valve, and is delivered to the cabin and flight station through the normal air conditioning ducts and outlets. The ram air shutoff valve is controlled by a RAM AIR switch on the pressurization control panel. Electrical power to operate the motor-driven ram air shutoff valve is available only after the NORMAL-DEPRESS switch, on the pressurization control panel, has been placed in the "DEPRESS" position. This *energizes an emergency depressurization relay, which* completes a power circuit to the RAM AIR switch and shuts off cabin (flow control valve) and flight station air conditioning. Inadvertent opening of the ram air shutoff valve during normal system operation is thus prevented. Holding the RAM AIR switch in the "OPEN" position energizes the valve motor to open the valve. It is necessary to hold the switch in this position for at least 12 seconds to allow the valve to move to the fully open position. To obtain airflow through the fuselage under these conditions, the outflow valves must be open, and this is accomplished by turning the MANUAL DEPRESS CONTROL counterclockwise to reduce secondary control pressure, thus overriding the pressure controller and opening the outflow valves.

21.1.3 COMPONENT OPERATION.

21.1.3.1 PRESSURIZATION CONTROL PANEL. The NORMAL-DEPRESS switch, the MANUAL DEPRESS CONTROL, the RAM AIR control switch, the emergency pressurization control switch, the emergency pressurization temperature control switch, a cabin rate of climb indicator, and a cabin differential pressure indicator are grouped on the pressurization control panel. The panel is overhead, in the flight station. For the functions of the control switches, refer to the system operation descriptions. The cabin rate of climb indicator shows the rate of change in cabin pressure in terms of rate of climb or descent in thousands of feet per minute. The indicator pressure port is open to cabin pressure. Cabin pressure rate of change can be monitored on the indicator and can be controlled by turning the rate control knob on the cabin pressure controller. (Refer to paragraph 21.1.2.2 for a description of rate control operation.) The differential pressure indicator, or dual pressure indicator, has an altitude scale, calibrated in thousands of feet, on which cabin altitude is indicated by a CABIN pointer and airplane altitude is indicated by a FLIGHT pointer. The CABIN pointer moves with a rotating card calibrated in pounds per square inch, while a DIFF pointer moves at a fixed angle to the FLIGHT pointer, to indicate differential pressure between the cabin and atmosphere on the rotating card. The pressure port for the cabin altitude indicator is open to cabin pressure, while the pressure port for flight altitude is connected to the third static pressure source in the airplane pitot-static system. (Refer to Chapter 34 for a description of the pitot-static system.) The FLIGHT and CABIN altitude scale is calibrated from 0 to 50,000 feet, and the DIFF scale from -1 to 10 PSI.

21.1.3.2 CABIN PRESSURE CONTROLLER. The outflow valves are regulated by the cabin pressure controller to maintain the desired cabin pressure. Operation of the pressure controller and its effect on the pressurization system is described in paragraphs 21.1.2.1 and 21.1.2.2. The controller is mounted in the overhead control panel in the flight station, adjacent to the bleed air, pressurization, and air conditioning control panels. Two controls are pro*vided on the cabin pressure controller, a RATE* control and a CABIN ALT selector. The RATE control is turned, in either the "MIN" or "MAX"

21.1.3.2 (Cont).

direction, to vary the opening of the needle valve controlling airflow between the control chamber and the rate chamber of the controller. This, in turn, varies the rate of change of isobaric metering valve aperture, which directly governs the rate of change of system control pressures and thus the rate of change of cabin pressure. The CABIN ALT selector varies the force on the cabin altitude selector spring, through a screw mechanism, which in turn adjusts the isobaric metering valve through a linkage including the evacuated bellows and the rate control spring. Thus, cabin pressure does not immediately change to a newly selected CABIN ALT setting, but the change is subject to the action of the RATE control. The CABIN ALT selector shaft drives a mechanical cabin altitude pointer through a gear train. The indicator scale is calibrated from -1,000 feet to 10,000 feet, but the pointer is not travel limited at these settings. The pointer may be rotated clockwise past the 10,000 feet point back on scale at -1,000 feet and if rotation is continued it will move to the 5,000 to 6,000 feet area on the dial, where an internal stop will prevent further movement. This imposes a proportionately higher load on the cabin altitude selector spring to hold the isobaric metering pin in the open position, thus causing the system to fail to hold the cabin altitude indicated by the pointer (i.e., cabin remains unpressurized or pressurized to a much higher cabin altitude than controller pointer setting). To prevent such a condition from inadvertently being set on the controller, prior to take-off the cabin altitude selector should be rotated counterclockwise until a mechanical limit is reached. From this point the knob should be rotated clockwise until the pointer is set approximately 100 to 300 feet above field pressure elevation for takeoff. A window in the controller dial shows the airplane altitude at which maximum differential pressure will be reached. A card, driven by the gear train, rotates with the pointer to display these figures in the dial window.

21.1.3.3 OUTFLOW VALVES. Air is released from the cabin during normal pressurized operation through the two outflow valves, which regulate the flow to obtain the desired degree of pressurization. Either valve has enough capacity for normal system operation. Valve position is governed by secondary control pressure and cabin pressure, as described in paragraphs 21.1.2.1 and 21.1.2.2. (See figure 21-1 for outflow valve location in the airplane).

21.1.3.4 PNEUMATIC RELAY. The function of the pressurization pneumatic relay is similar to that of an electrical relay, in that it enables one system to control operation of another system. The primary control pressure system controls the secondary control pressure system through the pneumatic relay. (Refer to paragraph 21.1.2.1 for a description of pneumatic relay operation.) An air filter, connected to the relay actuator chamber, prevents foreign matter from entering the secondary control pressure system. (See figure 21-1 for pneumatic relay location in the airplane.)

21.1.3.5 MANUAL DEPRESSURIZATION CONTROL VALVE. The MANUAL DEPRESS control valve is a normally closed needle valve installed in a line connecting the primary control pressure line to the jet pump partial vacuum line. Opening the needle valve will cause the primary control pressure to be reduced, thus opening the pneumatic relay metering pin to expose the outflow valve head chamber to the jet pump partial vacuum. This causes the outflow valves to open, resulting in depressurization of the cabin.

21.1.3.6 JET PUMP. Flow of primary and secondary air is maintained by the jet pump which induces by venturi action a secondary partial vacuum as the pressure-regulated bleed air flows through the primary nozzle. The jet pump has no moving parts. Exhaust air from the jet pump is discharged overboard through a hole in the fuselage skin or into the fuselage aft compartment, depending on the jet pump location. Two suction lines are connected to the jet pump, one routed to the control chamber of the cabin pressure controller, from which it draws air through the isobaric metering valve, and one routed to the pneumatic relay actuator chamber, where secondary control pressure is created by the combined action of the partial vacuum of the jet pump suction line and the diaphragm-controlled relay metering pin. The manual depressurization control valve, normally closed, is connected to the second suction line. When the valve is opened, jet pump suction acts on the cabin pressure controller control chamber and the relay head chamber to override the pressure controller, thus opening the outflow valves.

21.1.3.7 SAFETY VALVE. If the normal pressurization control system fails to limit the maximum differential pressure by outflow valve opening, the safety valves automatically open to prevent the differential pressure from exceeding 9.3 (±0.15) PSI. Either valve has enough capacity to prevent over-pressurization of the fuselage. The safety valves also open to prevent negative differential pressure from exceeding 0.38 PSI if cabin pressure altitude becomes higher than airplane altitude during an extremely rapid descent. The safety valves are actuated by differential between cabin and ambient pressures and are not connected to the pressurization control system. The safety valves are similar in construction to the outflow valves. As the differential pressure increases, cabin pressure enters the safety valve head chamber through a filter and restrictive orifice, and enters the actuator chamber through numerous ports around the poppet. The actuator chamber is formed by the base of the poppet and the diaphragm. Actuator pressure also reaches one side of a differential control diaphragm, where it is opposed on the other side of the diaphragm by a differential calibration spring and by atmospheric pressure. The differential control diaphragm carries a differential metering ball valve, which normally

21.1.3.7 (Cont).

closes a passage to the head chamber. When the differential control diaphragm is moved by the difference between actuator chamber pressure and ambient pressure, the ball valve is lifted from its seat, permitting head chamber pressure to bleed to atmosphere more rapidly than cabin pressure can enter the head chamber through the restrictive orifice. The resultant difference between head chamber pressure and actuator chamber pressure moves the safety valve knife-edge poppet from its seat to relieve cabin pressure to atmosphere. If atmospheric pressure becomes higher than cabin pressure, the atmospheric pressure outside the safety valve poppet, since it is higher than cabin pressure, pushes a free, external portion of the main diaphragm against the poppet, gradually peels the central portion of the diaphragm from its seat, and pushes a large area of it against the poppet. This forces the poppet open against the return spring, permitting ambient air to flow into the cabin. (Refer to figure 21-1 for safety valve location in the airplane.)

21.1.3.8 AIR FILTERS. Foreign matter in the small orifices and passages in the pneumatic control systems can cause malfunctions. Four external filters are used in the pressurization control systems: one for the control chamber inlet of the cabin pressure controller, one for the actuator chamber inlet of the pneumatic relay, and one on each of the safety valves at the inlet to the head chamber. These filters are accessible from outside the respective units. Internal filter-and-screen combination sets are incorporated in the cabin pressure controller, one to supplement the external filter, and one to filter the air flowing between the control chamber and the rate chamber through the internal connecting passage. All filters are designed so that sudden clogging is unlikely, while gradual clogging causes erratic system operation resulting from restricted airflow giving unbalanced pressures in the various chambers. (Refer to paragraph 21.1.7.3 for filter cleaning and replacement schedule.)

21.1.3.9 ISOBARIC SHUTOFF TEST VALVE. The isobaric shutoff test valve is a manual, plug-type, two-position, normally-open valve installed in the control chamber-to-jet pump line of the pressure controller. The valve is safety wired in the open position, and is closed only to facilitate testing the controller maximum differential calibration and outflow valves for maximum differential relief when the airplane is on the ground. In the closed position, the valve prevents secondary flow from the controller, thus causing the outflow valves to remain closed until the differential metering valve (see figure 21-9 or 21-10) is opened. See section 21.1.2.2 for a description of operation of the differential metering valve.

21.1.3.10 DIFFERENTIAL SHUTOFF TEST VALVE. The differential shutoff test valve is a manual, plug-type, two-position, normally-open valve installed in the atmosphere line connecting the differential chamber in the pressure controller to atmospheric pressure. (See figure 21-9 or 21-10.) The valve is the same as the isobaric shutoff test valve described in paragraph 21.1.3.9. It is normally safety wired in the open position and is closed only to facilitate testing the safety valves for maximum differential relief when the airplane is on the ground. When in the closed position, the valve closes the vent line from the high differential chamber in the controller to prevent outflow valves from opening at maximum differential, thus allowing pressure to increase to the safety valve relief setting.

21.1.3.11 JET PUMP PRESSURE REGULATOR. The jet pump pressure regulator is installed in the bleed air pressure line immediately upstream of the jet pump. It functions to limit the pressure at the inlet to the jet pump to a constant 8.25 (±1.25) PSI when the bleed air pressure varies due to engine power changes. This prevents surges in the secondary control partial vacuum created by the jet pump and the accompanying "pressure bump" in the cabin which results if the outflow valves movement is not smooth.

21.1.3.12 WARNING LIGHTS. Cabin pressure is monitored by a warning light system operated by pressure switches. A CABIN PRESS HIGH module on the annunciator panel goes on when a differential pressure switch is closed by cabin pressure exceeding ambient pressure by 9.0 to 9.5 PSI. A CABIN PRESS LOW module on the annunciator panel goes on when a pressure switch is closed as cabin pressure altitude reaches 9,000 to 10,000 feet. See Chapter 33 for electrical circuits.

21.1.3.13 EMERGENCY PRESSURIZATION SHUTOFF VALVE. See paragraph 21.0.2.4.15 for a description of this valve.

21.1.3.14 RAM AIR SHUTOFF VALVE. See paragraph 21.0.2.4.14 for a description of this valve.

21.1.4 OPERATIONAL CHECKOUT.

21.1.4.1 GENERAL REQUIREMENTS. The following equipment is required for checking out the operation of the pressurization system:

Compressor, Type MC-1A or equal
Stop watch
Generator Set, Type MD-3A or equal

WARNING

During this test, potentially dangerous conditions exist when the fuselage is pressurized. When pressurized, the fuselage can suffer major damage, with possible injury to personnel, if struck by any object, such as ground equipment. Make sure the main landing gear wheels are chocked forward and aft. Move

21.1.4.1 (Cont).

all ground equipment clear of the airplane. The compressor and generator set should be located at the full extent of the connecting line and cables. Do not permit personnel, except those engaged in the test, in or near the airplane while it is pressurized. Personnel in the airplane during the test must have medical clearance for work in pressurized areas.

Note

If a ground source of compressed air is not available, the airplane engines may be operated to furnish bleed air for the test. Run at least two engines, and set the air conditioning controls for normal operation at a comfortable temperature. One of the engines should be operated above 80 percent RPM.

a. Connect the generator set to the external DC power receptacle of the airplane. (Refer to Chapter 24.) Energize both the AC and the DC buses.

b. If an external supply of compressed air is to be used, connect the compressor to the air conditioning ground test connection, and adjust it to furnish 60 pounds per minute at 10 PSI. If the engines are to be used to pressurize the airplane, start at least two engines as instructed in Chapter 72.

21.1.4.2 PREPARATION FOR TESTING. Three men will be required to perform this checkout. Man A will be stationed in the flight station to operate the system controls, Man B will monitor the instruments and time the pressure changes with the stop watch, and Man C will observe the operation of the emergency pressurization ram air control actuator in the aft compartment and the electronics cooling flow regulator valve in the nose wheel well.

a. Check that the personnel entrance door, the pilot's and copilot's clear vision windows, and the emergency exits are closed and secured.

b. Check all air conditioning and gasper outlets to make sure they are open.

c. Check the isobaric and differential shutoff valves in the baggage compartment to make sure they are both open.

21.1.4.3 OPERATIONAL CHECKOUT PROCEDURE.

a. Man A: Close the ELECTRONIC COOLING FAN circuit breaker on the copilot's wainscot circuit breaker panel.
RESULT: The electronic compartment cooling fan should operate.

b. Man A: Turn the CABIN ALT selector, on the cabin pressure controller, to -1,000 feet. Turn the RATE control to the "12 o'clock" position. Allow the cabin pressure to stabilize at -1,000 (±200) feet.
Man B: Monitor the cabin altitude indicator.

WARNING

Do not exceed a differential pressure of 6 PSI or a cabin rate of descent of 1,000 feet per minute.

RESULT: During this pressure change, the cabin differential pressure should increase to and stabilize at approximately 1 PSI. The cabin rate of descent should be approximately 500 feet per minute. On airplanes 5001 thru 5065, not having Service Bulletin 329-216 incorporated, before the differential pressure reaches 0.5 PSI, the electronics compartment cooling fan should stop.

WARNING

If the cabin differential pressure reading is incorrect, do not attempt to adjust the cabin pressure controller to obtain the correct reading. All controller adjustments must be made on a calibrated test bench.

c. Man A: Turn the cabin altitude selector rapidly to 1,000 feet or more.
Man B: Observe the cabin rate of climb indicator and the cabin differential pressure gage.
Man C: Observe the electronic cooling flow regulator valve in the nose gear wheel well.
RESULT: The cabin rate of climb indicator should show a stabilized reading of approximately 500 feet per minute "UP" until the indicated differential pressure falls to approximately 0.5 PSI. The flow regulator valve should be fully open. On airplanes 5001 thru 5065, not having Service Bulletin 329-216 incorporated, the electronics compartment cooling fan should start before the differential pressure reaches zero. On airplanes 5066 and subsequent, and those having Service Bulletin 329-216 incorporated, the electronics compartment cooling fan should run continuously.

d. Prepare to pressurize the fuselage. Check that the area around the airplane is cleared of personnel and ground equipment, and that only personnel engaged in the test are in the airplane.

e. Man B: Monitor the cabin differential pressure gage.

21.1.4.3 (Cont).

Man A: Close the isobaric shutoff test valve, and check that the differential shutoff valve is still open. On airplanes 5066 and subsequent, and those having Service Bulletin 329-216 incorporated, open the ELECTRONIC COOLING FAN circuit breaker on the copilot's wainscot circuit breaker panel.
Man C: Observe the electronics cooling flow regulator valve in the nose gear wheel well.
RESULT: Cabin differential pressure should increase to and stabilize at 8.8 (±0.4) PSI. The safety valves shall remain closed. The electronic cooling flow regulator valve should be near the closed position.

WARNING

Do not permit the cabin differential pressure to exceed 10 PSI at any time during this or other steps requiring high pressures. If differential pressure exceeds 9.7 PSI, immediately open both test valves in the baggage compartment, to prevent over-pressurization.

f. **Man A: Close the differential shutoff test valve in the baggage compartment.**
Man B: Continue to monitor the differential pressure gage.
RESULT: Cabin pressure should increase rapidly until the safety valves open. After the safety valves have opened, the differential pressure should stabilize at 9.3 (±0.4) PSI.

Note

For airplanes 5126 and up and airplanes modified by Service Bulletin 329-259, if it appears during this check that only one safety valve is opening, the other safety valve can be checked as follows:

(1) Remove the filter/orifice assembly from the head chamber of the safety valve that did open.

(2) Repeat check per paragraph 21.1.4.3.f. Monitor cabin differential pressure closely. Do not exceed 10 PSI.

Note

The CABIN PRESS HIGH warning light might or might not go on during this operation. Either case is acceptable.

WARNING

If the safety valves do not maintain the differential pressure within the limits, do not attempt to adjust the safety valve release pressure. This adjustment must be made on a calibrated test bench.

g. **Man A: Place the NORMAL-DEPRESS switch, on the pressurization control panel, in the "DEPRESS" position, and hold the RAM AIR control switch in the "OPEN" position for approximately 10 seconds.**
Man B: Start the stop watch when cabin differential pressure reaches 7.0 PSI, and check the time required for differential pressure to decrease from 7.0 to 5.0 PSI.
RESULT: Air flow from all outlets should cease, and cabin pressure should decrease. Reduction in cabin differential pressure from 7.0 PSI to 5.0 PSI should require more than 24 seconds.

Note

If cabin differential pressure decreases from 7.0 PSI to 5.0 PSI in less than 24 seconds, excessive compartment leakage is indicated. Locate and repair leaks in accordance with Chapter 51 and perform an accurate leak check in accordance with paragraph 21.1.4.4 after repairs are made.

h. Man A: Return the NORMAL-DEPRESS switch to the "NORMAL" position when cabin differential pressure reaches 5.0 PSI. Assure that cabin altitude set on controller is -1000 feet. Open the isobaric shutoff test valve. Open the MANUAL DEPRESS CONTROL needle valve enough to depressurize the airplane at a rate of not more than 2,000 feet per minute of cabin rate of climb.
RESULT: Cabin rate of climb should be controllable by varying the MANUAL DEPRESS CONTROL needle valve opening.

CAUTION

In steps (i) through (l) proceed as rapidly as possible to prevent possible damage to the ducts. Do not allow the system to remain on for more than 30 seconds at a time on the ground.

i. Man A: Place the EMER PRESS SYSTEM switch, on the pressurization control panel, in the "ON" position.
Man B: Check the airflow from all outlets.
Man C: Observe the four bleed air shutoff valves in the fuselage aft compartment.
RESULT: All four bleed air shutoff valves should close, and the emergency pressurization shutoff valve and the emergency pressurization ram air control should open. This is indicated by complete stoppage of airflow from all cabin right-hand side outlets and gaspers, from all cabin left-hand side gaspers, and from all the flight station outlets, gaspers, and demisters. Airflow should continue only from the cabin left-hand side outlets.

21.1.4.3 (Cont).

j. Man A: Hold the EMER PRESS SYSTEM TEMP switch in the "INCR TEMP" position.
Man C: Observe the ram air control actuator in the fuselage aft compartment.
RESULT: The emergency pressurization ram air control actuator should slowly retract to the closed position and hold.

k. Man A: Hold the EMER PRESS SYSTEM TEMP switch in the "DECR TEMP" position.
Man C: Observe the ram air control actuator.
RESULT: The emergency pressurization ram air control actuator should slowly extend to the open position and hold.

l. Man A: Hold the EMER PRESS SYSTEM TEMP switch in the "INCR TEMP" position, and release it after a brief period.

Man C: Observe the ram air control actuator.

RESULT: The actuator should retract to a partially extended position and hold when the switch is released.

m. Man A: Hold the EMER PRESS SYSTEM TEMP switch in either "INCR TEMP" or "DECR TEMP" position, and while the control is moving, place the EMER PRESS SYSTEM switch in the "OFF" position.

21.1.4.3 (Cont).

Man C: Observe the ram air control actuator. RESULT: The ram air control actuator should retract to the closed position and hold.

n. Man A: Hold the EMER PRESS SYSTEM TEMP switch alternately in the "INCR TEMP" and "DECR TEMP" positions several times.
Man C: Observe the ram air control actuator. RESULT: The ram air control actuator should remain in the closed (retracted) position and should not be affected by any movement of the switch.

o. Return all system controls to normal. Disconnect external electrical power. Shut down and disconnect the compressor, if it was used, or shut down the engines.

WARNING

Secure the isobaric shutoff test valve and the differential shutoff test valve handles in the open (vertical) position by lock-wiring the handles to the lockwire securing clip to the frame at fuselage station 238.

21.1.4.4 CABIN UNCONTROLLED LEAKAGE. If cabin leakage is excessive when checked in accordance with paragraph 21.1.4.3, and/or after repairs, the following leak check should be conducted:

WARNING

When the fuselage is pressurized, major damage can occur with possible injury to personnel if it is struck by any object such as ground equipment. Only personnel actually engaged in the test should be allowed in or near the airplane. Personnel in the airplane should have medical clearances for work in pressurized areas.

a. Close the isobaric and differential shutoff test valves; plug the toilet venturi and the electronic cooling fan outlet. Close all doors and windows.

b. Connect a source of air pressure to the ground air conditioning connection (see figure 21-3) equipped with a flowmeter suitable for measuring the air flow into the cabin.

CAUTION

Do not allow cabin differential pressure to exceed 10.0 PSI.

c. Pressurize the fuselage at a rate not to exceed 1,000 feet per minute until the cabin differential reaches 8.8 PSI and stabilizes for at least three minutes. A stabilized pressure is indicated by the cabin rate of climb indicator indicating zero change.

d. The flowmeter indication should not exceed 19.0 pounds per minute flow for a three-minute period while differential pressure is maintained at 8.8 PSI. If the flowmeter is calibrated to read cubic feet per minute, conversion to pounds per minute is as follows:

$$\text{FLOW IN LBS PER MIN} = 2.7 \frac{P}{T} \text{ (FLOW IN CU FT PER MIN)}$$

WHERE P = CABIN PRESSURE (PSI ABSOLUTE)

T = CABIN TEMPERATURE (°RANKINE)

e. Reduce the pressure to zero at a rate not to exceed 3,000 feet per minute, and restore the system to normal.

WARNING

Open and secure the isobaric and differential shutoff test valves in the open position (handles vertical) by lockwiring the handles to the clip on fuselage station 238 frame.

21.1.4.5 LEAK TESTING CONTROL TUBING. This test should be accomplished subsequent to any component or tube replacement in the pressure control system. A vacuum source capable of 20 inches of mercury is required with a suction cup and a suitable shutoff valve such that when the shutoff valve is closed the pump is isolated from the tubing circuit being tested.

CAUTION

During this test, allow the vacuum to be increased slowly AND DO NOT EXCEED 20 inches of mercury as damage to components may result.

a. Disconnect and plug the two low pressure tubes at the jet pump.

b. Close the differential and isobaric shutoff test valves, and assure that the manual depressurization needle valve is closed. Position the rate control knob on the pressure controller to "MAX".

Note

This is a progressive-type check, and if a leak is detected in a segment of tubing under test, it must be isolated and repaired before proceeding to the next step.

21.1.4 5 (Cont).

c. Attach the suction cup of the vacuum pump over the pressure controller atmosphere sensing port, only (uppermost port on right side of fuselage at fuselage station 294, water line 80, butt line 37 right). DO NOT allow the suction cup to extend over the adjacent ports.

d. Slowly apply a negative pressure (partial vacuum) of 15 inches of mercury. Close the vacuum supply valve. There should be no change in the indication during a two-minute test period.

e. Open the differential shutoff test valve and slowly apply a negative pressure (partial vacuum) of 15 inches of mercury. Close the vacuum supply valve on the vacuum pump. There should be no change in the negative indication during a two-minute test period.

f. Slowly increase the negative pressure (partial vacuum) until a slight vacuum is detected at the pressure controller inbleed filter on the overhead panel in the flight station. This flow condition should occur at 17 to 19 inches of mercury. This opens the differential metering valve to expose the lines from the controller to the isobaric shutoff test valve, from the controller to the pneumatic relay, and to the manual depressurization valve for test. Cover the filter inlet or remove the filter, plug the line, slowly apply 20 inches of mercury negative pressure (partial vacuum), and close the vacuum supply valve. There should be no change in the negative pressure indication during a two-minute test period.

g. On airplanes 5046 and up, and on airplanes 5001 thru 5045 on which Service Bulletin 329-61 has been incorporated, slowly open the isobaric shutoff test valve, and increase negative pressure (partial vacuum) to 20 inches of mercury. There should be a slight inflow of air at the filter on the pneumatic relay. Tape or plug the filter opening, and increase the negative pressure to 20 inches of mercury. The outflow valves should be fully open. Close the vacuum supply valve. The negative pressure should not drop more than 1/2 inch of mercury per minute. (This is normal leakage through the clearance provided on the outflow valve poppet guide shaft.)

h. On airplanes 5001 thru 5045 except those on which Service Bulletin 329-61 is incorporated, slowly open the isobaric shutoff test valve and increase negative pressure (partial vacuum) to 20 inches of mercury. There should be no change in negative pressure during a two-minute test period. Slowly open the manual depressurization valve and increase negative pressure (partial vacuum) to 20 inches of mercury. There should be a slight inflow of air at the filter in the pneumatic relay. Tape or plug the filter opening and increase the negative pressure to 20 inches of mercury. The outflow valves should be fully open. Close the vacuum supply valve. The negative pressure should not drop more than 1/2 inch of mercury per minute. (This is normal leakage through the clearance provided on the outflow valve poppet guide shaft.)

i. On Airplanes 5001 through 5125, not modified by Service Bulletin 329-259, attach the suction cup of the vacuum pump over the safety valve static port only (uppermost port on left side of fuselage at fuselage station 294, water line 80, butt line 37 left). DO NOT allow the suction cup to extend over the adjacent ports. Slowly apply a negative pressure (partial vacuum) of 15 inches of mercury. Close the vacuum supply valve. There should be no change in negative pressure during a two-minute test period. On Airplane 5126 and subsequent, and those airplanes modified by Service Bulletin 329-259, the safety valve plumbing does not require leak testing.

21.1.5 TROUBLESHOOTING.

21.1.5.1 GENERAL. Three basic functions which affect cabin pressure are: (1) cabin air inflow, (2) cabin air outflow, and (3) cabin air outflow control. When a cabin pressure malfunction occurs frequently, the immediate reaction is that the cabin air outflow control system is malfunctioning, and components are changed indiscriminately in an attempt to correct a discrepancy that actually may exist elsewhere. The following checks are arranged in a sequence which considers the most likely cause of the malfunction and the simplest check first, then progresses to the more difficult checks and least likely causes. Some checks are applicable only to overpressurization and others only to underpressurization, while others are applicable to both. The check can therefore be eliminated where it is not applicable to the particular malfunction. It is suggested that the procedure be followed in sequence to minimize the need for specialized equipment and use of manpower.

21.1.5.2 AIR INFLOW CHECK.

21.1.5.2.1 BLEED AIR SHUTOFF VALVE OPERATION. (Applicable if malfunction is under pressurization.) With engines operating at 80% RPM or more, operate the temperature control valves to the full open position by placing the FLT STA and CABIN temperature control toggle switches on the air conditioning control panel in the "WARM" position for one minute, then release to "HOLD". Place all bleed air shutoff switches on the bleed air control panel in "SHUTOFF" and air conditioning shutoff switches in "NORMAL". There should be no flow of air into the cabin. Place the ENG NO. 1 bleed air shutoff switch in "NORMAL". Air should flow into the cabin and flight station. Place the switch in "SHUTOFF". Repeat for ENG 2, 3 and 4 valves.

21.1.5.2.2 BLEED AIR FLOW CONTROL VALVE OPERATION. (Applicable if malfunction is underpressurization, overpressurization, or erratic with engine power changes). Position the temperature control valves as in paragraph 21.1.5.2.1, and all bleed air shutoff switches to "NORMAL". Station an observer in the aft equipment compartment in a

21.1.5.2.2 (Cont).

position to observe the flow control valve position indicator. Advance power on at least one engine from idle to approximately 80% RPM. The valve should be fully open at idle, and should move toward the closed position to an intermediate controlling at the higher power setting.

21.1.5.2.3 BLEED AIR DUCT LEAKAGE. (Applicable if malfunction is underpressurization.) This check may be made simultaneously with either of the checks in paragraph 21.1.5.2.1 or 21.1.5.2.2 above. While at least one engine is operating at approximately 80% RPM, check all bleed air supply ducts between the bleed air shutoff valves and the aft pressure bulkhead for excessive leakage.

21.1.5.3 AIR OUTFLOW CHECK.

21.1.5.3.1 ELECTRONICS COOLING FAN AND FLOW REGULATOR OPERATION. (Applicable if the malfunction is underpressurization or erratic operation.) Close all cabin doors and windows and remove the access panel forward of the nose landing gear wheel well. Advance power on at least one engine to approximately 80% RPM. Engage the ELECT COOLING FAN circuit breaker, select minus 1,000 feet and "MAX" rate on the pressure controller. On aircraft 5001 through 5065, not having Service Bulletin 329-216 incorporated, the fan should stop and the regulator poppet valve should move toward the closed position. On aircraft 5066 and subsequent, and those having Service Bulletin 329-216 incorporated, disengage the ELECT COOLING FAN circuit breaker. The fan should stop and the regulator poppet valve should move toward the closed position.

21.1.5.3.2 CABIN UNCONTROLLED LEAKAGE. (Applicable if the malfunction is underpressurization.) Perform cabin leakage test in accordance with paragraph 21.1.4.4.

21.1.5.4 PRESSURIZATION CONTROL SYSTEM CHECK.

21.1.5.4.1 GENERAL. The checks specified in this section should be made regardless of the type of malfunction if all the applicable preceding checks have been made and no discrepancy found. Before proceeding to the component isolation procedure in paragraph 21.1.5.4.3, the system tubing must be leak checked. Leaks in the tubing will cause the system components to respond erratically or sluggishly when otherwise the components are in good working order.

21.1.5.4.2 TUBING LEAK CHECK. (Perform this check prior to use of the components isolation procedure in 21.1.5.4.3 below.) Perform a leak check of the pressure control system tubing in accordance with paragraph 21.1.4.5.

21.1.5.4.3 COMPONENT MALFUNCTION ISOLATION. Use the procedure shown below if all the above checks have disclosed no discrepancy, or if the system still malfunctions in any way after correction of discrepancies detected by the above procedures. After a component has been isolated by the following procedures as being the cause of a malfunction, or if two or more components are indicated by the isolation procedure, the components may be checked individually by the procedures specified in paragraph 21.1.5.4.4.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
CABIN PRESSURE LOWER THAN DESIRED AS SET ON CONTROLLER.		
Differential pressure gage malfunction.	Check gage readings against a calibrated gage.	Replace gage.
Pressure controller setting incorrect.	Rotate cabin altitude selector knob clockwise to 8,000 feet. If a stop is contacted at approximately 6,000 feet and further movement is restricted, the setting is incorrect.	Rotate cabin altitude selector knob counterclockwise one complete revolution and continue rotation until a stop is contacted. From this point rotate selector knob clockwise to the desired setting.
Pressure controller response to pressure changes is sluggish due to clogged filters.	Remove and check external filter in controller inbleed line on overhead panel.	Clean filter in accordance with paragraph 21.1.7.3 or replace.
Pressure controller malfunctioning.	Check in accordance with paragraph 21.1.5.4.4.	Replace controller.
Pneumatic relay malfunction.	Check in accordance with paragraph 21.1.5.4.4.	Replace relay.
Manual depressurization needle valve open or leaking.	Rotate valve clockwise to close.	Close valve or replace if leaking.
Excessive flow through electronics cooling flow regulator valve.	Check valve operation in accordance with paragraph 21.1.4.3(a, b, c, d, and e).	Remove obstruction from valve actuator pressure tube or replace fan and flow regulator assembly. Replace filter element, when applicable.

21.1.5.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
CABIN PRESSURE HIGHER THAN DESIRED AS SET ON CONTROLLER.		
Differential pressure gage malfunction.	Check gage readings against a calibrated gage.	Replace gage.
Excessive inflow of cabin air to control chamber of pressure controller.	Check for absence of external filter or broken controller housing.	Install filter or replace controller.
Jet pump outlet blocked.	Check jet pump outlet.	Clear jet pump outlet or replace jet pump.
Bleed air pressure regulator blocked or malfunctioning.	Check in accordance with paragraph 21.1.5.4.4.	Clear obstruction or replace bleed air pressure regulator.
Isobaric shutoff valve closed.	Check position of shutoff valve.	Open valve and safety in accordance with paragraph 21.1.3.9.
Differential shutoff valve closed.	Check position of shutoff valve.	Open valve and safety in accordance with paragraph 21.1.3.10.
Pneumatic Relay Malfunction.	Check in accordance with paragraph 21.1.5.4.4.	Replace relay.
EMERGENCY PRESSURIZATION SYSTEM INOPERATIVE.		
Lack of power to switch or defective switch.	Check circuit and switch continuity.	Replace switch or repair circuit as required.
Defective pressurization bleed air shutoff relay.	Check circuit and relay continuity.	Replace relay or repair circuit as required.
Defective emergency pressurization shutoff valve.	Check circuit continuity to valves.	Replace valve or repair circuit as required.
EMERGENCY PRESSURIZATION SYSTEM TEMPERATURE CONTROL INOPERATIVE.		
Defective emergency pressurization relay.	Check circuit and relay continuity.	Replace relay or repair circuit as required.
Emergency pressurization system switch defective.	Check circuit and switch continuity with switch in "INCR," "DECR," and neutral positions.	Replace switch or repair circuits as required.
Ram air control actuator defective.	Check circuit continuity to actuator.	Replace ram air control actuator or repair circuits as required.

21.1.5.4.4 VERIFICATION OF COMPONENT MALFUNCTION. After the component malfunction isolation procedure of paragraph 21.1.5.4.3 above has indicated malfunction of a component or components, the following checks may be made to verify the malfunction and/or further isolate the component if more than one is indicated:

CAUTION

Do not attempt to adjust any of the components on the airplane. This must be done only by an authorized overhaul station.

21.1.5.4.4.1 PRESSURE CONTROLLER. Install a vacuum gage, calibrated in inches of water, in the tubing from the controller to the pneumatic relay by installing a tee in the line at the most convenient location. Use a minimum length of tubing to connect the tee into the existing tubing and to connect the gage to the tee. With at least one engine running at approximately 80% RPM and the airplane doors and windows closed to pressurize, select various cabin altitudes from field elevation to minus 1,000 feet on the controller. The gage should stabilize at 6 to 8 inches of water below cabin pressure at all controller settings.

21.1.5.4.4.2 PNEUMATIC RELAY. Connect a gage similar to the one used in the previous paragraph in the tubing from the pneumatic relay to the outflow valves, and operate the engine in a similar manner. The gage should read from 4 to 6 inches of water below cabin pressure at all cabin altitudes below field elevation.

21.1.5.4.4.3 OUTFLOW VALVES. Connect a vacuum source directly to the tubing connection boss on top of the outflow valve with a gage similar to that used in paragraph 21.1.5.4.4.1, installed to indicate pressure in the line between the vacuum source and the valves. The valves should start to open at approximately 5 inches of water below cabin pressure, and should be fully open at approximately 6 inches. Check freedom of movement of the poppet with the tubing disconnected from the valve. Manually push the valve poppet fully open, and allow the spring to return it to the closed position. Movement should be steady with no indication of dragging, binding, or erratic movement.

21.1.5.4.4.4 JET PUMP. The jet pump should be inspected thoroughly with tubing disconnected for evidence of obstructions in the exit, high pressure inlet and two low pressure inlets. The sizes of orifices are: high pressure (bleed air) inlet - .080 inches dia; ATMOS 3 (to controller) - .100 inches dia; Opposite to ATMOS 3 (connected to relay) - .20 inches.

21.1.5.4.4.5 JET PUMP PRESSURE REGULATOR.

a. Connect pressure gages (at least 20 inches of mercury capability) in the inlet line to the regulator and between the regulator and the jet pump. With the engines at idle RPM, the drop in pressure across the regulator should not exceed .75 PSI.

b. With engines operating at approximately 80% RPM, the pressure between the regulator and the jet pump should be 8.25 (±1.25) PSI, and should be steady for all higher power settings.

21.1.6 REMOVAL AND INSTALLATION.

21.1.6.1 GENERAL. Replacement of pressurization system components is straightforward, but to obtain satisfactory system operation the following precautions must be taken:

a. Do not remove temporary caps from lines or ports until just before the connection is to be made. This reduces the possibility of foreign matter entering the system.

b. Flareless tube fittings are used in the system. Tighten these in accordance with the instructions in Section 29.0.

c. Make sure all gaskets, hose, and packings are serviceable and within cure date limitations before installing them.

d. Always perform a leak check of the system tubing in accordance with paragraph 21.1.4.5, and an operational checkout of the system in accordance with paragraph 21.1.4 after replacing components or tubing.

21.1.6.2 REMOVAL OF TUBING. Tubing removal is simple and straightforward, involving only the disconnection of tubing at each end and disconnection of the support clamps. Whenever a tube is disconnected for any reason, a leak check should be made in accordance with paragraph 21.1.4.5 except the check may be modified as required to check only the area disconnected.

21.1.6.3 INSTALLATION OF TUBING. Refer to Section 29.0 for information on installation of the flareless tube fittings.

21.1.6.4 REMOVAL OF OUTFLOW VALVES.

a. On airplanes 5001 thru 5045 not having Service Bulletin 329-61 incorporated, obtain access to the outflow valves by removing the walkway floor panel above the valves at approximately fuselage station 340. On airplanes 5046 and up and those with Service Bulletin 329-61 incorporated, obtain access to the valves on the forward face of the aft pressure bulkhead by removing interior furnishings as necessary.

b. Disconnect the secondary control pressure line from the valve elbow. It may also be necessary to disconnect the tube at the other end to obtain clearance for withdrawing the flareless fitting.

c. Cap the secondary control pressure line end. Remove the outflow valve from the fixed flange by loosening the clamp.

21.1.6.5 INSTALLATION OF OUTFLOW VALVES.

a. Install the elbow in the replacement valve port with a serviceable gasket and nut.

b. Position the valve, the clamp, and a serviceable valve gasket on the fixed flange in the fuselage bottom.

c. Rotate the valve as necessary to align the elbow with the end of the secondary control pressure line, and connect the line to the elbow. Tighten the tube nut in accordance with the instructions for flareless tube fittings in Section 29.0.

d. Tighten the clamp T-bolt to 40 inch-pounds, and tap gently all around the clamp with a soft-faced mallet, to seat it. Retighten the T-bolt to 40 inch-pounds.

e. Leak-check tubing in accordance with paragraph 21.1.4.5 (a, b, c, & g or h).

21.1.6.6 REMOVAL OF SAFETY VALVES.

a. Disconnect the static pressure line from the affected safety valve, through the circular opening in the valve support bowl, which extends into the aft fuselage compartment from the pressure bulkhead.

b. Station a man forward of the aft pressure bulkhead to remove interior furnishings as necessary to gain access to the valve and to support the valve. Remove the hardware securing the valve support flange to the bowl, from the aft side of the pressure bulkhead.

c. Withdraw the valve, flange, gasket, and clamp assembly forward into the cabin.

d. Separate the valve from the flange by loosening the clamp. Remove the reducer fitting and gasket from the valve static pressure port. Cap the static pressure line in the aft compartment, and cap the static port of the valve.

21.1.6.7 INSTALLATION OF SAFETY VALVES.

a. Connect the valve to the mounting flange with a serviceable gasket and the clamp. Tighten the clamp to 40 inch-pounds, and tap lightly all around the clamp with a soft-faced mallet, to seat it. Retighten the clamp to 40 inch-pounds.

b. Install the reducer fitting in the valve static pressure port with a serviceable gasket.

c. Position the valve and flange assembly in the valve support bowl, working from the cabin side of the aft pressure bulkhead, and with a serviceable gasket between the flange and the bowl.

d. Position the ring on the aft face of the support bowl, and secure the valve and flange assembly, gasket, and ring to the support bowl with eight screws and washers.

e. Leak-check tubing in accordance with paragraph 21.1.4.5.

f. Check valve operation in accordance with paragraph 21.1.4.

21.1.6.8 REMOVAL OF PRESSURE CONTROLLER.

a. Remove the screws and washers attaching the pressure controller to the overhead control panel, and lower the controller to the extent permitted by the flexible lines connected to it. Identify all lines with tape to facilitate reconnecting them to the replacement controller.

b. Support the controller, and disconnect all the flexible lines from the fittings. Lower the controller clear of the panel. Cap the open ends of the disconnected lines.

c. Remove the packing rings and fittings from the controller, and retain them for use on the replacement unit. Cap all the ports on the controller.

21.1.6.9 INSTALLATION OF PRESSURE CONTROLLER.

a. Install the elbows in the controller ports with nuts and serviceable gaskets. Position the elbows to face in an approximately forward direction when the controller is installed.

b. Support the controller below the overhead panel, and connect the flexible lines to the elbows. Make sure the correct lines are connected to the respective ports. Remove any identification tape which had been applied to the lines.

c. Position the controller on the overhead panel, making sure the flexible lines are not kinked, and secure it to the panel with four screws and four washers.

d. Leak-check tubing connections in accordance with paragraph 21.1.4.5 (a, b, c, d, e and f).

21.1.6.10 REMOVAL OF PNEUMATIC RELAY.

a. On airplanes 5001 thru 5045 not having Service Bulletin 329-61 incorporated, obtain access to the pneumatic relay by removing the walkway floor panel above it at approximately fuselage station 350. On airplanes 5046 and up and those with Service Bulletin 329-61 incorporated, obtain access to the relay on the forward face of the aft pressure bulkhead by removing interior trim as necessary.

b. Disconnect one secondary control pressure line and two suction lines from the union and elbows on the relay.

c. Remove the screws, washers, and nuts securing the relay and external filter assembly to the mounting bracket, and remove the relay and filter assembly.

d. Cap the open ends of the lines in the fuselage.

e. Remove the union, elbows, nuts, and packings from the relay. Cap the relay ports.

21.1.6.11 INSTALLATION OF PNEUMATIC RELAY.

a. Install one reducer and two elbows, with serviceable gaskets, in the relay ports, securing the elbows with nuts. Make sure the elbows face in the correct direction for connection to the tube assemblies in the fuselage.

b. Position the relay on the support bracket, and secure the relay to the bracket with two screws, two nuts, and four washers.

21.1.6.11 (Cont).

c. **Tighten the tube nuts according to the instructions for flareless fitting in Section 29.0.**

d. **Leak-check tubing connections in accordance with paragraph 21.1.4.5**

21.1.6.12 REMOVAL OF MANUAL DEPRESSURIZATION CONTROL VALVE. Remove the pressurization control panel to gain access to the tubing connections. Disconnect the two flexible tubes connected to the valve ports. Remove the two attaching screws from the plastic panel and remove the panel to gain access to the four valve attaching screws. Remove the valve attaching screws and remove the valve. Retain the four screws (MS24693BB270) for reuse if serviceable.

21.1.6.13 INSTALLATION OF MANUAL DEPRESSURIZATION CONTROL VALVE. Attach the valve to the attaching plate with the four screws previously removed. Install the plastic edge lighted panel with two screws (NAS 1100-06-6). Connect the two flexible tubes to the valve ports. Leak-check the pressurization control tubing in accordance with paragraph 21.1.4.5.

21.1.6.14 REMOVAL OF AIR FILTERS.

Note

The filter elements for the controller and pneumatic relay are replaceable. Refer to paragraph 21.1.7.3 for replacement instructions. The filter elements in safety valves are of the permanent type. Refer to paragraph 21.1.7.3 for cleaning instructions. Removal of housing is necessary only if damage or obstruction in parts prevents proper operation.

21.1.6.14.1 REMOVAL OF CONTROLLER EXTERNAL FILTER. Remove the overhead panel assembly in accordance with Chapter 31 to gain access to the filter **support. Disconnect the flexible tube from the elbow fitting. Remove the elbow fitting. Remove two attaching screws from the filter retainer. Remove jam nut from the filter boss and remove the filter from the retainer. Retain serviceable parts for reinstallation. Install protective caps and/or plugs in exposed openings.**

21.1.6.14.2 REMOVAL OF PNEUMATIC RELAY FILTER.

Note

On airplanes 5001 thru 5045 except those on which Service Bulletin 329-61 is incorporated, the filter is located under center aisle floor at fuselage station 350. On airplanes 5046 and up and those on which Service Bulletin 329-61 has been incorporated, the pneumatic relay filter is located on the fuselage station 570 pressure bulkhead.

Obtain access to the filter by removal of the floor panel at fuselage station 350 or interior furnishings at fuselage station 570 as required. Loosen the jamnut on the union fitting installed in the relay housing. Rotate the filter housing as required to disengage the threads. Install protective caps and/or plugs in exposed openings.

21.1.6.14.3 REMOVAL OF SAFETY VALVE FILTER. Remove interior furnishings as required to gain access to the safety valves. Remove the fitting which contains the filter cartridge by rotating counterclockwise as required to disengage threads. Install protective caps and/or plugs in exposed openings.

21.1.6.15 INSTALLATION OF AIR FILTERS.

Note

Refer to paragraph 21.1.7.3 for replacement or cleaning filter elements or cartridge.

21.1.6.15.1 INSTALLATION OF CONTROLLER EXTERNAL FILTER. The overhead panel must be removed and the filter retainer removed in accordance with paragraph 21.1.6.14 to facilitate installation of the filter. Remove jamnut from filter boss and insert boss in hole in filter retainer assembly. Install jamnut. Install the elbow fitting (MS21908D6) and a new MS28778-6 gasket. Install the filter retainer with the two screws and washers. Connect the flexible tube to the elbow. Install the overhead panel assembly. Leak-check tubing in accordance with paragraph 21.1.4.5, and perform an operational checkout in accordance with paragraph 21.1.4.

21.1.6.15.2 INSTALLATION OF PNEUMATIC RELAY FILTER. Install a new MS28778-6 gasket, position the relay to engage the threads of the filter boss with the threads on the relay fitting, and rotate as required to engage. Tighten fitting in accordance with instructions in Section 29. Leak-check system tubing in accordance with paragraph 21.1.4.5, and perform an operational checkout in accordance with paragraph 21.1.4.

21.1.6.15.3 INSTALLATION OF SAFETY VALVE FILTER. Install a new MS28775-012 packing. Position the orifice fitting with the filter and rotate into the threaded boss on the valve housing.

21.1.6.16 REMOVAL OF JET PUMP. The jet pump is located under the center aisle floor at fuselage station 346 on airplanes 5001 thru 5045 except those on which Service Bulletin 329-61 has been incorporated, and is located on the fuselage station 570 bulkhead on airplanes 5046 and up and those on which Service Bulletin 329-61 has been incorporated. Obtain access to the jet pump by removal of the floor panel or interior furnishings as required. Disconnect the three tubes from the jet pump, and plug and/or cap with protective covers.

21.1.6.16 (Cont).

Remove two screws attaching the jet pump flange to the fuselage skin or to the bulkhead, and remove the jet pump. Retain screws for reuse if serviceable.

Note

> The jet pump is installed with sealant between the flange and fuselage skin faying surfaces, therefore slight force may be necessary to separate the surfaces.

21.1.6.17 INSTALLATION OF JET PUMP. Obtain access to the jet pump installed location in accordance with paragraph 21.1.6.16. Clean the jet pump attaching flange and fuselage skin faying surfaces, and apply sealant to surfaces in accordance with Chapter 51. Orient the jet pump with the ATMOS 3 port toward the right-hand side of the airplane, and position it with the air exit and attaching holes aligned with the holes in the fuselage skin. Install the two attaching screws while the sealant is wet.

CAUTION

> DO NOT permit sealant to extrude into the air exit, and make sure the exit and the hole in the fuselage skin align.

Connect the tubing to the jet pump ports. Perform a leak check of the tubing in accordance with paragraph 21.1.4.5 and an operational checkout in accordance with paragraph 21.1.4.

21.1.6.18 REMOVAL OF TEST VALVES. Obtain access to the test valves by removing interior furnishings as necessary at fuselage station 242, water line 90, right side. Remove safety wire from the valve handle. Disconnect the tubes from the two valve ports. Cap and/or plug openings with protective covers. Remove safety wire from the four attaching screws and remove the screws from the threaded holes in valve body. Push the valve outboard, rotating downward as required for the handle to pass through the hole in the mounting bracket, and remove the valve.

21.1.6.19 INSTALLATION OF TEST VALVES. The valve should be open (handle aligned with ports) when installation is made. Position the valve outboard of the mounting bracket with the valve rotated downward, and insert the valve handle into the hole in the mounting bracket. Rotate the valve upward and inboard until the attaching flange is positioned against the outboard side of the mounting bracket. Align the holes, and reinstall the four attaching screws, and safety-wire. Connect the tubing to the two valve ports. Safety-wire the valve handles in the open position to the adjacent clip on the fuselage station 238 frame. Perform a tubing leak check in accordance with paragraph 21.1.4.5.

21.1.6.20 REMOVAL OF JET PUMP PRESSURE REGULATOR. The jet pump pressure regulator is located under the center aisle floor at fuselage station 350 on airplanes 5001 thru 5045 except those on which Service Bulletin 329-61 has been incorporated, and is located on the fuselage station 570 bulkhead on airplanes 5046 and up and those on which Service Bulletin 329-61 has been incorporated. Obtain access to the jet pump pressure regulator by removal of the floor panel or interior furnishings as required. Disconnect the two tubes from the regulator. Install protective covers over exposed openings. Remove the two attaching screws from the mounting base. Retain screws and nuts for reuse if serviceable. Remove the regulator.

21.1.6.21 INSTALLATION OF JET PUMP PRESSURE REGULATOR. Obtain access to the regulator installed location in accordance with paragraph 21.1.6.20. Position the regulator to align the attaching holes with the holes in the mounting bracket and the ports with the tubing, and install the two attaching screws and nuts. Connect the tubing to the inlet and outlet ports. Perform a tubing leak check in accordance with paragraph 21.1.4.5.

21.1.7 SERVICING.

21.1.7.1 ADJUSTMENT. Due to the small movement of the outflow valves, safety valves, and the internal metering valves and diaphragms of the pressure controller and pneumatic relay, adjustment of these units is very critical. Adjustment is therefore not to be considered a function of line maintenance. If adjustment of a unit is found necessary, the unit must be removed and replaced with a serviceable like item, and the removed unit overhauled using calibrated test equipment and qualified personnel. Never attempt adjustment of an installed unit.

21.1.7.2 WATER DRAINAGE. On Airplanes 5001 through 5125, a capped drain plug is provided in the left-hand wheel well for drainage of condensed moisture from the safety valve static line. The amount of moisture collected in this line is a function of the type of climate in which the airplane is operated, therefore the time period between drainings must be established on that basis. It is recommended that the cap be removed from the plug and checked once each day that the airplane is flown until sufficient experience is obtained to establish a satisfactory schedule compatible with the operating environment. Drainage may be discontinued if the airplane has been modified by Service Bulletin 329-259.

21.1.7.3 FILTER CLEANING AND ELEMENT REPLACEMENT.

21.1.7.3.1 CONTROLLER EXTERNAL FILTER. The controller external filter is located in the overhead panel in the flight station. The filter element is composed of glass and Esparto fiber, and cannot be cleaned and reused. The filter element should be replaced at intervals of 500 flying hours or more often as operating experience dictates. To replace the element, it is

21.1.7.3.1 (Cont).

necessary to lower the overhead panel and then remove the filter retainer in accordance with paragraph 21.1.6.14.1. The filter element is supported in a rubber housing. Roll the housing back, lift out the element and replace with a new element. Replace with element part number indicated on the outside of the rubber housing.

21.1.7.3.2 CONTROLLER INTERNAL FILTERS The controller internal filters are composed of cotten gauze sponge and should be replaced at 500-flying-hour intervals. The filters are accessible by removing the controller from its mounting and lowering it until the back of the case is accessible. Remove the hexagon fittings which are labeled "FILTER", and remove the cotton packing. Replace with 3 x 1-1/2 inches gauze sponge 220, 12 ply, Bauer and Black Division, Kendall Corp., or equivalent, install new packings and replace hexagon fittings.

21.1.7.3.3 PNEUMATIC RELAY FILTER. The pneumatic relay filter is the same as the controller external filter described in paragraph 21.1.7.3.1. See paragraph 21.1.6.14.2 for location of filter in the airplane. Obtain access to the filter by removing the floor panel or interior furnishings at fuselage station 570 bulkhead as applicable. Roll the rubber support back as required to lift the element out. Replace with a new element. The part number of the correct replacement element is indicated on the outside of the housing. Replacement should be accomplished at intervals of 500 flying hours, or more often if operating experience indicates the necessity.

21.1.7.3.4 SAFETY VALVE FILTERS.

WARNING

Provide adequate ventilation when using cleaning solvent (Federal Specification P-D-680, Type II). Avoid prolonged breathing of vapors and minimize skin contact.

The safety valve filter is a permanent type, but should be cleaned at intervals of 500 flying hours or more often if experience indicates the necessity for it. Gain access to the safety valves by removing interior furnishings as required at the fuselage station 570 bulkhead. Remove the orifice fitting and filter in accordance with paragraph 21.1.6.14.3. Clean the fitting and filter, using dry cleaning solvent conforming to Federal Specification P-D-680, Type II or equivalent, and reinstall in the safety valve boss.

21.1.7.4 CLEANING VALVE SEATS. The outflow valves and safety valves should be inspected at intervals of 500 flying hours, or more often if experience indicates it is necessary, for evidence of tobacco tars, grease or other foreign deposits on the edges of poppets and at the poppet seats. Wipe the surfaces with a clean, soft, lint-free cloth. To avoid possible damage to diaphragms, DO NOT use compressed air and, if solvent is used, the cloth should be only slightly dampened to prevent liquid or excessive vapor products from reaching the diaphragms. A light coat of parting agent (MIL-S-8660 or equivalent) should be applied to the knife edge of the safety valve poppet and to the rubber safety valve poppet seat.

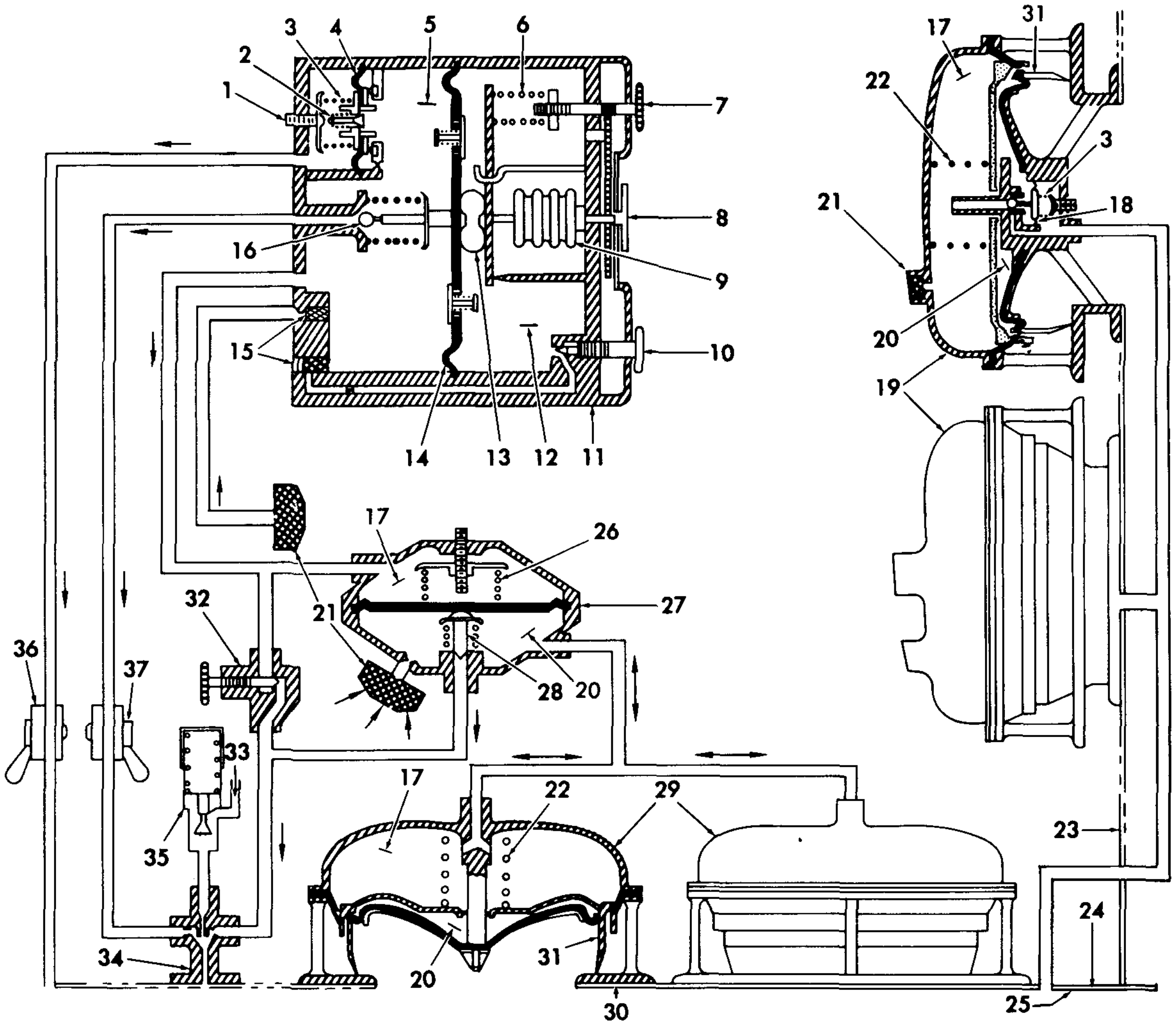
21.1.7.5 CLEANING STATIC PORTS. Obstructed static ports can cause erratic operation of the system and should be avoided by periodic checks and by proper precautions when performing operations such as cleaning and polishing the airplane. Avoid application of any cleaning materials or polish to the ports or the skin surfaces in the immediate vicinity of the ports. If the ports are obstructed, the obstruction should be removed by disconnecting the tubing from the fitting inside the airplane and blowing from inside to outside with compressed air.

21.1.7.6 OVERHAUL. When a component malfunctions and the normal servicing operations described above will not return it to a serviceable condition, or upon completion of the operating time intervals indicated in Chapter 5, the component should be removed from the airplane and overhauled. Overhaul should be accomplished only by qualified personnel, using the required calibrated test equipment in accordance with the equipment manufacturers' instructions.

21.1.7.7 SYSTEM PRESERVATION.

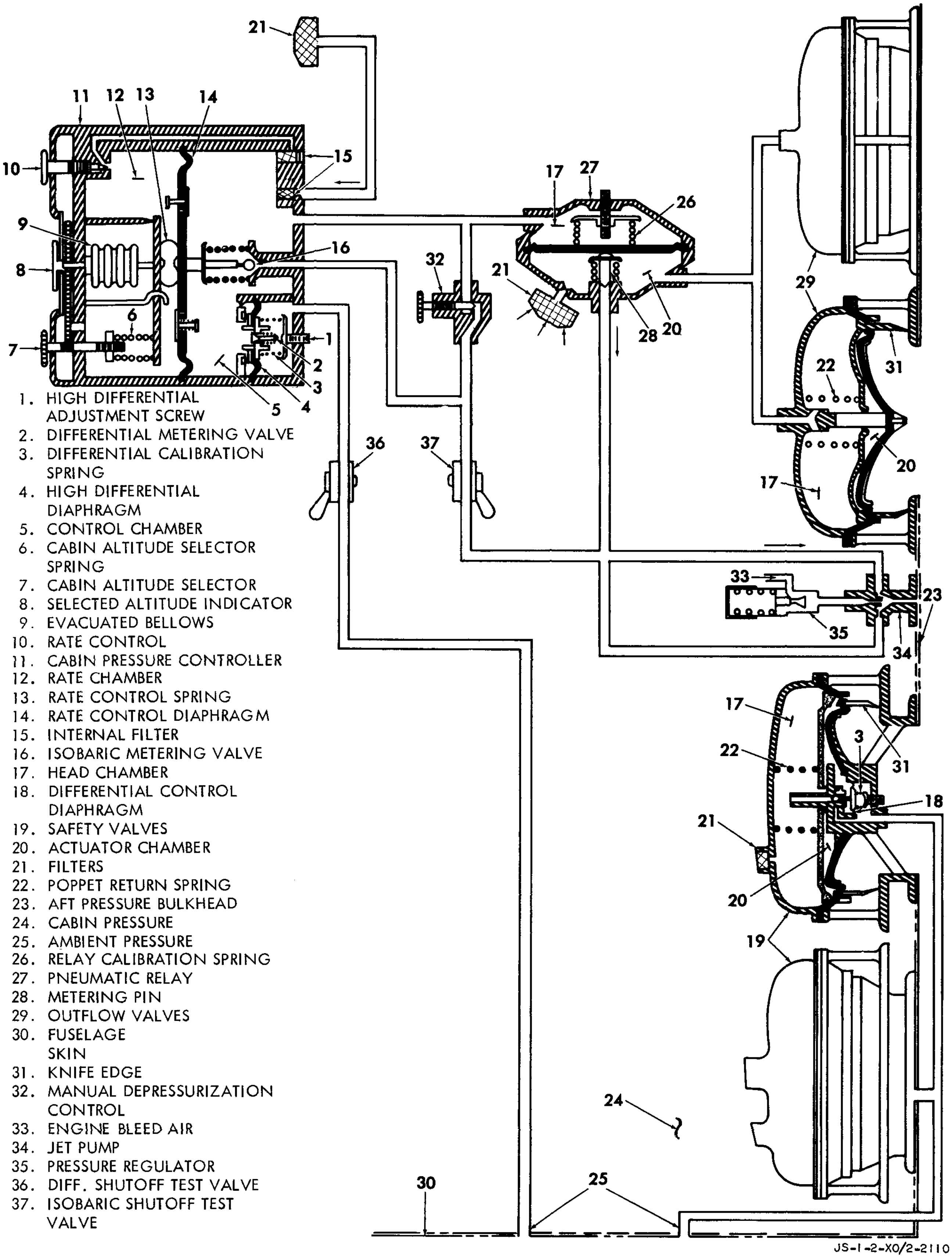
21.1.7.7.1 NORMAL OPERATION. During normal operating condition no special considerations are required to obtain satisfactory performance from the pressure control system.

21.1.7.7.2 STORAGE OF THE AIRPLANE. If the airplane is to be stored for periods in excess of 30 days, the static ports should be covered with water proof paper to prevent the entry of dust, water or other foreign material.



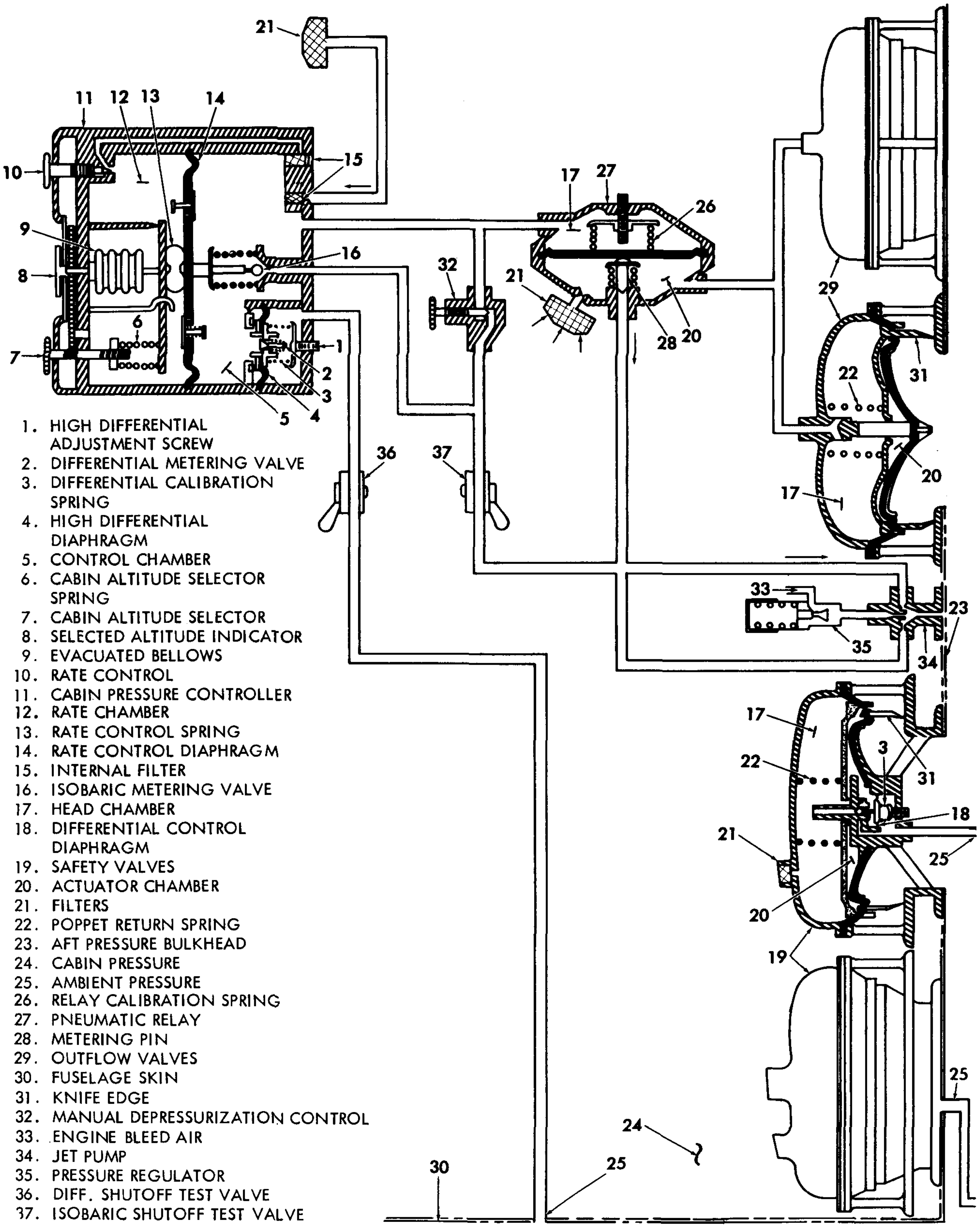
1. HIGH DIFFERENTIAL ADJUSTMENT SCREW
2. DIFFERENTIAL METERING VALVE
3. DIFFERENTIAL CALIBRATION SPRING
4. HIGH DIFFERENTIAL DIAPHRAGM
5. CONTROL CHAMBER
6. CABIN ALTITUDE SELECTOR SPRING
7. CABIN ALTITUDE SELECTOR
8. SELECTED ALTITUDE INDICATOR
9. EVACUATED BELLOWS
10. RATE CONTROL
11. CABIN PRESSURE CONTROLLER
12. RATE CHAMBER
13. RATE CONTROL SPRING
14. RATE CONTROL DIAPHRAGM
15. INTERNAL FILTER
16. ISOBARIC METERING VALVE
17. HEAD CHAMBER
18. DIFFERENTIAL CONTROL DIAPHRAGM
19. SAFETY VALVES
20. ACTUATOR CHAMBER
21. FILTERS
22. POPPET RETURN SPRING
23. AFT PRESSURE BULKHEAD
24. CABIN PRESSURE
25. AMBIENT PRESSURE
26. RELAY CALIBRATION SPRING
27. PNEUMATIC RELAY
28. METERING PIN
29. OUTFLOW VALVES
30. FUSELAGE LOWER SURFACE SKIN
31. KNIFE EDGE
32. MANUAL DEPRESSURIZATION CONTROL
33. ENGINE BLEED AIR
34. JET PUMP
35. PRESSURE REGULATOR
36. DIFF. SHUTOFF TEST VALVE
37. ISOBARIC SHUTOFF TEST VALVE

Figure 21-9. Pressurization System Flow Schematic Diagram (Airplanes 5001 thru 5045 Not Modified by Service Bulletin 329-61)



1. HIGH DIFFERENTIAL ADJUSTMENT SCREW
2. DIFFERENTIAL METERING VALVE
3. DIFFERENTIAL CALIBRATION SPRING
4. HIGH DIFFERENTIAL DIAPHRAGM
5. CONTROL CHAMBER
6. CABIN ALTITUDE SELECTOR SPRING
7. CABIN ALTITUDE SELECTOR
8. SELECTED ALTITUDE INDICATOR
9. EVACUATED BELLOWS
10. RATE CONTROL
11. CABIN PRESSURE CONTROLLER
12. RATE CHAMBER
13. RATE CONTROL SPRING
14. RATE CONTROL DIAPHRAGM
15. INTERNAL FILTER
16. ISOBARIC METERING VALVE
17. HEAD CHAMBER
18. DIFFERENTIAL CONTROL DIAPHRAGM
19. SAFETY VALVES
20. ACTUATOR CHAMBER
21. FILTERS
22. POPPET RETURN SPRING
23. AFT PRESSURE BULKHEAD
24. CABIN PRESSURE
25. AMBIENT PRESSURE
26. RELAY CALIBRATION SPRING
27. PNEUMATIC RELAY
28. METERING PIN
29. OUTFLOW VALVES
30. FUSELAGE SKIN
31. KNIFE EDGE
32. MANUAL DEPRESSURIZATION CONTROL
33. ENGINE BLEED AIR
34. JET PUMP
35. PRESSURE REGULATOR
36. DIFF. SHUTOFF TEST VALVE
37. ISOBARIC SHUTOFF TEST VALVE

Figure 21-10. Pressurization System Flow Schematic Diagram (Airplanes 5046 thru 5125 and Airplanes Modified by Service Bulletin 329-61)



1. HIGH DIFFERENTIAL ADJUSTMENT SCREW
2. DIFFERENTIAL METERING VALVE
3. DIFFERENTIAL CALIBRATION SPRING
4. HIGH DIFFERENTIAL DIAPHRAGM
5. CONTROL CHAMBER
6. CABIN ALTITUDE SELECTOR SPRING
7. CABIN ALTITUDE SELECTOR
8. SELECTED ALTITUDE INDICATOR
9. EVACUATED BELLOWS
10. RATE CONTROL
11. CABIN PRESSURE CONTROLLER
12. RATE CHAMBER
13. RATE CONTROL SPRING
14. RATE CONTROL DIAPHRAGM
15. INTERNAL FILTER
16. ISOBARIC METERING VALVE
17. HEAD CHAMBER
18. DIFFERENTIAL CONTROL DIAPHRAGM
19. SAFETY VALVES
20. ACTUATOR CHAMBER
21. FILTERS
22. POPPET RETURN SPRING
23. AFT PRESSURE BULKHEAD
24. CABIN PRESSURE
25. AMBIENT PRESSURE
26. RELAY CALIBRATION SPRING
27. PNEUMATIC RELAY
28. METERING PIN
29. OUTFLOW VALVES
30. FUSELAGE SKIN
31. KNIFE EDGE
32. MANUAL DEPRESSURIZATION CONTROL
33. ENGINE BLEED AIR
34. JET PUMP
35. PRESSURE REGULATOR
36. DIFF. SHUTOFF TEST VALVE
37. ISOBARIC SHUTOFF TEST VALVE

Figure 21-10A. Pressurization System Flow Schematic Diagram (Airplanes 5126 and up and Airplanes Modified by Service Bulletin 329-259)

SECTION 21.2
ELECTRONIC EQUIPMENT COOLING SYSTEM

DESCRIPTION AND OPERATION 21-53
  General 21-53
  Normal Operation 21-53

OPERATIONAL CHECKOUT 21-54

TROUBLESHOOTING 21-54

REMOVAL AND REPLACEMENT 21-54A
  Removal of Electronics Cooling Fan and Flow Regulator Assembly 21-54A
  Installation of Electronics Cooling Fan and Flow Regulator Assembly 21-55
  Removal of Flow Limiter 21-55
  Installation of Flow Limiter 21-56

SERVICING 21-56
  Periodic Maintenance 21-56
  Adjustments 21-56
  Overhaul 21-56
  Flow Regulator Filter 21-56
  Replacement 21-56

SYSTEM PRESERVATION 21-56
  Normal Operation 21-56
  Airplane Storage 21-56

FIGURES
  21-11. Electronic Cooling Schematic Diagram (Airplanes 5001 thru 5065) 21-57
  21-11A. Electronic Cooling Schematic Diagram (Airplanes 5066 and Subsequent) 21-58
  21-11B. Electronic Cooling Flow Limiter 21-58A

21.2.1 DESCRIPTION AND OPERATION.

21.2.1.1 GENERAL. During normal operation, the nose electronics compartment receives air from the cabin through a flow limiter located in the forward pressure bulkhead. The nose compartment pressure is nominally the same as the cabin. The forward pressure bulkhead and flow limiter are only to maintain cabin pressure if a failure occurs in the nose which causes loss of cabin pressure. (See figure 21-11B.) The equipment in the nose compartment is cooled by exhausting air overboard through the equipment. During ground operation, the flow of air overboard is induced by a fan which is part of the electronics cooling fan and flow regulator unit. During pressurized flight, the flow of air through the equipment is induced by the pressure differential between the nose compartment and ambient pressure. It is controlled by a flow regulator, which is a part of the electronics cooling fan and flow regulator unit. On airplanes 5001 thru 5065, not having Service Bulletin 329-216, when the difference between the air pressure in the nose and ambient air pressure becomes less than a preset amount, a differential pressure switch is closed to energize the fan. On airplanes 5001 thru 5065, not having Service Bulletin 329-216 incorporated, the fan, the flow regulator, and the pressure switch are incorporated into a single unit, and operation is completely automatic. On airplanes 5066 and subsequent, and those having Service Bulletin 329-216 incorporated, the pressure switch is omitted. On these airplanes the fan receives power through the NLG down-lock switch so that the fan operates continuously during powered ground operation. On all airplanes, the fan receives electrical power from the main 28-volt DC bus, through a circuit breaker on the copilot's wainscot circuit breaker panel. The unit is equipped with a thermal switch to prevent damage to internal wiring.

21.2.1.2 NORMAL OPERATION. The amount of air flowing through the electronic equipment is dependent upon the differential pressure across the equipment and the size of the air passage. During pressurized operation, the flow regulator valve will be closed or opened as necessary to maintain the required flow for pressure differential. This is accomplished by the action of the flow regulator metering pin (See figure 21-11). The metering pin opens to admit cabin pressure to the actuator chamber of the flow regulator valve to close the valve poppet; or closes to restrict cabin pressure from the actuator chamber, allowing springs to open the poppet. On airplanes 5066 and

21.2.1.2 (Cont).

subsequent, and those having Service Bulletin 329-216 incorporated, the cabin pressure supply line contains a *filter* (*see* paragraph 21.2.5.4). The metering pin is controlled by the action of the regulator diaphragm. The diaphragm is spring loaded by a calibration spring to maintain the metering pin in the closed position. This chamber of the regulator is connected by a sensing tube to the downstream side of the electronics equipment (plenum) while the other chamber (opposite side of the diaphragm) is connected to the upstream side of the equipment (electrical compartment). When the pressure in the plenum becomes less than the pressure in the electronics compartment by a pre-set amount, the resulting differential pressure across the diaphragm is sufficient to overcome the force of the calibration spring and open the regulator metering pin. The actuator chamber of the valve assembly is thus exposed to higher pressure through the tube which is connected to cabin pressure, causing the valve to close to reduce flow and decrease pressure *differential across the electronics* equipment. The size of the air passage is varied with different equipment configurations to provide the correct cooling requirements. (Refer to Chapter 34.) On airplanes 5001 thru 5065, not having Service Bulletin 329-216 incorporated, the pressure port of the pressure switch is connected to the electronics compartment pressure sensing tube, so that when the pressure upstream of the equipment becomes low enough that differential flow cannot be maintained the switch closes to energize the circuits and start the fan operation.

21.2.2 OPERATIONAL CHECKOUT. Perform the applicable portions of the cabin pressure control operational checkout procedure in accordance with paragraph 21.1.4.

21.2.3 TROUBLESHOOTING.

DISCREPANCY	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Excessive flow overboard through electronics cooling fan and flow regulator unit.	Obstruction in cabin pressure line to actuator chamber of valve. Filter clogged.	Pressurize the airplane to maximum differential (8.80 PSI) and check position of valve (should be nearly closed).	Replace tube or remove obstruction from tube. Replace filter element when applicable.
	Regulator metering pin stuck.	Same as above.	Replace regulator assembly.
Fan operation intermittent.	Thermal protection device tripped.	Not required.	Replace fan assembly.
	Defective pressure switch. (Airplanes 5001 thru 5065 without Service Bulletin 329-216 incorporated only)	Check that switch closes with 9 (±2) inches water pressure at pressure port and opens with 5 (±2) inches water.	Replace pressure switch.
Fan will not operate.	Defective pressure switch. (Airplanes 5001 thru 5065, without Service Bulletin 329-216 incorporated only)	Same as above.	Same as above.
	Defective circuit.	Perform continuity check of fan circuits.	Repair circuits as required.
Unusual pulsating airflow noise i.e., "choo choo."	Excessive airflow supplied to fan inlet.	Check for proper installation of duct access panel. Check for missing sealant in plenum holes and joints. Check for looseness of flexible ducts to plenum joints.	Correct access panel installation and seal plenum as required.

21.2.4 REMOVAL AND REPLACEMENT.

21.2.4.1 REMOVAL OF ELECTRONICS COOLING FAN AND FLOW REGULATOR ASSEMBLY.

a. Open the nose cone assembly in accordance with Chapter 34.

b. Disconnect the lower flexible duct from the access plate on the fuselage station 164 bulkhead (electronic equipment exhaust air duct).

c. Remove the nine attaching screws and remove the access plate.

21.2.4.1 (Cont).

Note

The upper edge of plate is retained under a lip on the upper duct skin. With screws removed, a slight downward force will disengage the plate.

d. Remove eight attaching screws through the opening provided when the plate is removed. Retain all grommets, gaskets, washers, screws, etc. for re-use if serviceable. Refer to paragraph 21.2.4.2 for information to determine if synthetic rubber gaskets and washers are serviceable.

e. Remove the access panel from the bottom of the fuselage between fuselage stations 178.75 and 199.00.

f. Disconnect the two flexible hoses from the unit.

g. Disconnect the electrical connector from the unit.

h. Disconnect the bonding jumper.

i. Support the unit and remove the two attaching screws from the upper hanger support. Retain grommets, gaskets, washers, screws, nuts, etc., for re-use if serviceable.

j. Lower the unit aft and downward from the airplane.

CAUTION

Use care to prevent damage to the regulator sensing tube at the fan inlet as the unit is removed.

21.2.4.2 INSTALLATION OF ELECTRONICS COOLING FAN AND FLOW REGULATOR ASSEMBLY.

Note

The unit is supported against the fuselage structure with gaskets and grommets installed between the metal faying surfaces to aid in vibration isolation and noise abatement. Refer to figure 21-11 for correct placement of these parts. Do not over torque the screws. Minimum allowable thickness of gaskets when re-used is as follows:

Part No.	Material	Thickness of New Part	Min. Allow. Thk. (Comp.)
JQ 912-9	AMS 3198 Synthetic Rubber	3/8-inch	1/4-inch
JQ 912-10	AMS 3208 Synthetic Rubber.	1/8-inch	1/16-inch

a. Gain access for installation in accordance with paragraph 21.2.4.1.

b. Make sure serviceable grommets are installed in the two upper hanger attaching holes and in the eight attaching holes in the fuselage station 164 bulkhead.

c. Place the unit in position with the inlet flange holes aligned with the holes in fuselage station 164 bulkhead, and align the upper hanger attaching holes.

d. Install gaskets, washers and screws at the two upper hanger attaching holes. Do not over torque the screws.

e. Place the JQ 912-9 gasket between the fuselage station 164 bulkhead and the inlet flange of the unit. On airplanes 5066 and subsequent and those with Service Bulletin 329-216 incorporated, place the screen assembly on the forward side of the station 164 bulkhead.

f. Install the eight inlet flange attaching screws, washers and gaskets. Do not overtorque the screws.

g. Connect the flexible tubes and the electrical connector. Refer to figure 21-11 for proper location of the flexible tubes.

h. Install the duct access panel. Assure that the upper edge of the panel is inserted under the lip of the upper duct skin, or excessive leakage from the nose section into the duct will result. Assure that the panel does not contact the end of the plenum sensing tube by placing a straightedge across the opening before installing the panel. Clearance should be .06 to .12 inch.

i. Close and secure the nose cone in accordance with Chapter 34.

j. Install the lower access panel in the fuselage between fuselage stations 178.75 and 199.00.

k. Perform an operational checkout of the fan and flow regulator assembly in accordance with paragraph 21.1.4.

21.2.4.3. REMOVAL OF FLOW LIMITER.

a. Open the nose cone assembly in accordance with Chapter 34.

b. Remove the eight screws from the support flange of the flow limiter assembly.

c. Move the unit forward and up until the flexible hose connection is accessible through the hole in fuselage station 164 bulkhead.

d. Disconnect the flexible tube from the bulkhead fitting, and remove the unit from the airplane. Retain screws and gasket for re-use if serviceable.

21.2.4.4 INSTALLATION OF FLOW LIMITER.

a. Open the nose cone assembly in accordance with Chapter 34 to gain access to the forward side of fuselage station 164 bulkhead.

b. Place a serviceable gasket on the mounting flange.

c. Insert the end of the flexible tube through the hole in the bulkhead, and attach to the fitting on aft side of bulkhead.

d. Position the unit to align holes, and install screws.

e. Close nose cone assembly in accordance with Chapter 34.

21.2.5 SERVICING.

21.2.5.1 PERIODIC MAINTENANCE. See Chapter 5 for periodic inspections required. Operational checkout when applicable should be accomplished in accordance with paragraph 21.1.4.

21.2.5.2 ADJUSTMENTS. Due to the small movement of the valve poppets and metering pins and the overall effect of these movements on the functioning of the system, adjustment is extremely critical. This type of adjustment cannot be accomplished with sufficient accuracy with the units installed on the airplane. If adjustment is found necessary, the unit must be removed from the airplane and replaced with a serviceable unit. Usually when the need for adjustment is indicated, the unit will require overhaul. See paragraph 21.2.5.3.

21.2.5.3 OVERHAUL. Overhauls, when required, should be accomplished only by qualified personnel using the required calibrated test equipment in accordance with the equipment manufacturers' instructions.

a. FAN AND FLOW REGULATOR ASSEMBLY. When operational checkout indicates that the regulator valve fails to move towards the closed position with increasing cabin pressure (to approximately 0.06 inch from housing at 8.8 PSIG), and/or if the pressure switch fails to energize and de-energize the fan at the correct differential pressure, when any other malfunction that cannot be corrected by use of the trouble shooting procedures of paragraph 21.2.3 occurs, or at the expiration of the normal replacement periods (see Chapter 5) the unit should be overhauled.

b. FLOW LIMITER. At the expiration of the normal replacement period (see Chapter 5) the flow limiter should be overhauled.

21.2.5.4 FLOW REGULATOR FILTER.

Note

The flow regulator filter is located in the pressurized section, and is normally mounted on the right-hand side of the nose wheel well at approximately fuselage station 168, water line 68. At the operator's option, the filter may be mounted in the center of the most forward access plate on the left-hand side of the wheel well, at approximately fusclage station 175, water line 68 (see Service Bulletin 329-216). In the optional location, access is gained by removing the access plate on which the filter is mounted.

Airplanes 5066 and subsequent, and those having Service Bulletin 329-216 incorporated, have a filter installed in the cabin pressure supply line routed to the flow regulator valve. The filter element is composed of cellulose, and cannot be cleaned and reused.

21.2.5.4.1 REPLACEMENT. The filter element should be replaced at intervals of 500 flying hours or more often as flying experience dictates. Two persons are required to perform this procedure. Replace the element as follows:

a. Open the nose cone in accordance with Chapter 34, and remove flow limiter in accordance with paragraph 21.2.4.3.

b. Man A: Hold the filter assembly with a thin wall ¾-inch open end wrench. Obtain access through flow limiter opening.

c. Man B: Remove line attached to end fitting of the filter assembly. This fitting is located in the pressurized area forward of nose wheel well on right side between fuselage station 164 and 171.

d. Man B: Remove jamnut on end fitting of filter assembly.

e. Man A: Remove filter assembly and wrench.

f. Release snap ring on attach fitting end of filter cover and remove cover.

g. Replace filter element using same part number as removed element.

h. Reinstall filter cover and snap ring.

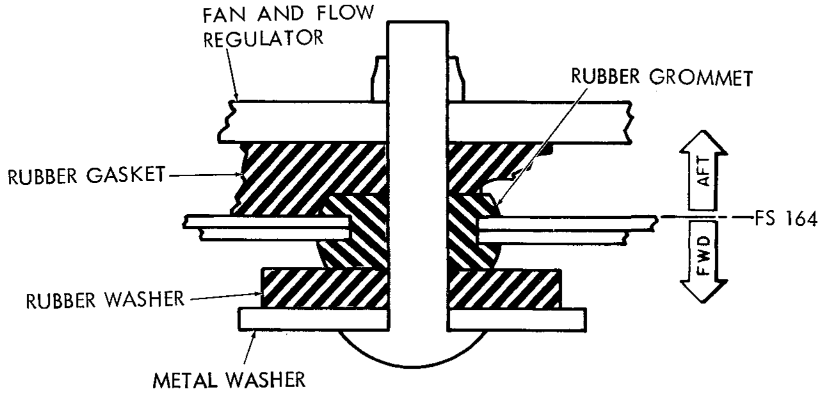
i. Man A: Reinstall filter assembly. Obtain access through flow limiter opening in bulkhead.

j. Man B: Install jamnut and tighten while Man A is holding filter assembly with ¾-inch open end wrench from inside.
Man B: Reinstall line and torque "B" nut.

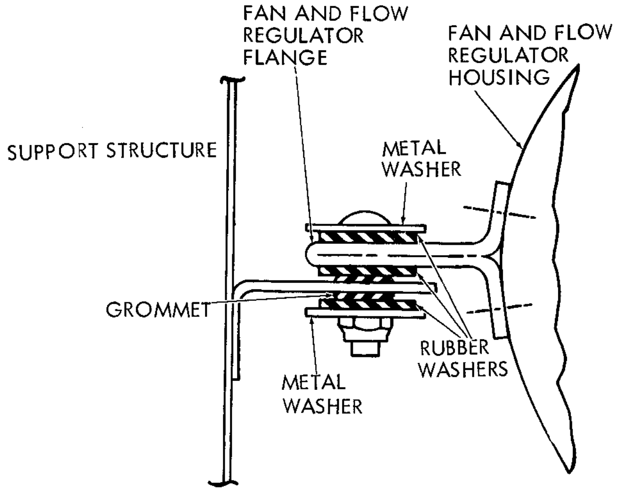
21.2.6 SYSTEM PRESERVATION.

21.2.6.1 NORMAL OPERATION. During normal operation no special precautions are required in order to obtain satisfactory service from the electronics cooling system.

21.2.6.2 AIRPLANE STORAGE. If the airplane is to be stored for periods in excess of 30 days, the fan and flow regulator outlet should be covered with waterproof paper or equivalent to prevent entry of dust, birds, etc.



SECT C C
TYPICAL ATTACHMENT AT FAN INLET (8 PLACES)



B
LOOKING AFT
TYPICAL ATTACHMENT UPPER HANGER BRKT
(2 PLACES)

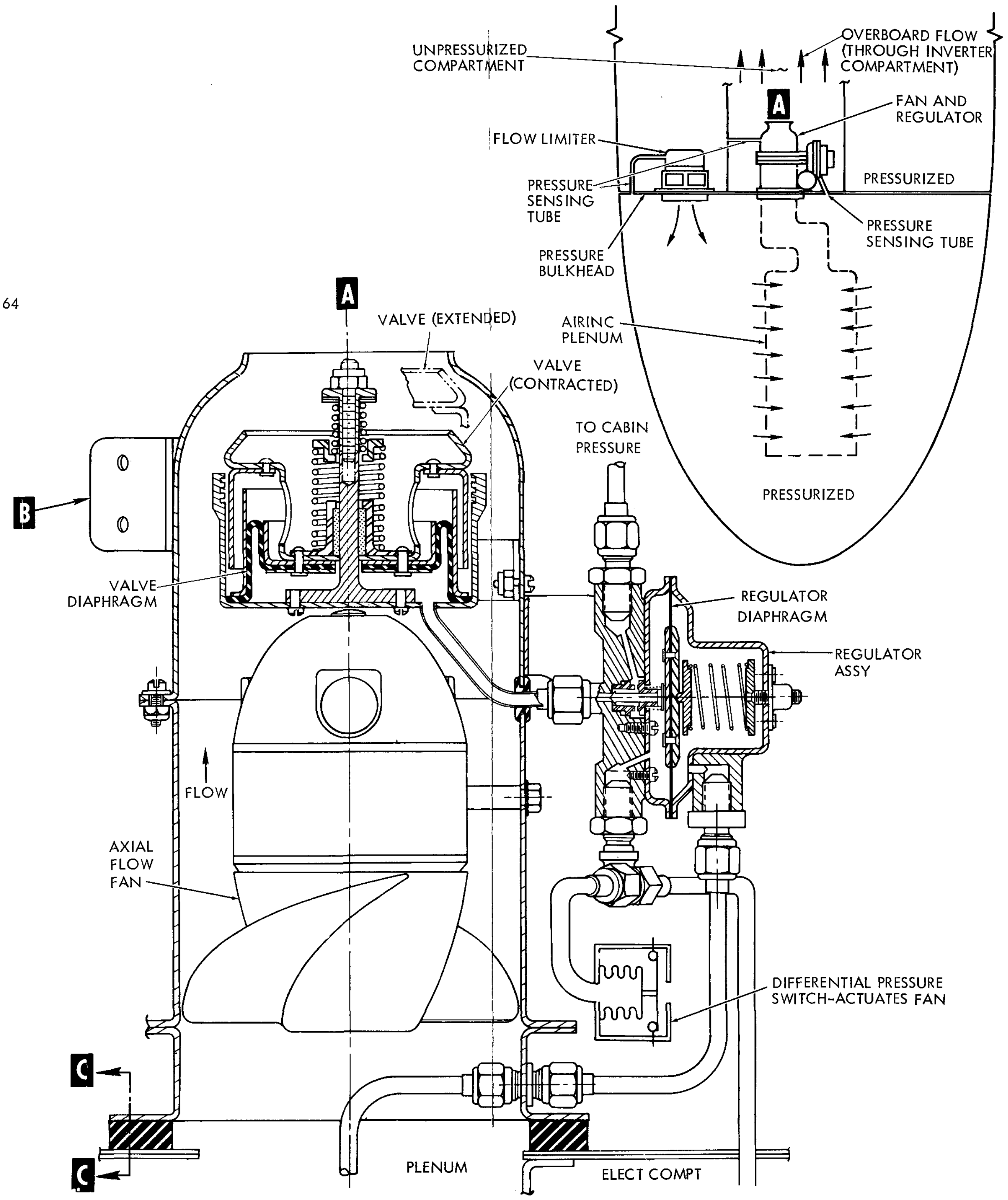
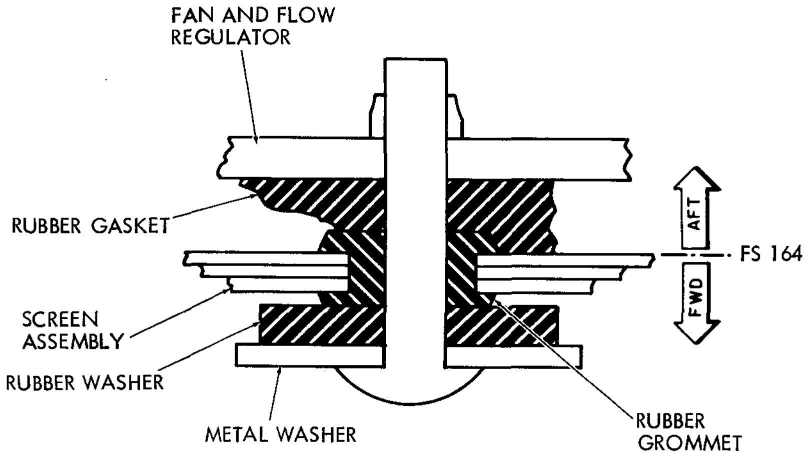
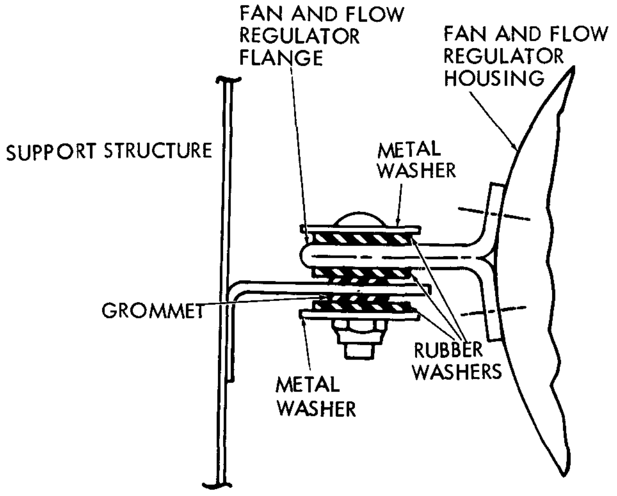


Figure 21-11. Electronic Cooling Schematic Diagram (Airplanes 5001 thru 5065 Not Having Service Bulletin 329-216 Incorporated)



SECT **D D**

TYPICAL ATTACHMENT AT FAN INLET (4 PLACES) (SEE SECT CC, FIGURE 2-11 FOR OTHER 4 PLACES)



B

LOOKING AFT
TYPICAL ATTACHMENT
UPPER HANGER BRKT
(2 PLACES)

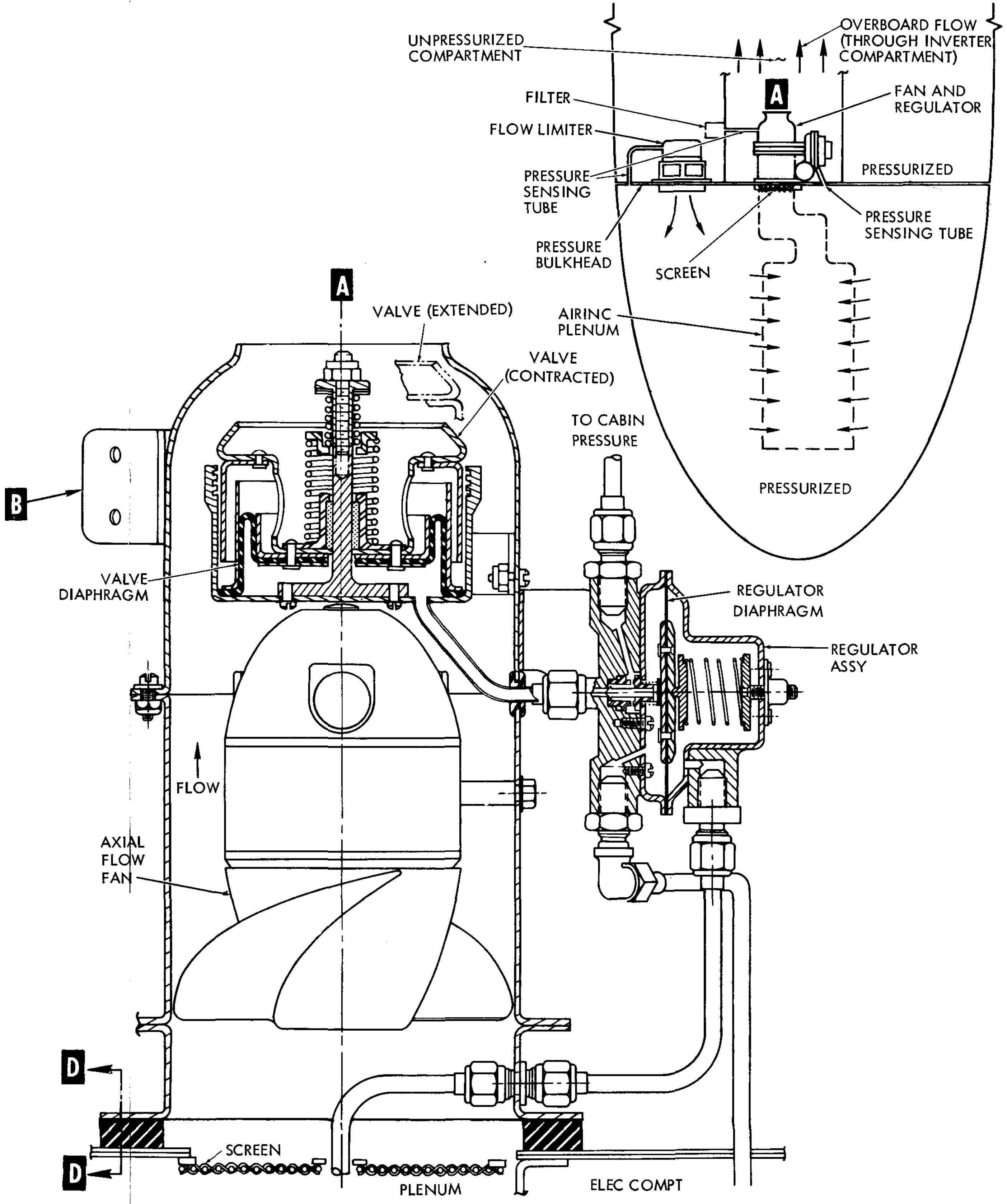
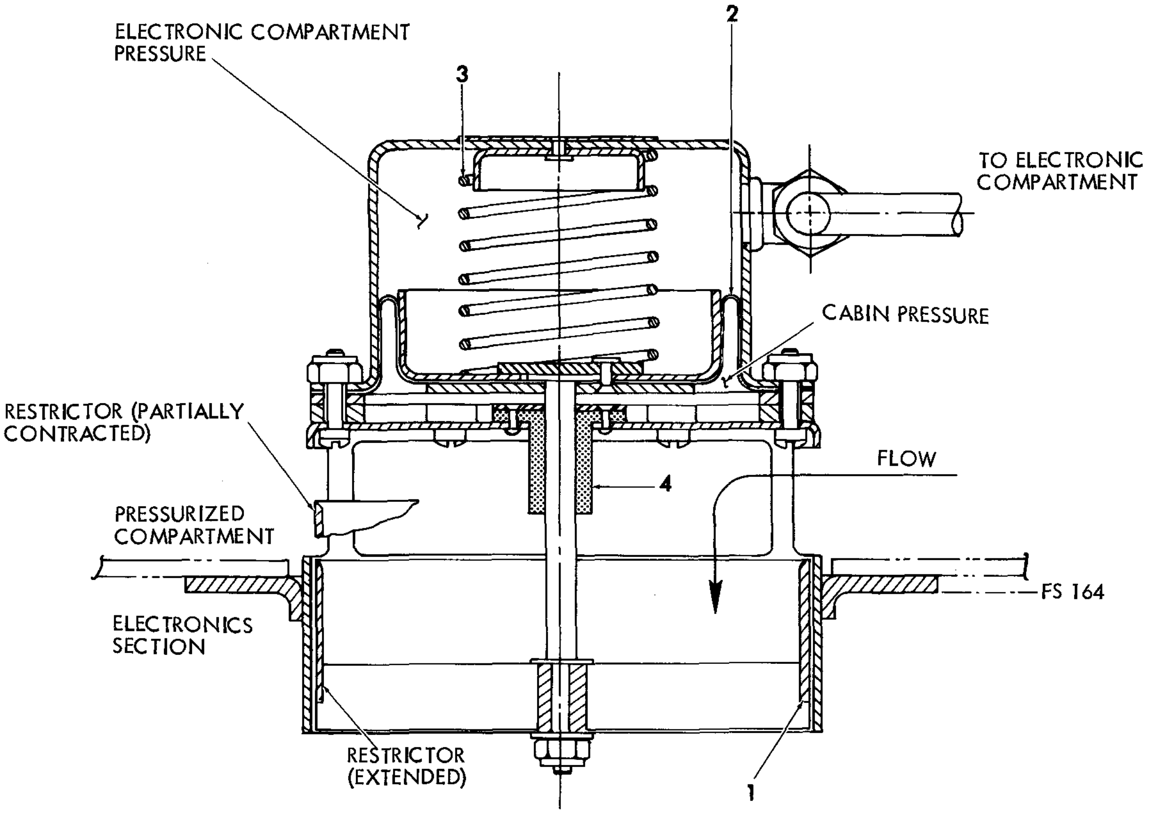


Figure 21-11A. Electronic Cooling Schematic Diagram (Airplanes 5066 and Subsequent, and Those Having Service Bulletin 329-216 Incorporated)



ELECTRONIC COOLING FLOW LIMITER
OPERATION: POPPET (1) IS SPRING - LOADED TO THE OPEN POSITION. THE POPPET WILL MOVE TOWARD THE CONTRACTED POSITION WHEN THE DIFFERENTIAL BETWEEN THE CABIN PRESSURE AND THE ELECTRONICS COMPARTMENT PRESSURE BECOMES GREAT ENOUGH TO PERMIT THE FORCE APPLIED BY THE CABIN PRESSURE TO OVERCOME THE RESULTANT FORCE APPLIED BY THE SPRING (3) AND THE ELECTRONICS COMPARTMENT PRESSURE. THE VALVE WILL NEVER COMPLETELY BLOCK ALL FLOW.

1. RESTRICTOR POPPET
2. DIAPHRAGM (POPPET)
3. SPRING
4. STATIONARY GUIDE

Figure 21-11B. Electronic Cooling Flow Limiter

SECTION 21.3
TOILET VENTILATION SYSTEM

Contents Page

DESCRIPTION AND OPERATION 21-59
- General 21-59
- Normal Operation 21-59

REMOVAL AND REPLACEMENT 21-59
- Removal of Toilet Venturi 21-59
- Installation of Toilet Venturi 21-59

SERVICING 21-60

SYSTEM PRESERVATION 21-60
- Normal Operation 21-60
- Aircraft Storage 21-60

21.3.1 DESCRIPTION AND OPERATION.

21.3.1.1 GENERAL. The toilet ventilation system provides for the immediate removal of odors from the toilet area before they can be circulated in the cabin. The system consists of a venturi installed with the inlet in the cabin and the outlet connected to the outside of the airplane (see figure 21-1). A tube is installed as a part of the toilet installation to connect the toilet bowl to the low pressure connection on the venturi.

21.3.1.2 NORMAL OPERATION. The toilet ventilation system is operational when the air conditioning system is on and air is flowing into the cabin. A small part of the cabin air flows overboard through the venturi, causing a lower pressure in the throat of the venturi. This lower pressure initiates flow from the toilet bowl to the venturi through the connecting tube. A check valve is installed between the downstream end of the venturi and the exit tube to prevent reverse flow when cabin pressure is lower than ambient pressure. The check valve will also prevent water from entering the cabin through the line in the event of ditching.

21.3.2 REMOVAL AND REPLACEMENT.

21.3.2.1 REMOVAL OF TOILET VENTURI. The venturi is installed under the floor at fuselage station 550, left buttock line 5.0 approximately.

a. Obtain access to the area by removal of interior furnishings as required.

b. Disconnect the vent tube from the wye forward of the venturi.

c. Loosen the forward venturi support clamp and slide it forward off the flange of the support bracket.

d. Remove the clamp at the aft end of the venturi, and slide the coupling sleeve aft on the exit tube.

e. Remove the venturi and check valve.

f. Retain clamps, coupling sleeves and check valve for re-use, if serviceable.

21.3.2.2 INSTALLATION OF TOILET VENTURI.

a. Obtain access in accordance with paragraph 21.3.2.1.

b. Position the check valve flange against the end of the exit tube with the locating pins engaged with notches in the tube. Ensure that the flappers are free to swing open.

c. Position the venturi with the wye for the vent tube connection at the forward end, and align the aft end with the exit tube. Hold the check valve in position with force in the aft direction on the venturi body, and slide the coupling into place; install the clamp.

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21.3.2.2 (Cont).

d. Slide the forward support clamp aft over the support bracket flange, and tighten.

e. Connect the toilet vent tube to the wye at the forward end.

21.3.3 SERVICING. No servicing is required for the toilet ventilation system except for inspection for obstructions and for security of mountings, etc. See Chapter 5 for required inspections and time intervals between inspections.

21.3.4 SYSTEM PRESERVATION.

21.3.4.1 NORMAL OPERATION.

None required.

21.3.4.2 AIRCRAFT STORAGE. If the airplane is to be stored for periods in excess of 30 days, the venturi outlet in the bottom fuselage skin should be covered with waterproof paper to prevent entry of dust, insects, etc.

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CHAPTER 22
AUTOPILOT

Section	Page
22.0 AUTOPILOT SP40, SPERRY	22-2A
22.1 AUTOPILOT AP104, COLLINS	22-19

SECTION 22.0
AUTOPILOT
SP40 (SPERRY)

TABLE OF CONTENTS

Contents	Page
DESCRIPTION AND OPERATION	22-1
General	22-1
List of Components	22-2
Autopilot Flight Controller	22-3
Airspeed Differential Pressure Switch	22-3
Altitude Controller	22-3
Accelerometers	22-3
Autopilot Flight Control Computer	22-3
Servomotors	22-4
Engage Coupler	22-4
OPERATIONAL CHECKOUT	22-4
Test Equipment Required	22-4
Autopilot System	22-4
Engage Coupler (Special Test)	22-7
Servo Circuitry Check	22-8
Flight Control Computer	22-8A
TROUBLESHOOTING	22-8A
REMOVAL AND INSTALLATION	22-16
Removal of Accelerometers	22-16
Installation of Accelerometers	22-16
Removal of Autopilot Servos	22-16
Installation of Autopilot Servos	22-16
FIGURES	
22-1 Autopilot System Components	22-17
22-2 Autopilot System Block Diagram	22-18

22.0.1 DESCRIPTION AND OPERATION.

22.0.1.1 GENERAL. Any of several modes of flight or compatible combinations can be commanded by the pilot and maintained by a Sperry SP-40 autopilot system. The airplane can be made to fly smoothly a selected magnetic heading, a selected VOR station radial, an ILS approach, a constant altitude, a constant rate of climb or descent, a commanded turn, or a Doppler radar track if the Doppler system is installed. A magnetic heading is selected by the heading select knob on the copilot's course indicator (aircraft 5001 thru 5045 or on the pilot's course indicator on aircraft 5046 and up). Depressing the heading select button on the autopilot flight controller commands the autopilot to fly the selected heading. A VOR radial is selected by the course knob and the intercept angle by the heading select knob on the copilot's course indicator (aircraft 5001 thru 5045) or on the pilot's course indicator (aircraft 5046 and up). Using the turn knob to maneuver the aircraft to the correct angle of intercept, 60°, and depressing the radio

22.0.1.1 (Cont).

button on the flight controller will command the autopilot to intercept and bracket the selected radial and roll out to wings level on the selected course. An ILS approach is established by the heading select knob on the copilot's course indicator (aircraft 5001 thru 5045) or on the pilot's course indicator (aircraft 5046 and up). By establishing an intercept angle of 45 to 90 degrees with the beam and depressing the radio button on the flight controller, a localizer bracketing maneuver will be initiated and the aircraft will execute a coordinated turn toward the localizer transmitter, and roll out on course, ready for glidepath interception. A constant altitude is maintained by depressing the ALT button on the flight controller. On airplanes equipped with Doppler the Doppler mode is established by depressing the DOP button. Operating the pitch switch or capture of a glidepath automatically disengages the altitude hold function. Operating the pitch switch on the flight controller will result in a constant rate of climb or descent. Maximum pitch angle is 20 degrees up or down. A commanded coordinated turn in an unlimited angle to the right or left can be achieved by the use of the turn knob on the flight controller. Operation of the turn knob cancels any selected heading signal previously existing through the heading select mode of the autopilot flight controller. The major components in the system include a flight controller, an airspeed differential pressure switch, an altitude control, two accelerometers, a flight control computer, three servomotors, and an engage coupler.

22.0.1.1 (Cont).

Power is supplied to the autopilot system through an AUTOPILOT AC and an AUTOPILOT DC circuit breaker on the copilot's wainscot circuit breaker panel. For the AUTOPILOT system to operate power must be supplied to the No. 2 vertical gyro through a VERT GYRO No. 2 AC and a VERT GYRO No. 2 DC circuit breaker, and to the No. 2 DIRECTIONAL GYRO through GYRO COMPASS No. 2 circuit breaker on the copilot's wainscot circuit breaker panel. Vertical gyro No. 2 furnishes pitch and roll signals to the autopilot flight control computer as well as to the No. 2 flight director system. Directional gyro No. 2 furnishes heading signal information to the autopilot. When all the engage interlock functions are compatible, the autopilot may be engaged by depressing the AFCS engage switch on the flight controller. The engage interlock circuitry is composed of relays in the flight control computer, in the flight controller, and in the engage coupler, micro switches in the servomotors which are closed when the three servomotor clutches pull in, and relays and switches in the vertical gyroscope liquid level switches in the Vertical Gyro prevent engagement of the autopilot if, the turn is uncoordinated and the bank angle is greater than six degrees. Disengagement of the autopilot may be accomplished by depressing the autopilot disconnect button on the pilot's or copilot's control wheel, or by lifting the toggle of the AFCS engage switch located on the flight controller in the center pedestal. The autopilot disconnect buttons are the normally closed type.

22.0.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Flight Control Computer	1	Sperry	JR1000-57	1781218-21 △1
Flight Control Computer	1	Sperry	JR1000-59	1781218-24 △2
Flight Control Computer	1	Sperry	JR1000-81	1781218-242△2△3
Flight Control Computer	1	Sperry	JR1000-89	1781218-242△2△4
Flight Controller (red lighted) or	1	Sperry	JR1000-53 or	1781216-13 △1
Flight Controller (white lighted) or	1	Sperry	JR1000-69 or	1781216-131△1
Flight Controller (white lighted) or	1	Sperry	JR1000-77 or	1781216-131△1
Flight Controller (white lighted) or	1	Sperry	JR1000-85 or	1781216-131△1
Flight Controller (red lighted) or	1	Sperry	JR1000-55 or	1781216-23 △2
Flight Controller (white lighted) or	1	Sperry	JR1000-71 or	1781216-231△2
Flight Controller (white lighted) or	1	Sperry	JR1000-79 or	1781216-231△2

22.0.1.2 (Cont).

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Flight Controller (white lighted)	1	Sperry	JR1000-87	1781216-231 [2]
Elevator Servo Drive	1	Sperry	JR1000-63	615743-22
Rudder Servo Drive	1	Sperry	JR1000-63	615743-22
Aileron Servo Drive	1	Sperry	JR1000-61	615743-21
Fore Yaw Accelerometer	1	Sperry	JR1000-3	615794-1
Aft Yaw Accelerometer	1	Sperry	JR1000-3	615794-1
Differential Pressure Switch	1	Sperry	JR1000-25	2553919-1
Altitude Control	1	Sperry	JR1000-11	620399-1
Autopilot Interlock Relay	1	Hartman		C12A or CR16AF
Autopilot Transfer Relay	1			MS25268-D1
Engage Coupler	1	Sperry	JR1000-67	2587349
Vertical Gyro	1	Sperry	JR1000-35 or JR1000-65	1782356-111 [5] or 2586137-1 [5]
Vertical Gyro (VG203M)	1	Sperry	JR1000-75	2586137-6
Phase Adapter	1	Sperry	JR1000-9	661102B [5]
Gain Calibrator	1	Sperry	JR1000-83	1782483-710 [6]

[1] Installed on airplanes without Doppler.

[2] Installed on airplanes with Doppler.

[3] Category II Autopilot Computer installed airplanes 5122 thru 5130 (production).

[4] Latest modified version of [3] installed airplanes 5131 and up (unit includes variable trim coupler and radio beam coupler modifications).

[5] Not used on airplanes 5115 and up.

[6] This unit used with [3] or [4].

22.0.1.3 AUTOPILOT FLIGHT CONTROLLER. (See figure 22-1.) The pilots command the autopilot from an autopilot flight controller. In addition to three trim meters, this panel has one switch, two controls, and three pushbutton selector switches. A two-position ("ENG," "OFF") AFCS switch engages the autopilot. As an alternate to the use of one of the two control wheel AP DISC pushbuttons, the AFCS switch also can be used to disengage the autopilot. A warm-up of 2 to 3 minutes is required before a time delay relay enables the autopilot to be operative. A smooth turn can be commanded using a TURN control. Ordinarily, in the detent position, the TURN control can be rotated in the desired direction enough to effect up to a 40 degree ±2° bank. The bank angle is now symmetrical in all cases. To obtain a symmetrical bank angle the MAY BANK limit has been reduced to 32 degrees ±2° on later versions of the flight controller. The pitch control is a three position self-centering switch, (OFF, UP, and DN). The control is moved from the center-off position to command a pitch attitude change. The pitch attitude of the aircraft changes, at a rate proportional to the period of time the control is held out of the center-off position. A positive movement of the control from the center-off position is essential to command a pitch attitude change, due to an "easy on" feature in the pitch axis circuit. Once the PITCH control is returned to the center-off position, the amount of pitch commanded is maintained until canceled by a conflicting command. Three mode selector pushbutton switches, one for ALT HOLD, one for HDG SEL, and one for RADIO mode, light when closed. A fourth switch, for Doppler, is installed on airplanes equipped with the Doppler radar nav system. If a mode has been selected and its pushbutton switch light is on, that mode is maintained until a new command is given. Then the new command becomes effective, the mode previously selected is discontinued, and its switch light goes off. There is a priority of RADIO over HDG SEL within beam limits. Contradictory commands are never effective. A RUD, an AIL, and an EL trim meter on the flight controller indicate the amount of DC voltage from the respective servoamplifiers being directed to the servomotors. If any aircraft out-of-trim condition exists (pitch axis only) after the autopilot is engaged, the trim coupler will automatically retrim the stabilizer. Before engaging the autopilot, the trim meters serve as alignment meters. Unless all three meters are fairly well centered, the autopilot should not be engaged. Aircraft 5066 and up have white-lighted flight controllers installed.

CAUTION

All inflight operation of the autopilot must be in accordance with limitations and procedures in the pilot's handbook.

22.0.1.4 AIRSPEED DIFFERENTIAL PRESSURE SWITCH. Pitot and static differential pressure serves to actuate a three-position airspeed differential pressure switch. At speeds up to 200 knots, the switch remains in the initial position, from 200 to 300 knots in a second, and at 300 knots and above in a third. The position of the airspeed differential pressure switch determines which of three sets of rate parameters is effective in the gain calibrator of the autopilot flight control computer. Optimum rate gains are thus employed over the range of operating speeds.

CAUTION

When bench testing the ALT controller, do not change altitude with DC power applied, as the unit may become damaged.

22.0.1.5 ALTITUDE CONTROL. When the ALT mode selector pushbutton is depressed, the barometric pressure, and hence the altitude, is established as a reference for the autopilot to maintain. This is done with an altitude control consisting of an aneroid barometer, an E-pickoff variable transformer, and a mechanical coupler. The aneroid barometer is connected to a static port and is always free to respond to the static pressure. When operating in the "ALT" mode, the mechanical coupler connects the barometer to the movable part of the variable transformer. As pressure variations are sensed by the barometer, they are reflected in the output of the variable transformer. This output is then directed to the autopilot flight control computer.

Note

If a hardover is encountered, in the ALT. HOLD mode, causing the aircraft to change altitude by more than 500 feet the altitude controller should be bench tested, per the manufacturer's recommendations.

22.0.1.6 ACCELEROMETERS. Two identical accelerometers provide a rate of yaw signal to the flight control computer. Each consists of an E-pickoff variable transformer in which the movable part of the transformer is suspended between two springs to permit response to lateral acceleration. The system is fluid-damped. One accelerometer is on the normal or vertical axis through the center of gravity of the airplane, and the other is in the nose. The accelerometers are differentially connected so sideslip acceleration signals are canceling, but yaw acceleration signals are additive.

22.0.1.7 AUTOPILOT FLIGHT CONTROL COMPUTER. All of the computing and amplification for the autopilot system is done by a flight control computer. This computer is composed of seven modules and a gain calibrator which provide the proper rate parameters according to the airspeed range detected by the airspeed differential pressure switch. A yaw computer module and a signal computer module receive information, compute it for commanded results, and supply necessary signals to the appropriate servoamplifiers. Each of three identical servoamplifier modules, one corresponding to each of the servomotors, routes signals through a two-stage, transistorized preamplifier and then through a magnetic amplifier. The DC output signals obtained are then directed to the appropriate servomotors. When

22.0.1.7 (Cont).

the "RADIO" mode selector pushbutton is depressed, a radio beam coupler module receives the available signals from the No. 1 VOR-localizer, the No. 1 glideslope receiver, and the No. 1 flight director systems. Nose-up or nose-down pitch trim signals from the autopilot are routed from a trim coupler module to the autopilot transfer relay.

22.0.1.8 SERVOMOTORS. Signals from their respective servoamplifiers operate three servomotors. Each of these units is a series-wound, split-field, reversible DC motor connected to the control rigging through a solenoid-operated clutch. A rate generator provides rate feedback from the servomotor to the servoamplifier. The rudder servomotor and the elevator servomotor are mechanically and electrically interchangeable. The aileron servomotor differs mechanically and electrically from the rudder and elevator servomotors. The torque output of the aileron and elevator servomotors is limited by the use of close tolerance resistors in series with the respective servo-amplifier input excitation. The rudder servomotor is not limited in torque output. The force required to overpower the servomotors under all conditions of operation does not exceed the following values, excluding mechanical input friction:

a. 53 pounds force on one rudder pedal.

b. 32 pounds up and 39 pounds down force applied normal to the column at the centerline of the control wheel.

c. 157 inch-pounds force applied tangentially to a control wheel.

22.0.1.9 ENGAGE COUPLER. The primary function of the engage coupler is to ensure disengagement of the autopilot. The engage circuitry of the flight control computer contains miniature relays. If a pole in a relay should weld, the engage coupler ensures a safe condition by grounding the engage relays and servomotor clutch circuits. The engage coupler contains a pair of relays and a transistorized time delay circuit. The relays are capable of carrying high current loads. The transistorized time-delay ensures isolation of the power supply before the relay poles are grounded.

22.02 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Collins 479-T2	Signal Generator	Supplies directional signal to receiver.

22.0.2.1 AUTOPILOT SYSTEM.

a. Remove the servoamplifiers from the autopilot flight control computer. Remove control surface locks.

b. Connect external power to the airplane, (refer to Chapter 24) and energize the AC buses.

c. Make certain the following circuit breakers are closed: VERT GYRO NO. 2 AC, VERT GYRO NO. 2 DC, AUTOPILOT AC, AUTOPILOT DC, COMPASS NO. 2 115V AC, and COMPASS NO. 2 26V AC.
RESULT: After being energized for 3 minutes, the No. 2 vertical gyro should be erected. This is verified by the disappearance of the GYRO warning flag on the copilot's approach horizon.

d. With the autopilot disengaged and hydraulic boost pressure on, measure the force required to move the aileron, elevator, and rudder controls at either the pilot's or the copilot's position, using spring scales. Record the values indicated.

e. Place the SLAVE COMPASS-DG switch in the "SLAVE COMPASS" position.

f. With the aileron, elevator and rudder controls in neutral, and the turn knob in the detent position, place the AFCS switch in the "ENG" position. Then press the pilot's AP DISC pushbutton.
RESULT: The autopilot should engage and remain so until the AP DISC is pressed. Then the autopilot should disengage as evidenced by the AFCS switch dropping out of the "ENG" position.

g. Repeat step (f) using the copilot's AP DISC pushbutton.

h. Place the AFCS switch in the "ENG" position; then place the HOR STAB TRIM SYS switch in the "EMERGENCY" position.
RESULT: The autopilot should engage and remain so until the HOR STAB TRIM SYS switch is placed in the "EMERGENCY" position. Then the autopilot should disengage as evidenced by the AFCS switch dropping out of the "ENG" position.

i. Place the AFCS switch in the "ENG" position; then pull out the VERT GYRO NO. 2 AC circuit breaker.
RESULT: The autopilot should engage and remain so until the circuit breaker is pulled out. Then the autopilot should disengage as evidenced by the AFCS switch dropping out of the "ENG" position.

j. Repeat step (i) for the VERT GYRO NO. 2 DC circuit breaker.

k. With the AFCS switch in the "OFF" position, depress the RADIO, HDG-SEL, and ALT mode selector switches.
RESULT: None of the switches should engage.

22.0.2.1 (Cont).

This is indicated if the lights in the switches do not go on.

l. Rotate the TURN knob out of detent position, and place the AFCS switch in the "ENG" position.
RESULT: The AFCS switch should not remain in the "ENG" position, thereby indicating that the autopilot will not engage.

m. Set the TURN knob in the detent position, place the AFCS switch in the "ENG" position, and rotate the TURN knob out of the detent position. Then depress the RADIO and the HDG SEL mode selector switches.
RESULT: The autopilot should engage and remain so, but neither of the mode selector switches should engage. This is indicated if the lights in the switches do not go on.

n. Set the TURN knob in the detent position, and depress the RADIO mode selector switch. Then rotate the TURN knob out of the detent position.
RESULT: When depressed, the RADIO mode selector switch should engage and its light should go on. When the TURN knob is rotated out of detent, the RADIO mode selector switch should disengage and its light should go off.

o. Repeat step (n) for the HDG SEL mode selector switch.

p. Set the TURN knob in the detent position, and depress the ALT mode selector switch. Then place the PITCH control in the "UP" position.
RESULT: When depressed, the ALT mode selector switch should engage and its light should go on. When the PITCH control is operated, the ALT mode selector switch should disengage and its light should go off.

q. Repeat step (p), placing the PITCH control in the "DN" position.

r. Tune the No. 2 VOR-localizer (airplanes 5001 thru 5045) and the No. 1 VOR-localizer (airplanes 5046 and up) to a vacant VOR frequency, and depress the RADIO mode selector switch. Place the pitch control in the "UP" and the "DN" positions. Then rotate the TURN knob out of detent.
RESULT: The RADIO mode selector switch should engage and its light should go on. The switch should remain engaged when the PITCH control is operated, but when the TURN knob is rotated out of detent, the switch should disengage and its light should go off.

s. Set the TURN knob in detent, depress the ALT mode selector switch, and select a vacant localizer frequency on the NO. 2 VOR-localizer (airplanes 5001 thru 5045) and on the No. 1 VOR-localizer (airplanes 5046 and up). Then depress the RADIO mode selector switch.
RESULT: The ALT mode selector switch should disengage and its light should go off when the RADIO mode selector switch is depressed. The RADIO mode selector switch should then engage and its light should go on.

t. Place the PITCH control in the "UP" position.
RESULT: The RADIO mode selector switch should disengage and its light should go off.

u. Pull out the AUTOPILOT AC and the AUTOPILOT DC circuit breakers.

v. Replace the servoamplifiers in the autopilot flight control computer, and push in the AUTOPILOT AC and the AUTOPILOT DC circuit breakers.

w. Place the AFCS switch in the "ENG" position, and attempt to overpower the aileron, elevator, and rudder controls.
RESULT: Noticeable force should be required to overpower the controls, indicating that the autopilot system is operating.

CAUTION

Do not allow the servomotors to maintain full torque output for periods longer than 30 seconds. If any servo should deliver a hardover command, immediately disengage the system. In subsequent activity to determine the cause of such a malfunction, the servomotor should be left in a hard-over condition for a minimum length of time. Also, if a servomotor is removed from the system, power should not be applied to the system until the corresponding servoamplifier has been removed from the computer.

x. With the autopilot engaged and hydraulic boost pressure on, measure the force required to move the aileron, elevator, and rudder controls at either the pilot's or copilot's position, using spring scales. Record the indicated results, and subtract the related values obtained in step (d).
RESULT: The following differences should be obtained from the subtractions:

	MAX	MIN
aileron	157 in-lbs	115 in-lbs
elevation		
(nose-up)	32 lbs	27 lbs
(nose-down)	39 lbs	33 lbs
rudder	53 lbs	47 lbs

Note

In the aileron axis, measure force with spring scale attached to wheel. Allowable inch-pounds may then be calculated by multiplying force in pounds times distance in inches from spring attach point to center of wheel.

22.0.2.1 (Cont).

y. Place the AFCS switch in the "OFF" position.

z. Remove the mounting bolts for the No. 2 vertical and the No. 2 directional gyros.

aa. Place the AFCS switch in the "ENG" position.

ab. Tilt the No. 2 vertical gyro to simulate an airplane roll to the right.
RESULT: The aileron control wheel should rotate counterclockwise.

ac. Tilt the No. 2 vertical gyro to simulate a nose-up airplane attitude.
RESULT: The control column should move forward.

ad. Rotate the No. 2 directional gyro to simulate an airplane turn to the right.
RESULT: The aileron control wheel should rotate counterclockwise, and the left rudder pedal should move forward initially and then return approximately to the neutral position.

ae. Place the AFCS switch in the "OFF" position.

af. Remount the No. 2 vertical and the No. 2 directional gyros.

ag. Place the AFCS switch in the "ENG" position.

ah. Overpower the elevator by pushing the column forward.
RESULT: The elevator trim meter should give a downward deflection indication, and the horizontal stabilizer should move in a direction to cause a pitch up.

ai. Repeat step (ah), pulling the control column aft.

aj. Rotate the TURN knob clockwise.
RESULT: The control wheel should rotate clockwise.

ak. Return the TURN knob to detent, and rotate it counterclockwise.
RESULT: The control wheel should rotate counterclockwise.

al. Return the TURN knob to detent.

am. Select a heading, on the copilot's course indicator (airplanes 5001 thru 5045) and on the pilot's course indicator (airplanes 5046 and up), at least 10 degrees greater than the airplane heading. Depress the HDG SEL mode selector switch.
RESULT: The HDG SEL switch should engage and its light should go on; the control wheel should rotate clockwise.

an. Repeat step (am) for a selected heading at least 10 degrees less than the airplane heading.

ao. Place the PITCH control in the "UP" position, then in the normal position, and then place the PITCH control in the "DN" position.
RESULT: The control column should move aft, then stop and remain stationary, and then move forward.

ap. Return the control column to the center position, using the PITCH control.

aq. Align the course arrow on the copilot's course indicator (airplanes 5001 thru 5045) and on the pilot's course indicator (airplanes 5046 and up) with the airplane heading. Select a vacant VOR frequency channel on the No. 2 VOR-localizer (airplanes 5001 thru 5045) and on the No. 1 VOR-localizer (airplanes 5046 and up). Depress the RADIO mode selector switch.
RESULT: The RADIO mode selector switch should engage and its light should go on.

ar. Place the HDG-ILS knob on the copilot's (airplanes 5001 thru 5045) or pilot's (airplanes 5046 and up) approach horizon in the "HDG" position, and select a heading greater than the airplane heading, using the COURSE knob on the course indicator.
RESULT: The control wheel should rotate clockwise.

as. Return the course arrow on the course indicator to the airplane heading, and place the AFCS switch in the "OFF" position.

at. Set up a Collins 479T-2 signal generator approximately 75 feet from the VOR-localizer antenna.

au. Originate a lateral signal which will cause a displacement to the right of the course bar on the copilot's (airplanes 5001 thru 5045) or pilot's (airplanes 5046 and up) course indicator. Place the AFCS switch in the "ENG" position, and depress the RADIO mode selector switch.
RESULT: The control wheel should rotate clockwise.

av. Repeat step (au) with a left displacement of the course bar.

aw. Return the course bar to the center position.

ax. Place the HDG-SEL knob on the copilot's approach horizon (airplanes 5001 thru 5045) and on the pilot's approach horizon (airplanes 5046 and up) in the "ILS" position, and select a localizer frequency on the No. 2 VOR-localizer (airplanes 5001 thru 5045) and on the No. 1 VOR-localizer (airplanes 5046 and up). Originate a signal with the signal generator to cause an upward deflection of the glideslope pointer on

22.0.2.1 (Cont).

the copilot's approach horizon (airplanes 5001 through 5045) and on the pilot's approach horizon (airplanes 5046 and up).
RESULT: The control column should move aft.

ay. Repeat step (ax) with a downward deflection of the glideslope pointer on the approach horizon.

az. Place the AFCS switch in the "OFF" position.

ba. Perform an operational checkout of autopilot system pertaining to third pitot-static section per 34.10.2.6.

bb. Place a test magnet of 5 gauss or less to the left side against the forward accelerometer housing at a point 1 inch up from the mounting surface, six-tenths inch aft of the nameplate end, and parallel to the sensitive axis.
RESULT: The left rudder pedals should move forward, and then return slowly to neutral.

bc. Repeat step (bb), placing the test magnet to the right side of the accelerometer housing.

bd. Gain access to the aft accelerometer through the cabin floor, and place the test magnet to the left side of the housing.
RESULT: The right rudder pedals should move forward, and then return slowly to neutral.

be. Repeat step (bd), placing the test magnet to the right side of the aft accelerometer housing.

bf. Automatic trim. Aircraft 5001 thru 5121 are equipped with a Trim Coupler module, part no. 1775390-6, Mod. B, or earlier. This unit has a fixed output and provides a horizontal stabilizer trim rate of 8 degrees per minute. Aircraft 5122 and up are equipped with a variable Trim Coupler module part no. 1775390-6, Mod. D.

bg. The variable trim coupler is adjustable externally to provide the desired horizontal stabilizer trim rate of 6 min. The external potentiometer provides a pulse period which can be varied from 200 to 1,000M sec. An internal trim potentiometer provides for a pulse width adjustment of 55 to 70M sec. The pulse width is factory set at 60 to 65M sec.

When it is considered necessary to adjust the horizontal stabilizer autopilot trim rate the following action is required.

1. Remove equipment to the left of the autopilot computer. To obtain access to the pot located inside the trim coupler, remove the small piece of tape covering the 1/4 inch hole in the trim coupler case.

2. Engage the autopilot.

3. Push forward on the control column simultaneously starting the aircraft clock and causing the horizontal stabilizer to drive nose up.

4. Stop the clock at 6 degrees nose up trim if clock indicates between 54 and 66 seconds trim rate is acceptable.

5. If trim rate is considered too fast or too slow, adjust external pot, CW or CCW minimum amount.

Note

Some operators of early airplanes had the Variable Trim Coupler installed on a retrofit basis and prefer a 4 degree per minute trim rate. This rate apparently reduces the tendency to porpoise on "altitude hold." However the Lockheed recommended setting on new airplanes, which are comparatively tighter, is 6 degrees per minutes ±10%.

bh. Disconnect external power and restore the airplane to normal.

22.0.2.2 ENGAGE COUPLER (SPECIAL TEST). The engage coupler must be periodically checked to confirm that certain components have not failed undetected. This test must be performed at every 100-hour inspection.

CAUTION

This test must be accomplished in sequence as instructed, or malfunction of the equipment will occur.

a. Pin A must be grounded for the following tests. All voltages should be applied through a 0.5-amp fuse or circuit breaker:

	Procedure	Location
(1)	Apply 28 V DC.	Pin I
(2)	Measure zero volts.	Pins B, C, E, F, and G
(3)	Remove 28 V DC.	Pin I
(4)	Apply 28 V DC.	Pins C and G

22.0.2.2 (Cont).

(5)	Remove 28 V DC.	Pin C
(6)	The engage coupler should produce two distinct clicks.	
(7)	Apply 28 V DC.	Pin C and G
(8)	Apply 28 V DC.	Pin I
(9)	Remove 28 V DC.	Pin C
(10)	Measure 28 V DC.	Pins B, C, E, and F
(11)	Remove 28 V DC.	Pin I
(12)	Measure zero volts.	Pins B, C, E, and F
(13)	Apply 28 V DC.	Pin C
(14)	Measure 28 V DC.	Pin B
(15)	Remove voltage.	
(16)	Read continuity between Pins A, E, and F.	
(17)	Connect a 200-ohm, 2-watt resistor in series with the positive lead of a Simpson 260D V.O.M. or equal.	
(18)	Connect the free end of the resistor to Pin B.	
(19)	Connect the negative lead of the V.O.M. to Pin A.	
(20)	Set the V.O.M. 100 MA. Scale.	
(21)	Apply 28 V DC to Pins C and G. The meter should read between 30 and 50 MA.	
(22)	Remove 28 V DC from Pin C.	
(23)	The current should drop to zero.	
(24)	Remove all voltage and all external grounds. Restore system to normal.	

b. **Failure to meet requirements of any part of the above test is reason for rejection of the part.**

22.0.2.3. SERVO CIRCUITRY CHECK. If a servo is suspected of being defective, some checks may be made from the hose rack connector for the autopilot computer before removing the servo. See wiring diagram JR818.
Tach feedback and excitation circuitry can be checked as follows: (All resistance readings ± 20 percent.)

a. With a Simpson 260 ohmmeter or equivalent, ground one probe and apply the other probe to pin 13 of the flight control computer's aircraft connector, read approximately 22.0 ohms (rudder servo). Conduct the same check on pin 14 (aileron servo) and pin 15 (elevator servo) and read approximately 22.0 ohms. This checks the excitation circuits.

b. Apply the probes to pin 48 and pin 49 and read approximately 400 ohms. Apply probes to pins 49 and 50 and read approximately 9.0 ohms (aileron servo). Apply probes to pins 38 and 39 and read approximately 400 ohms. Apply probes to pins 39 and 40 and read approximately 9.0 ohms (rudder servo). Apply probes to pins 82 and 83 and read approximately 400 ohms. Apply probes to pins 83 and 84 and read approximately 9.0 ohms (elevator servo). This checks the Tach feedback circuits.

22.0.2.4 FLIGHT CONTROL COMPUTER. Test the 1781218-21 (JR1000-57) or 1781218-24 (JR1000-59) flight control computer as follows to ensure that the engage relays have not failed:

Note

In the tests described below, terminal J1800-6 shall be used as the negative side of the applied 28 V DC.

a. Apply 28 V DC to J1800-24.

b. There shall be continuity between J1800-16 and pins J1800-18, -19, and -20.

c. Connect the leads of an ohmmeter to the J1800 terminals indicated below. Note the resistance readings obtained in each step.

	Ohmmeter Lead	Terminal
(1)	Positive	J1800-18
	Negative	J1800-6
(2)	Positive	J1800-6
	Negative	J1800-18
(3)	Positive	J1800-19
	Negative	J1800-6
(4)	Positive	J1800-6
	Negative	J1800-19
(5)	Positive	J1800-20
	Negative	J1800-6
(6)	Positive	J1800-6
	Negative	J1800-20

d. The resistances obtained in c(1), c(3), and c(5) shall differ from the resistances obtained in c(2), c(4), and c(6) by a factor of approximately 10.

e. Remove all voltage from the flight control computer.

f. Repeat step c.

g. All resistance readings shall indicate a short circuit.

h. Restore the aircraft to service.

22.0.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective system component refer to the following handbooks: Autopilot Manual, Sperry Part Number LJ-15-0142.

22.0.3.1 AFCS ENGAGE FUNCTION.

PROBABLE CAUSE	ISOLATION PROCEDURE
EDGELIGHTS ON FLIGHT CONTROLLER DO NOT LIGHT WHEN CIRCUIT BREAKERS ARE CLOSED.	
Cable not connected to J1000 on flight controller.	Examine cable and socket to ensure that there are no bent pins and connect cable securely to J1000.
Edgelight lamps defective.	Replace an edgelight lamp. If the lamp lights, replace the other three lights.

22.0.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
EDGELIGHTS ON FLIGHT CONTROLLER DO NOT LIGHT WHEN CIRCUIT BREAKERS ARE CLOSED (Cont).	
DC short circuit in flight controller.	Check that the AFCS circuit breakers have not returned to the "OFF" position. If a circuit breaker has returned to the "OFF" position, replace the flight controller.
Aircraft power failure.	Check for presence of 28-volts DC at J1000 pins 2 and 5 on cable connector. If voltage is not present, check aircraft wiring.
AFCS SWITCH WILL NOT REMAIN IN "ENG" POSITION.	
TURN knob not in detent before engagement of AFCS switch.	Place the TURN knob in detent.
Defective microswitch or solenoid in rudder, aileron, or elevator servo drive units.	Disconnect cable from J3100 on rudder servo drive unit and place a jumper across pins P and N on the aircraft cable connector. If AFCS switch will not engage, repeat jumper procedure on J3200-P and N (elevator servo-drive) and J3300-P and N (aileron servo-drive). Replace the (rudder, aileron or elevator) servo drive unit indicated to be defective by the isolation procedures.
Defective relay or thermal relay in vertical gyro.	Substitute with a known operative vertical gyro.
Defective relay or switch in flight controller.	Substitute with a known operative flight controller.
Aircraft trim disconnect switch open.	Close switch.
Release buttons defective.	Continuity check buttons.
Trim disconnect relay defective.	Substitute with a known operative relay.
Defective relay in flight control computer.	Substitute with known operative flight control computer.
Engage coupler defective.	Substitute with known operative engage coupler.

22.0.3.2 INTERLOCKS AND INDICATOR LAMPS.

PROBABLE CAUSE	ISOLATION PROCEDURE
AFCS SWITCH WILL NOT RETURN TO "OFF" POSITION WHEN PILOT'S OR COPILOT'S AFCS RELEASE PUSHBUTTON IS DEPRESSED.	
Pilot's AFCS release button defective.	Depress copilot's AFCS release button and observe whether AFCS switch disengages. If AFCS switch disengages, replace pilot's AFCS release button.

22.0.3.2 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
AFCS SWITCH WILL NOT RETURN TO "OFF" POSITION WHEN PILOT'S OR COPILOT'S AFCS RELEASE PUSHBUTTON IS DEPRESSED (Cont).	
Copilot's AFCS release button defective.	Depress pilot's AFCS release button and observe whether AFCS switch disengages. If AFCS switch disengages, replace copilot's AFCS release button.
Pilot's and copilot's AFCS release buttons defective.	Place EMERG TRIM DISCON switch in the "DISCON" position and observe whether AFCS switch disengages. If AFCS switch disengages, replace both pilot's and copilot's release buttons.
Solenoid in flight controller defective.	Substitute with a known operative flight controller.
PILOT'S EMERG TRIM DISCON SWITCH WILL NOT DISENGAGE THE AFCS SWITCH.	
Defective EMERG TRIM DISCON switch.	Depress pilot's or copilot's release button and observe whether AFCS switch disengages. If AFCS switch disengages, replace the EMERG TRIM switch.
Solenoid in flight controller defective.	Substitute with a known operative flight controller.
"ALT" PUSHBUTTON LIGHTS WHEN DEPRESSED, WITH AFCS SWITCH IN "OFF" POSITION.	
Defective relay in flight control computer.	Substitute with a known operative flight control computer.
Defective AFCS switch on flight controller.	Depress pilot's or copilot's release button. If "ALT" light goes out, replace flight controller.
"HDG SEL" AND "RADIO" PUSHBUTTONS LIGHT WHEN DEPRESSED, WITH AFCS SWITCH IN "OFF" POSITION.	
Defective AFCS switch on flight controller.	Substitute with a known operative flight controller.
AFCS SWITCH ENGAGES WITH TURN KNOB OUT OF DETENT.	
Defective relay or detent switch in flight controller.	Substitute with a known-operative flight controller.
"HDG SEL" OR "RADIO" PUSHBUTTON LIGHTS WHEN DEPRESSED, WITH AFCS SWITCH ENGAGED AND TURN KNOB OUT OF DETENT.	
Defective relay in flight controller.	Substitute with a known operative flight controller.
"HDG SEL" PUSHBUTTON DOES NOT LIGHT WHEN DEPRESSED, WITH SYSTEM ENGAGED, TURN KNOB IN DETENT, AND "RADIO" MODE NOT ENGAGED.	
"HDG SEL" pushbutton lamp burned out.	Depress "RADIO" pushbutton. If "RADIO" pushbutton lights, replace "HDG SEL" pushbutton lamp.
Defective relay or pushbutton in flight controller.	Substitute with a known operative flight controller.

22.0.3.2 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
"HDG SEL" PUSHBUTTON LIGHT DOES NOT GO OUT WHEN TURN KNOB ROTATED OUT OF DETENT.	
Defective detent switch or relays in flight controller.	Substitute with a known operative flight controller.
"RADIO" PUSHBUTTON DOES NOT LIGHT WHEN DEPRESSED WITH SYSTEM ENGAGED AND TURN KNOB IN DETENT.	
"RADIO" pushbutton lamp defective.	Depress "HDG SEL" pushbutton. If "HDG SEL" pushbutton lights, replace "RADIO" pushbutton lamp.
"RADIO" PUSHBUTTON LIGHT DOES NOT GO OUT WHEN TURN KNOB ROTATED OUT OF DETENT.	
Defective detent switch or relays in light controller.	Substitute with a known operative flight controller.
"ALT" PUSHBUTTON DOES NOT LIGHT WHEN DEPRESSED WITH SYSTEM ENGAGED.	
"ALT" pushbutton lamp burned out.	Replace "ALT" pushbutton lamp.
Defective relay in flight control computer.	Substitute with a known operative flight control computer.
Defective relay or pushbutton in flight controller.	Substitute with a known operative flight controller.
"ALT" PUSHBUTTON LIGHT DOES NOT GO OUT WHEN PITCH SWITCH OPERATED TO EITHER "DN" OR "UP" POSITION.	
Defective relay in flight control computer.	Substitute with a known operative flight control computer.
Defective relay in flight controller.	Substitute with a known operative flight controller.
"RADIO" PUSHBUTTON LAMP GOES OUT WHEN PITCH SWITCH IS OPERATED.	
Defective relay in flight control computer.	Substitute with a known operative flight control computer.
"RADIO" PUSHBUTTON LIGHT DOES NOT GO OUT WHEN PITCH SWITCH OPERATED AND RECEIVER IS TUNED TO AN ILS FREQUENCY.	
Defective relay in flight control computer.	Substitute with a known operative flight control computer.
Defective ILS receiver.	While receiver is tuned to an ILS frequency, check for 28-volts DC at connector JL800-35. If 28-volts DC is not present, receiver is defective.

22.0.3.3 MAXIMUM SERVO FORCES.

PROBABLE CAUSE	ISOLATION PROCEDURE
SERVO DRIVES CANNOT BE OVERPOWERED WHEN ENGAGED BY OPERATING CONTROL WHEEL OR RUDDER PEDALS.	
Defective elevator, or aileron, or rudder servomechanism.	Disengage AFCS and check that control switch, or control wheels, or rudder pedals will move control surface.
SERVO DRIVES CANNOT MOVE THE CONTROL SURFACE.	
Defective servo drive mechanism.	Disengage AFCS and check that control wheel, or rudder pedals will move control surfaces.

22.0.3.4 VERTICAL AND DIRECTIONAL GYROS.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH THE SYSTEM ENGAGED, VERTICAL GYRO IS ROTATED BY HAND TO SIMULATE AN AIRCRAFT ROLL, BUT CONTROL WHEEL DOES NOT ROTATE.	
Defective vertical gyro.	Rotate TURN knob out of detent while observing control wheel. If control wheel rotates when TURN knob is moved out of detent, replace vertical gyro.
Defective flight computer.	Substitute with a known operative flight control computer.
Defective aileron servo drive unit.	Observe aileron trim meter while vertical gyro is being rotated. If aileron trim meter deflects when vertical gyro is rotated, replace aileron servo drive unit.
WHEEL MOVES IN SAME DIRECTION AS GYRO (RIGHT ROLL OF GYRO SHOULD GIVE A WHEEL-TO-LEFT REACTION).	
Incorrect aircraft wiring.	Check wiring.
WITH SYSTEM ENGAGED, VERTICAL GYRO IS ROTATED BY HAND TO SIMULATE AN AIRCRAFT NOSE UP, BUT CONTROL COLUMN DOES NOT MOVE FORWARD.	
Defective elevator servo drive unit.	Observe elevator trim meter when vertical gyro is rotated. If elevator trim meter deflects when vertical gyro is rotated, replace elevator servo drive unit.
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective vertical gyro.	Operate PITCH switch and observe control column. If control column moves when PITCH switch is operated, replace vertical gyro.

22.0.3.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH SYSTEM ENGAGED, DIRECTIONAL GYRO IS ROTATED BY HAND TO SIMULATE A CHANGE IN AIRCRAFT HEADING, BUT CONTROL WHEEL DOES NOT ROTATE AND THERE IS NO INITIAL MOVEMENT OF RUDDER PEDALS.	
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective directional gyro.	Set C-6 to a heading about 10 degrees greater than the aircraft heading. Depress "HDG SEL" pushbutton. If control wheel turns when "HDG SEL" pushbutton is pushed, check directional gyro.

22.0.3.5 ELEVATOR, RUDDER, AND AILERON TRIM.

PROBABLE CAUSE	ISOLATION PROCEDURE
WHEN SYSTEM IS ENGAGED AND IS OVERPOWERED BY MOVING THE CONTROL WHEEL, OR RUDDER PEDAL, EITHER THE RUD, AIL, OR EL TRIM METERS FAIL TO INDICATE ANY MOVEMENT OF THE CONTROL SURFACES WHILE CONTROL SURFACES ACTUALLY MOVE.	
Defective rudder, or aileron, or elevator servo amplifiers.	Substitute with a known operative flight control computer.
Defective rudder, aileron, or elevator servo drive units.	Substitute with a known operative servo drive unit (rudder, aileron, or elevator) corresponding to the control surface movement which did not yield an indication on the trim meter.
Defective relays, or trim meter in flight controller.	Replace with a known operative flight controller.

22.0.3.6 TURN KNOB OPERATION.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH SYSTEM ENGAGED, CONTROL WHEEL DOES NOT ROTATE AS TURN KNOB IS ROTATED.	
Defective potentiometer in flight controller.	Set C-6 to a heading about 10 degrees greater than the aircraft heading. Depress "HDG SEL" pushbutton. If control wheel turns when "HDG SEL" pushbutton is pushed, replace flight controller.
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective aileron servo drive unit.	Substitute with a known operative aileron servo drive unit.

22.0.3.6 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH SYSTEM ENGAGED, CONTROL WHEEL MOVES IN WRONG DIRECTION AS TURN KNOB IS ROTATED. (CLOCKWISE ROTATION OF KNOB SHOULD RESULT IN RIGHT WHEEL MOVEMENT.)	
Defective aircraft wiring.	Continuity check aircraft wiring.

22.0.3.7 HEADING SELECT OPERATION.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH SYSTEM ENGAGED AND COMPASS INDICATOR SET TO A HEADING GREATER THAN THAT OF AIRCRAFT, CONTROL WHEEL FAILS TO ROTATE WHEN "HDG SEL" PUSHBUTTON IS DEPRESSED.	
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective aileron servo drive unit.	Rotate TURN knob out of detent while observing control wheel. If control wheel does not rotate, replace aileron servo drive unit.
Defective compass indicator.	Check compass indicator.

22.0.3.8 PITCH SWITCH OPERATION.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH THE SYSTEM ENGAGED, THE CONTROL COLUMN DOES NOT MOVE WHEN THE PITCH SWITCH IS OPERATED TO EITHER THE "DN" OR "UP" POSITION.	
Defective elevator servo drive unit.	Observe whether the EL trim meter deflects when the PITCH switch is operated. If the EL trim meter deflects when the PITCH switch is operated, replace the elevator servo drive unit.
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective switch or relay in flight controller.	Substitute with a known operative flight controller.
THE CONTROL COLUMN CONTINUES TO MOVE AFTER THE PITCH SWITCH IS RELEASED.	
Defective relay in flight controller.	Substitute with a known operative flight controller.

22.0.3.9 VOR OPERATION.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH LATERAL RADIO RECEIVER TUNED TO A DISTANT OMNISTATION (TO PRODUCE NO RADIO OUTPUT) AND "RADIO" PUSHBUTTON DEPRESSED, SELECTION OF A HEADING ON COURSE INDICATOR FAILS TO CAUSE CONTROL WHEEL TO ROTATE.	
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective aileron servo drive unit.	Observe AIL trim meter for deflection while heading is being set on Course Indicator. If AIL trim meter deflects, replace servo drive unit.

22.0.3.10 ILS OPERATION.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH GLIDEPATH RECEIVER TUNED TO AN ILS FREQUENCY, PRODUCING A DOWNWARD DEFLECTION OF GLIDESLOPE POINTER ON APPROACH HORIZON INDICATOR, CONTROL COLUMN FAILS TO MOVE FORWARD WHEN "RADIO" PUSHBUTTON IS DEPRESSED.	
Defective flight control computer.	Substitute with a known operative flight control computer.
Defective elevator servo drive unit.	Observe EL trim meter for deflection when "RADIO" pushbutton is depressed. If EL trim meter deflects, replace the elevator servo drive unit.

22.0.3.11 ALTITUDE HOLD.

PROBABLE CAUSE	ISOLATION PROCEDURE
WITH SYSTEM ENGAGED AND "ALT" PUSHBUTTON DEPRESSED, DECREASED PRESSURE TO ALTITUDE CONTROL FAILS TO MOVE CONTROL COLUMN FORWARD.	
Defective flight control computer.	Substitute with known operative flight control computer.
Defective elevator servo drive unit.	Observe EL trim meter for deflection while pressure to altitude control is being decreased. If EL trim meter deflects, replace elevator servo drive unit.
Defective altitude control.	Substitute with known operative altitude control.

22.0.3.12 LINEAR ACCELEROMETERS.

PROBABLE CAUSE	ISOLATION PROCEDURE
PLACING A TEST MAGNET OF APPROXIMATELY 600 GAUSS IN POSITION AGAINST FORE OR AFT LINEAR ACCELEROMETER FAILS TO CAUSE RUDDER PEDALS TO MOVE.	
Defective fore or aft linear accelerometer.	Substitute known operative linear accelerometer for the accelerometer which fails to produce movement of rudder pedals.
Defective flight control computer.	Substitute known operative flight control computer.
Defective rudder servo drive unit.	Substitute known operative rudder servo drive unit.

22.0.4 REMOVAL AND INSTALLATION.

22.0.4.1 REMOVAL OF ACCELEROMETERS.

a. Fore yaw accelerometer.

  (1) Locate the fore yaw accelerometer installed on the top shelf of the nose electronic equipment rack.

  (2) Disconnect the 69-1R16S-1S plug.

  (3) Remove (3) NAS1291-03 nuts, (3) AN960D8 washers, (2) NAS1632-8 screws, and (1) NAS1632-9 screw.

  (4) Remove accelerometer from aircraft.

b. Aft yaw accelerometer.

  (1) Locate the aft yaw accelerometer installed on the centerline of the aircraft at fuselage station 480 (approx.) and W.L. 60.

  (2) Disconnect the 69-1R16S-1S plug.

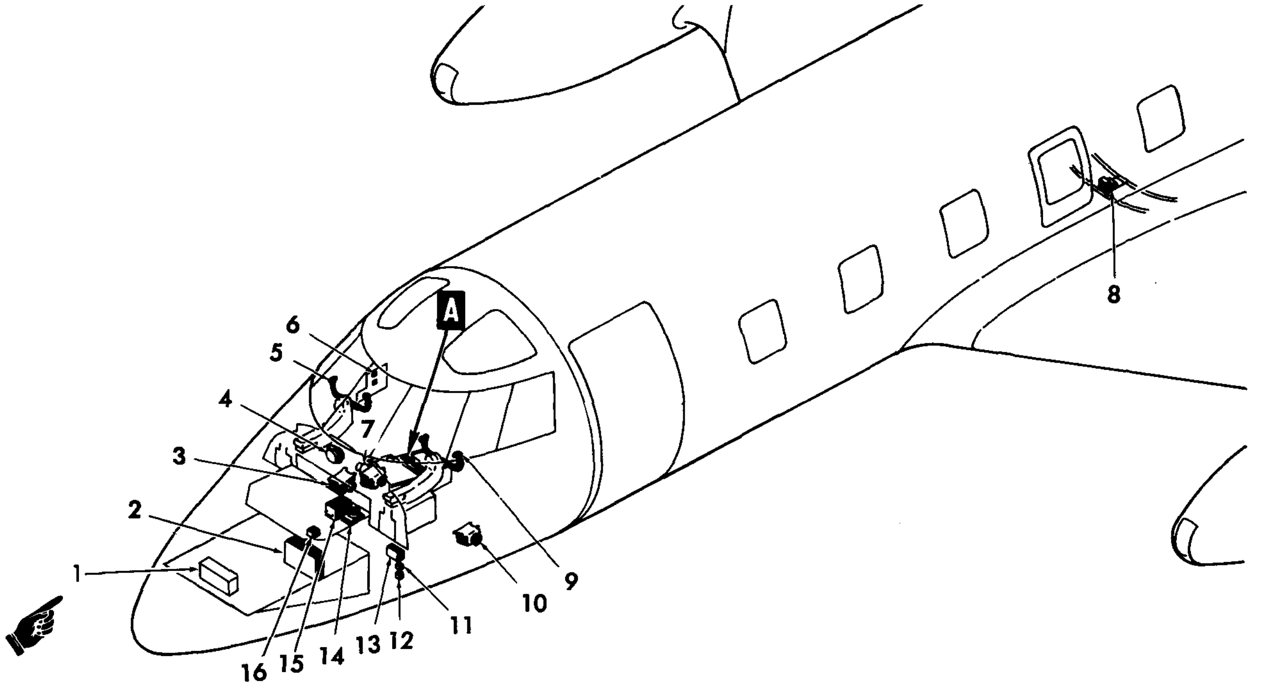
  (3) Remove (1) LS5907-08-8 screw and (2) LS5907-08-7 screws.

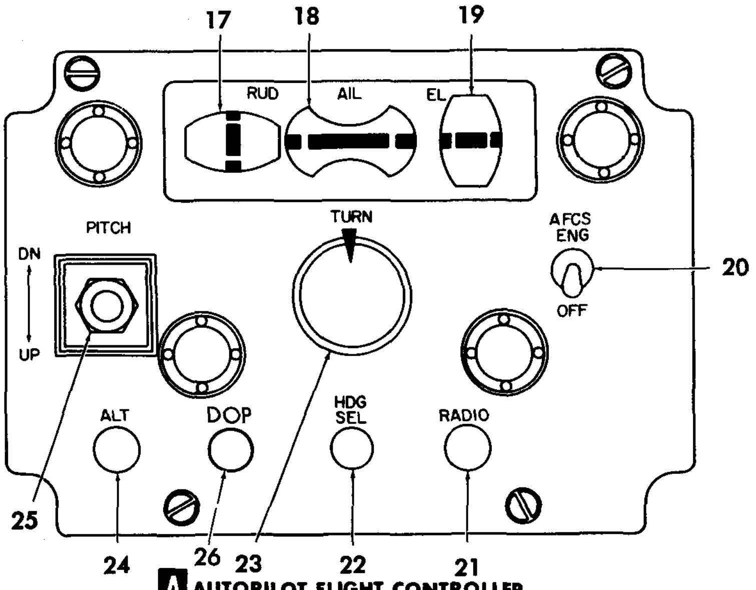
  (4) Remove accelerometer from aircraft.

22.0.4.2 INSTALLATION OF ACCELEROMETERS. Install the fore and aft yaw accelerometers in their respective locations with two feet touching the "stop angle" which is installed parallel to two mounting holes using the hardware listed above.

22.0.4.3 REMOVAL OF AUTOPILOT SERVOS. The rudder, elevator and aileron autopilot servos are located under the cockpit floor between fuselage station 199.0 and 213.0. It is necessary to remove the copilot's seat and seat deck to reach the rudder and elevator servos and the pilot's seat and seat deck to remove the aileron servo. After access is gained, disconnect the electrical plug. It is possible to remove the servo drive unit without removing the servo bracket and drum, therefore not affecting cable rigging. Six screws, located on the outer periphery of the motor side of the servo drive unit case attach the drive unit to the servo bracket. Remove these screws and slide drive unit out. Note on the aileron and rudder units it may be necessary to remove the four servo bracket mounting bolts in order to shift unit around to obtain working room to remove the six attach screws.

22.0.4.4 INSTALLATION OF AUTOPILOT SERVOS. Reverse removal procedure. After units have been reinstalled, inspect cables for proper installation. See Chapter 27 for installation and rig tension information.





A **AUTOPILOT FLIGHT CONTROLLER**
(LOOKING DOWN)

1.	ENGAGE COUPLER
2.	AUTOPILOT FLIGHT CONTROL COMPUTER
3.	RUDDER AUTOPILOT SERVOMOTOR
4.	AUTOPILOT AIRSPEED DIFFERENTIAL PRESSURE SWITCH
5.	COPILOT'S AUTOPILOT DISCONNECT PUSHBUTTON
6.	AUTOPILOT TRANSFER RELAY
7.	ELEVATION AUTOPILOT SERVOMOTOR
8.	AFT YAW ACCELEROMETER
9.	PILOT'S AUTOPILOT DISCONNECT PUSHBUTTON
10.	AILERON AUTOPILOT SERVOMOTOR
11.	VERTICAL GYRO NO. 2 AC INTERLOCK RELAY
12.	VERTICAL GYRO NO. 2 DC INTERLOCK RELAY
13.	VERTICAL GYRO NO. 2 PHASE ADAPTER
14.	VERTICAL GYRO NO. 2
15.	AUTOPILOT ALTITUDE CONTROL
16.	FORE YAW ACCELEROMETER
17.	RUDDER TRIM METER
18.	AILERON TRIM METER
19.	ELEVATOR TRIM METER
20.	AFCS SWITCH
21.	RADIO MODE SELECTOR
22.	HDG SEL MODE SELECTOR SWITCH
23.	TURN CONTROL
24.	ALT MODE SELECTOR SWITCH
25.	PITCH CONTROL
26.	DOPPLER MODE SELECTOR SWITCH (ON AIRPLANES WITH DOPPLER RADAR SYSTEM ONLY)

JS-1-2-X0/8-2201

Figure 22-1. Autopilot System Components (SPERRY)

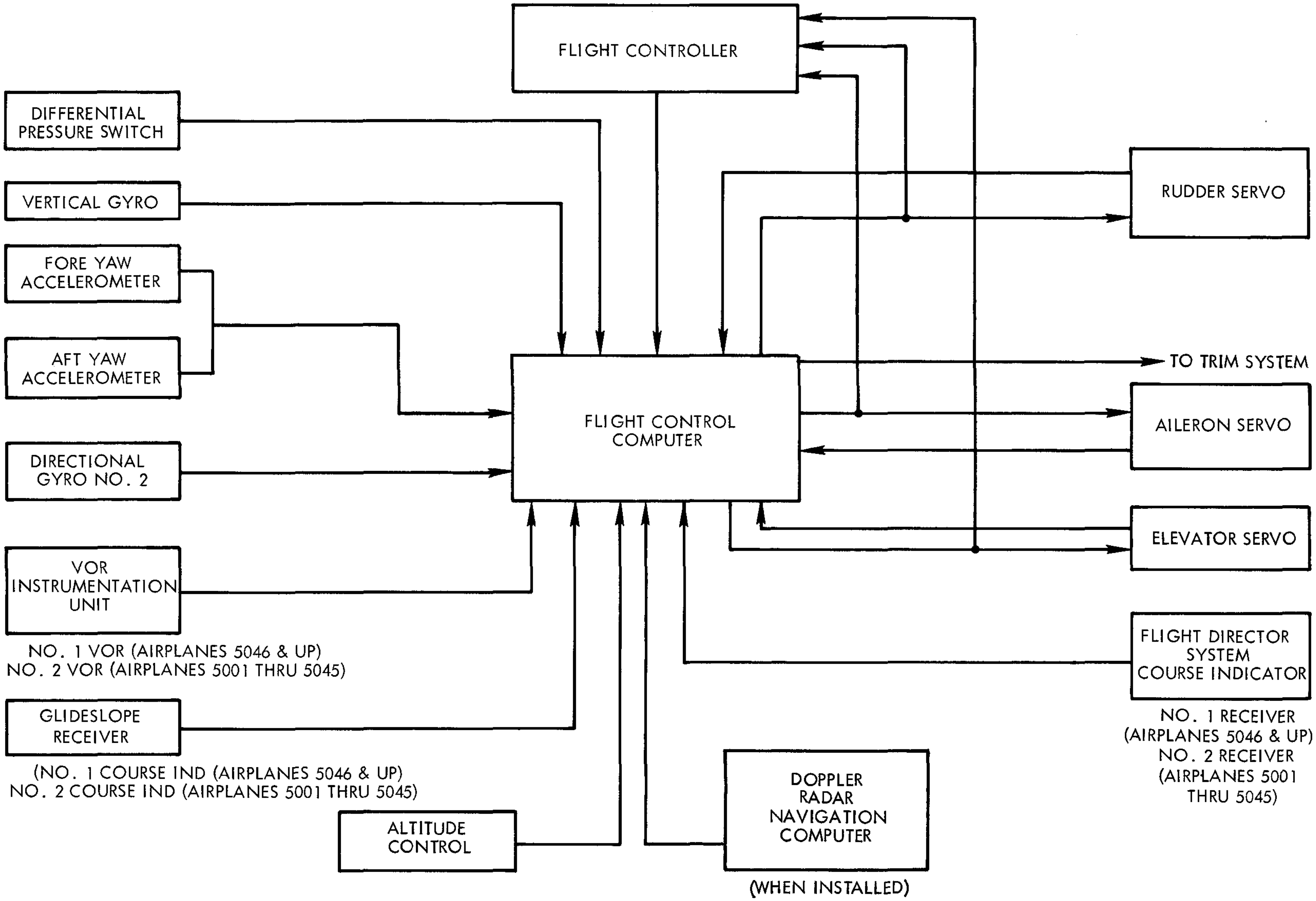

FLIGHT CONTROLLER
DIFFERENTIAL PRESSURE SWITCH
VERTICAL GYRO
FORE YAW ACCELEROMETER
AFT YAW ACCELEROMETER
RUDDER SERVO
FLIGHT CONTROL COMPUTER
TO TRIM SYSTEM
AILERON SERVO
DIRECTIONAL GYRO NO. 2
ELEVATOR SERVO
VOR INSTRUMENTATION UNIT
NO. 1 VOR (AIRPLANES 5046 & UP)
NO. 2 VOR (AIRPLANES 5001 THRU 5045)
FLIGHT DIRECTOR SYSTEM COURSE INDICATOR
NO. 1 RECEIVER (AIRPLANES 5046 & UP)
NO. 2 RECEIVER (AIRPLANES 5001 THRU 5045)
GLIDESLOPE RECEIVER
(NO. 1 COURSE IND (AIRPLANES 5046 & UP)
NO. 2 COURSE IND (AIRPLANES 5001 THRU 5045)
ALTITUDE CONTROL
DOPPLER RADAR NAVIGATION COMPUTER
(WHEN INSTALLED)

Figure 22-2. Autopilot System Block Diagram

SECTION 22.1

AUTOPILOT

AP-104 (COLLINS)

Contents Page

DESCRIPTION AND OPERATION 22-19

GENERAL 22-19
  Autopilot Engaged 22-20
  Lateral Modes 22-20
  Vertical Modes. 22-21
  List of Components 22-22
    Yaw Rate Gyro 22-22
    Autopilot Amplifier 22-22
    Servomotors 22-22
    Trim Coupler 22-22
    Trim Indicator 22-23
    Vertical Gyro 22-23
    Autopilot Controller 22-23
    Pitch and Roll Rate Sensor 22-23

OPERATIONAL CHECKOUT 22-23
  Test Equipment Required 22-23
  Autopilot System 22-23

TROUBLESHOOTING 22-27

REMOVAL AND INSTALLATIONS 22-27
  Removal of Autopilot Servos 22-27
  Installation of Autopilot Servos 22-28

PRIMARY SERVO, SERVO TORQUE SETTINGS 22-28

TRIM COUPLER, GAIN SETTING 22-28

FIGURES
  22-3 Autopilot System Components 22-29
  22-4 Primary Servo, Torque Setting Location Diagram 22-30
  22-5 Spanner Wrench Fabrication Diagram 22-31
  22-5A Air Data Computer Self Test (590D- 1) 22-32
  22-6 Collins Autopilot System Block Diagram 22-33

22.1.1 DESCRIPTION AND OPERATION.

22.1.1.1 GENERAL. The following is a functional theory of operation in which the system is described from an operational point of view. The 614E-4 Autopilot Controller, see figure 22-6, is the control center for the autopilot. It provides the autopilot mode and command functions. The function of the various controls are as follows:

a. ENGAGE Lever.

The ENGAGE lever is used to engage or disengage the autopilot. It is the master disconnect for the autopilot. It is also used to engage the yaw damper when YAW DAMP is selected as an independent function. The switch is solenoid held and spring loaded to the DISENGAGE position.

b. YAW DAMP Switch.

The YAW DAMP switch is a push-on/push-off solenoid held switch. It is used to select yaw damping action independent of autopilot operation. When the switch is in the ON position, as indicated by a mechanically actuated flag, the engage lever engages only the rudder channel of the autopilot.

c. Autopilot Manual Switch.

The autopilot manual switch is a push-on/push-off switch used to uncouple the autopilot from flight director guidance. A mechanically actuated flag appears when the switch is in the AP MAN position.

d. Vertical Command Control.

The vertical command control provides either vertical speed command or pitch command. The

22.1.1.1 (Cont).

nose is lowered when the control is rotated toward DN and raised when the control is rotated toward UP. In the pitch mode of operation, the control commands aircraft pitch attitude, except when VS mode is selected. Control rotation causes any selected vertical mode pushbutton to release to the off position, giving pitch mode. When VS mode is selected, the control commands vertical speed. Control rotation from the UP index to the DN index will command a change of approximately 500 feet per minute.

If a vertical rate of less than 250 feet per minute is commanded, the system automatically switches to altitude hold.

e. Turn Control.

The turn control is used to command bank when the autopilot is not coupled to the flight director. The control has a center detent that provides a compass heading hold.

22.1.1.1.1 AUTOPILOT ENGAGED. When the autopilot is engaged, it has two basic modes of operation. It is either COUPLED TO or UNCOUPLED from the flight director. When COUPLED, the autopilot accepts guidance commands from the flight director. When ENGAGED and UNCOUPLED it accepts pitch and roll commands from the Vertical command and turn controls. It may be uncoupled either by turning the flight director OFF or by selecting the autopilot "AP MAN" button on the flight controller. "AP MAN" allows the pilot to control the airplane attitude while observing the computed commands on the command bars. Recoupling the autopilot will clear any selected vertical modes.

The autopilot may be engaged in any reasonable attitude and in either the coupled or uncoupled mode. When coupled, the autopilot will smoothly acquire the commanded attitude. Engagement will automatically clear any selected vertical mode. When uncoupled, the autopilot will maintain the heading and pitch attitude existing at the time of engagement. The autopilot may be engaged with the turn knob out of detent, but if it is, the knob must be returned to the detent before bank can be commanded.

The autopilot may be disengaged by the following:

a. Actuation of the AP disengage buttons on the wheel.

b. Moving the ENGAGE lever to the DISENGAGE position.

c. Selection of go-around mode. (Optional)

The following functions will cause the autopilot to automatically disengage:

a. Gyro monitor failure.

b. Autopilot power or circuit failure.

When coupled to the flight director, the autopilot will revert back to heading and pitch hold if a failure occurs in the guidance computation. The autopilot will remain in the hold modes until the failure clears or another mode is selected. The pitch and heading hold condition is indicated by the UNCPLD annunciator.

The YAW DAMP button provides selection of an independent yaw damper. The yaw damper is engaged by depressing the YAW DAMP button on the autopilot controller and putting the ENGAGE lever in the ENGAGE position. Operation of the pilot's or copilot's A/P disconnect button will cause the engage lever to return to disengage and the yaw damper button to off.

22.1.1.1.2 LATERAL MODES. The autopilot lateral modes are as follows:

a. Heading.

When the HDG mode is selected, the autopilot will turn the aircraft to and maintain the heading set on the HEADING marker.

b. VOR/LOC.

When the VOR/LOC mode is selected, the system initially switches to a VOR/LOC arm submode condition. During this period, the autopilot commands the aircraft to follow the heading shown by the HDG marker. The HDG marker is set to produce the desired VOR or localizer intercept angle.

With the aircraft on the desired intercept heading, the flight computer computes a capture point based on deviation from the desired radio beam, the rate at which the aircraft is approaching this radio beam, and the course intercept angle. The capture point occurs at the moment the annunciators show a change from V/L ARM to V/L CAPT. At this time, the autopilot will turn the aircraft to track the selected radio course and provide automatic crosswind corrections. In VOR, radio signal commands are suppressed to provide smooth station passage.

c. APPR I and APPR II.

When in APPR I or APPR II, localizer capture is accomplished as described for the VOR/LOC mode. The APPR mode also provides automatic glideslope arm and capture.

Once the APPR mode has been selected, reception of a satisfactory glideslope signal will cause the GS ARM annunciator to light. Glideslope capture is independent of localizer capture and thus may occur before, after, or at the same time as localizer capture.

Prior to glideslope capture, any vertical mode can be selected. If a vertical mode is selected, it will automatically switch off with glideslope capture.

Glideslope capture may be accomplished from either above or below the glideslope and will be

22.1.1.1.2 (Cont).

signified by a change in annunciation from GS ARM to GS CAPT.

The glideslope gain is programmed as a function of radio altitude. The automatic gain programming begins at 1000 feet above the ground. In the event a radio altimeter is not used or malfunctions, the middle marker signal provides backup gain change called glideslope extension.

d. Go-Around. (Optional)

Go-around is a flight director function only and is selected with the GA button on the control wheel. Actuation of the GA button causes the autopilot to disengage. The computed steering presented to the pilot on the command bars is a wings-level and predetermined pitch-up command. GA may be selected any time after APPR I or APPR II mode is selected. The GA annunciator on the flight director indicator lights when in the GA submode.

22.1.1.1.3 VERTICAL MODES. The autopilot vertical modes are as follows:

In addition to the lateral modes, there are standard vertical modes. They are pitch CMD, IAS hold, ALT hold, VS CMD, and MACH hold.

The pitch CMD mode is selected by having all of the vertical mode select buttons in the OFF position. Each of the other vertical modes is selected with its respective button.

When manually flying the airplane with the autopilot disengaged, all of the vertical modes are HOLD MODES.

22.1.1.1.4 THE ANNUNCIATORS. The annunciator (when installed) will light under the following conditions:

ANNUNCIATOR	CONDITION UNDER WHICH IT LIGHTS
UNCPLD	Autopilot engaged and uncoupled from the flight director.
TRIM FAIL	Failure of the autopilot trim circuits.
*DISENG.	Autopilot is disengaged from the airplane controls.
*ELEV. TRIM	Elevator is mistrimmed; annunciator light will go out when pitch trim is complete.
N/L ARM	Prior to beam capture, after APPR or VOR/LOC mode has been selected.
N/L CAPT	After beam capture in APPR or VOR/LOC mode.
GS ARM	Prior to beam capture after a satisfactory glideslope is being received in a front course approach.
GS CAPT	After beam capture in APPR mode on a front course approach.
*GS EXT	After the middle marker on a front course approach and in APPR mode in the event of a radio altimeter failure.
B/L	Selected course differs from airplane heading more than 105° approx. when APPR I mode is selected (back localizer).
GA	Selection of GO-AROUND sub mode.
MDA	Descent below preselected radio altitude.
*IAS	Selection of INDICATED AIRSPEED HOLD (vertical mode).
*VS ON	Selection of VERTICAL SPEED mode (vertical mode).
*ALT	Selection of ALTITUDE HOLD mode.
*MACH	Selection of MACH HOLD (vertical mode).
*NAV	Selection of DOPPLER mode.

*Optional Annunciators

22.1.1.2 LIST OF COMPONENTS

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Trim Coupler	1	Collins	161A-1B	522-4649-001
Trim Indicator	1	Collins	327D-1W	458-0405-00
Yaw Rate Gyro	1	Collins	332G-2	229-0236-00
Primary Servo	3	Collins	334C-3B	772-5374-001
Vertical Gyro	1	Collins	332D-11	522-3985-001
Autopilot Amplifier	1	Collins	562C-4A	772-5131-002
Autopilot Controller	1	Collins	614E-4	522-3551-001
Primary Servo Mount	3	Collins	351B-3	522-1226-001
Sensing Unit Rate	1	Collins	345A-4B	522-3851-00
Transfer Relay	1			MS25268-D1
Interlock Relay	1	Hartman		CR16AF
Disconnect Relay	1	GE		3SAE5041K1

22.1.1.2.1 YAW RATE GYRO. The yaw rate gyro senses airplane yaw. Its electrical output is inserted into the autopilot amplifier. The autopilot amplifier output is fed to the rudder servo for yaw damping.

22.1.1.2.2 AUTOPILOT AMPLIFIER. The autopilot amplifier receives inputs from the: Flight computer, vertical gyro, compass, air data, pitch and roll rate sensor, yaw rate gyro, flight controller. The autopilot amplifier provides the power outputs required, to drive the control surface servos.

22.1.1.2.3 SERVOMOTORS. Signals from the autopilot amplifier operate three servomotors. The servos position the control surfaces of the airplane. Each servo contains a motor drive, a torque-limiting clutch, an engage clutch, and a position feedback syncro. The motor drive and the engage clutch, which is solenoid operated, are powered by 28V DC. The torque-limiting clutch in each servo is adjusted manually and mechanically to safe limits which can be overpowered by the pilot. The torque values differ for each axis and are not normally adjusted to the correct value when shipped from the vendor. It is imperative that the servos be installed in the appropriate location in the airplane. The torque set values for each servo clutch are as follows:

a.	Rudder	125 inch - pounds
b.	Aileron	90 inch - pounds
c.	Elevator	70 inch - pounds

After adjusting the servo clutches to the above values, these values and the common control surface are to be metal stamped on the appropriate plates on the servo-drive and the servo-mount. The torque wrench accuracy is within ±2.0 inch-pounds. The force required to overpower the servomotors under all conditions of operation, does not exceed the following values, excluding mechanical input friction:

a. 53 pounds force on one rudder pedal.

b. 32 pounds up and 39 pounds down force applied normal to the column at the centerline of the control wheel.

c. 157 inch-pounds force applied tangentially to a control wheel.

22.1.1.2.4 TRIM COUPLER. The trim coupler links the autopilot to the electrical trim system of the airplane, and drives the horizontal stabilizer thru the pitch trim actuator. The trim coupler has two isolated floating DC signal input terminals. One used for the trim-up signal, the other for the trim-down signal. The signals are supplied by the autopilot amplifier. Four 28V DC outputs are provided by the trim coupler. These are trim-up arm, trim-up, trim-down arm and trim-down. Maximum load 2.0 amperes at 20.0 volts. Output current is the sum of the trim and trim-arm currents. The trim coupler outputs are interlocked to preclude the possibility of a trim-up and a trim-down at the same time. To obtain a trim-up output a trim-up arm output and the absence of a trim-down arm output is required. Conversely; to obtain a trim-down output, a trim-down arm output and the absence of a trim-up arm output is required. If both the trim directions are commanded at the same time, both trim-arm outputs are actuated which results in no trim outputs. A 28V DC output is provided for failure warning when the following occurs:

22.1.1.2.4 (Cont).

a. Loss of 115 V AC power.

b. Failure of the maximum trim rate limiting circuitry.

c. Actuation of both trim-arm outputs simultaneously.

The following adjustments are available when the dust cover is removed:

a. Trim-up threshold.

b. Trim-down threshold.

c. Maximum trim rate.

22.1.1.2.5 TRIM INDICATOR. The trim indicator reflects the direction of the servo force being applied to a control surface. It does not reflect the position of the surface.

22.1.1.2.6 VERTICAL GYRO. The vertical gyro is the source of pitch and roll data for the autopilot and flight director system.

22.1.1.2.7 AUTOPILOT CONTROLLER. The autopilot controller contains the operating devices for the AP-104 autopilot. The functions of these controls are described in paragraph 22.1.1.

22.1.1.2.8 PITCH AND ROLL RATE SENSOR. The pitch and roll rate sensor consists of two rate gyros. The outputs are used by the autopilot in stabilizing the airplane. Each gyro consists of an induction motor mounted within a support ring. The motor and support ring can tilt in either direction. A potentiometer wiper arm is mounted to the support ring. At zero rate input, the wiper is positioned at the center of the potentiometer winding, which is the reference position. As an input rate is applied the gyro precesses and the wiper moves from the center position. The voltage change which results is a function of the input rate. This output is fed thru the autopilot amplifier to the elevator and aileron servos.

22.1.2 OPERATIONAL CHECKOUT.

22.1.2.1 TEST EQUIPMENT REQUIRED.

NOMENCLATURE	DESCRIPTION	USE AND APPLICATION
Collins 479T-2	Signal Generator	Supplies deviations/directional signals to receiver.

22.1.2.2 AUTOPILOT SYSTEM. Switching and interlock checks.

a. Ensure all control surfaces are free to move.

b. Connect external power to the airplane (Refer to Chapter 24) and energize the busses.

c. Make certain the following circuit breakers are closed: VERTICAL GYRO NO. 1, AUTOPILOT 115VAC, AUTOPILOT 28VDC (5A), ELEVATOR 28VDC (5A), RUDDER AND AILERON 28VDC (7.5A), COMPASS NO. 1 26VAC, COMPASS NO. 1 115VAC, FLIGHT DIRECTOR NO. 1 28VDC. ATT. AND HDG. 115VAC, GLIDESLOPE NO. 1 28VDC AND 26VAC, VOR NO. 1 28VDC AND 26VAC, COUPLER 115VAC. Confirm normal aircraft, boost, trim, etc., circuit breakers are closed. Allow five minutes warm up for systems to stabilize. The VG flag on the flight director should mask.

d. Check the autopilot trim indicators for centering. If an indice continues to move or is more than a half bar width off center, discontinue the test until the malfunction is corrected.

e. Ensure all mode buttons reflect DISENGAGEMENT on the FLIGHT MODE SELECTOR NO. 1. Set the ENGAGE switch to the ENGAGE POSITION. The switch shall latch in.

f. On the pilot's control wheel DISENGAGE the autopilot. (Use the A/P trim disc. switch.) The ENGAGE switch should move to DISENGAGE. Repeat for copilot.

g. Move the ENGAGE switch to the ENGAGE position. On the pilot's control wheel DEPRESS the TRIM DISC. switch. The autopilot shall DISENGAGE. Repeat g. for copilot's side.

h. On the center console, depress the TRIM SYSTEM RESET switch. ENGAGE the autopilot.

j. On the pilot's control wheel DEPRESS the TRIM SWITCH to NOSE DOWN, the horizontal stabilizer shall not move. DEPRESS the TRIM SWITCH to NOSE UP, the horizontal stabilizer shall not move.

k. Deleted

l. Open the VERTICAL GYRO No. 1 circuit breaker. The autopilot shall DISENGAGE and the GYRO FLAG on the pilot's flight director indicator will be in view.

m. Close the VERTICAL GYRO NO. 1 circuit breaker. ENGAGE the autopilot after the flight director indicator GYRO FLAG masks.

n. Open the AUTOPILOT 115VAC circuit breaker. The autopilot shall DISENGAGE.

p. Close the AUTOPILOT 115VAC circuit breaker. ENGAGE the autopilot.

q. Deleted

r. Deleted

22.1.2.2 (Cont).

s. Deleted

t. Deleted

u. On the center console select NOSE UP or NOSE DOWN using the EMERGENCY TRIM switch. The autopilot shall DISENGAGE. Reset the emergency trim switch to normal.

v. On the center console ROTATE the AUTOPILOT TURN KNOB out of DETENT. ENGAGE the autopilot.

w. With VHF NAV NO. 1 on tune to a Nav. frequency. DEPRESS the following No. 1 mode selector buttons. NAV., HDG., VOR LOC, APPR I, APPR II, MACH, IAS, VS, ALT. They shall latch in.

x. Place the AUTOPILOT TURN KNOB in CENTER DETENT. DEPRESS the vertical mode buttons on the NO. 1 MODE SELECTOR. The vertical mode buttons shall not latch in unless a lateral mode is first engaged.

y. With autopilot still ENGAGED, turn knob in DETENT, YAW DAMPER and AP MAN modes off, ENGAGE the NAV mode. ROTATE the A/P TURN knob, the NAV mode shall not DISENGAGE.

z. ROTATE the TURN KNOB to CENTER DETENT. ENGAGE HDG mode. ROTATE TURN KNOB out of DETENT. HDG mode shall not DISENGAGE.

aa. ROTATE the TURN KNOB to CENTER DETENT. With VHF NAV RECEIVER NO. 1 ON, select a localizer frequency. Ensure NAV FLAG OUT OF VIEW.

ab. ENGAGE VOR LOC mode. ROTATE TURN KNOB, VOR LOC mode shall not DISENGAGE.

ac. ROTATE TURN KNOB to CENTER DETENT and ENGAGE APPR I mode on NO. 1 FLIGHT MODE SELECTOR. ROTATE TURN KNOB and APPR I mode shall not DISENGAGE.

ad. ROTATE TURN KNOB to CENTER DETENT and ENGAGE APPR II mode. ROTATE TURN KNOB and APPR II mode shall not DISENGAGE. Return turn knob to center detent.

ae. On the flight mode selector ENGAGE HDG and MACH. The buttons shall latch on.

af. Ensure the autopilot is ENGAGED. YAW DAMPER and AP MAN buttons are OFF. ROTATE the VERT CMD control UP or DOWN. The MACH mode shall DISENGAGE.

ag. Repeat (af) for IAS mode.

ah. Repeat (af) for ALT mode.

ai. Repeat (af) for VS Mode. The VS Mode shall stay LATCHED. The control column shall move as commanded.

aj. With NO. 1 VHF NAV receiver ON, select a usable VOR frequency.

ak. Select VOR LOC ON flight mode selector. DEPRESS THE VOR/ILS test on the VHF NAV control panel. The VOR/ILS test shall be inoperative.

al. On the front of the autopilot amplifier short the AILERON and ELEVATOR HIGH PASS JACKS. Check the aircraft DC bus voltage (at the power shield). This voltage should be 28.5 +.5, -0+DC for this test.

am. With the autopilot ENGAGED, use the aircraft clock to TIME the stabilizer travel thru 8.0 degrees, by overpowering the pitch axis. Stabilizer trim speed - Flaps UP 4° ±10% per minute.
- Flaps DOWN 8° ±10% per minute.

Note

The trim rate is adjustable on the 161A-1 (B) trim coupler.

an. DISENGAGE the autopilot.

ao. REMOVE the mounting bolts from the NO. 1 VERTICAL GYRO and ENGAGE the autopilot in The AP MAN mode.

ap. TILT the vertical gyro, simulating aircraft roll to the RIGHT. The control wheels shall ROTATE CCW. The control column shall move forward then AFT.

aq. LEVEL the VERTICAL GYRO and the controls shall return to neutral.

ar. TILT the vertical gyro to the LEFT. The control wheels shall ROTATE CW. The control column shall move forward then AFT.

as. TILT the vertical gyro UP and DOWN in the pitch axis. The control column shall move FORWARD then AFT. RE-INSTALL the vertical gyro.

at. REMOVE the mounting screws from the PITCH and ROLL RATE SENSING unit.

au. ROTATE the SENSING UNIT to simulate an aircraft movement as follows:

RIGHT ROLL - Control wheels momentarily ROTATE CCW.
LEFT ROLL - Control wheels momentarily ROTATE CW.
PITCH DOWN - Column momentarily moves AFT.
PITCH UP - Column momentarily moves FWD.

Controls will return to NEUTRAL when sensing unit is HELD STEADY.

REINSTALL the sensing unit.

22.1.2.2 (Cont)

av. REMOVE the mounting screws from the YAW RATE GYRO.

avA. Engage the autopilot, ensure AP MAN and all other modes are DISENGAGED.

aw. ROTATE the YAW RATE GYRO in azimuth as follows:

Forward face CLOCKWISE - control wheels ROTATE CCW, left rudder pedal moves FORWARD.

Forward face COUNTERCLOCKWISE - control wheels ROTATE CW, right rudder pedal moves FORWARD.

awA. Repeat (aw) with HDG mode selected.

ax. DISENGAGE the autopilot - AP MAN and ENGAGE YAW DAMP.

ay. The controls should move freely in ROLL and PITCH. ROTATE the YAW RATE GYRO CW and verify the left rudder pedal moves FORWARD.

RETURN the YAW RATE GYRO to its NORMAL POSITION.

az. DEPRESS the pilot's right rudder pedal approximately half its travel and hold. The autopilot servo force on the pedal will gradually wash out.

REPEAT for left rudder.

REINSTALL the YAW RATE GYRO.

ba. Deleted

bb. On the autopilot controller DISENGAGE YAW DAMP. Assure autopilot is engaged, turn knob in DETENT. HDG mode ENGAGED.

bc. Deleted

bd. SLEW the NO. 1 COMPASS in an increasing heading direction. The control wheels will ROTATE CCW.

be. SLEW the NO. 1 COMPASS in a decreasing heading direction. The control wheels will ROTATE CW.

bf. ROTATE the NO. 1 flight director indicator COURSE KNOB. The autopilot should not RESPOND.

bg. Set the autopilot turn knob out of detent. SLEW the COMPASS NO. 1 in either direction. The autopilot should NOT RESPOND. Return the turn knob *to center detent.*

bh. Ensure autopilot ENGAGED in AP MAN mode, YAW DAMP OFF. ROTATE the TURN KNOB out of detent, CLOCKWISE. The control wheels WILL ROTATE in the SAME DIRECTION as the commanded turn. RETURN THE TURN KNOB to DETENT.

bj. Ensure autopilot ENGAGED in AP MAN mode. ROTATE the VERT CMD control as follows:

DOWN DIRECTION - Control column will MOVE FORWARD.

UP DIRECTION - Control column will MOVE AFT.

bk. On the FLIGHT MODE SELECTOR NO. 1 ENGAGE the following modes:

HDG & ALT - ROTATE THE VERT CMD CONTROL in either direction The ALT mode will DISENGAGE.

IAS - *ROTATE the VERT CMD CONTROL in* either direction - The IAS mode will DISENGAGE.

VS - ROTATE the VERT CMD control in either direction - The VS mode WILL REMAIN ENGAGED. The column WILL FOLLOW the COMMAND.

bl. DISENGAGE and REENGAGE the autopilot - ENGAGE AP MAN. The VS mode shall DISENGAGE. ENGAGE the ALT mode. DISENGAGE and REENGAGE the autopilot; the ALT mode shall DISENGAGE. ENGAGE the IAS mode. DISENGAGE and REENGAGE the autopilot; the IAS mode shall DISENGAGE.

bm. With the autopilot ENGAGED, AP MAN and YAW DAMP DISENGAGED, ROTATE the No. 1 course indicator HDG. SELECT BUG to the aircraft heading. ENGAGE the HDG. BUTTON on the flight mode selector. ROTATE the HDG "BUG" in a CCW direction. The control wheels shall ROTATE CCW. ROTATE the HDG "BUG" in a CW direction. The control wheels shall ROTATE CW. Return the HDG "BUG" to the aircraft heading.

bp. ROTATE the COURSE SELECT POINTER CW and CCW from the aircraft heading. The autopilot shall not RESPOND.

22.1.2.2 (Cont)

bq. Deleted

br. On the signal generator set up the following:

(a) LOC GS selector to RIGHT.

(b) AUDIO selector to TONE LOC.

(c) MEGAHERTZ dial to an OPEN LOCALIZER frequency.

bs. Set the No. 1 VHF NAV receiver to the SIGNAL GENERATOR FREQUENCY. The NO. 1 course indicator NAV FLAG should be OUT OF VIEW. Engage autopilot, select APPR I, ROTATE the HEADING "BUG" CW and CCW. The control wheels shall FOLLOW THE "BUG."

bt. ROTATE the signal generator COURSE DEVIATION CONTROL towards ON-COURSE. At the point of capture the autopilot should respond and the control wheels ROTATE CLOCKWISE.

Note

Capture depends on radio deviation rate.

bu. ROTATE signal generator course DEVIATION CONTROL to RIGHT and LEFT of ON-COURSE. The control wheels SHOULD FOLLOW the signal. Return the signal generator course deviation control to ON-COURSE.

bv. POSITION the 479T-2 signal generator WITHIN RANGE of the VHF NAV ANTENNA and:

1. CONNECT the TEST ANTENNA to the NAV OUTPUT CONNECTOR.
2. ENERGIZE the TEST SET.
3. POSITION the METER SWITCH to LINE.
4. ADJUST "SET TO LINE" control to POSITION the needle on the CALIBRATE LINE.
5. SET the NAV GS switch to NAV.
6. SET the MEGAHERTZ DIAL to an OPEN VOR FREQUENCY.
7. SET the MICROVOLTS DIAL to FULL OUTPUT.
8. SET the AUDIO SELECTOR to VOR.
9. SET the VOR RADIAL SELECTOR to 0°.
10. SET the METER SWITCH to CAL.
11. ADJUST CARRIER SET to POSITION meter needle to CALIBRATE POSITION.
12. SET the NO. 1 COURSE INDICATOR course Sel. control to 015°.
13. SET the NO. 1 COMPASS SW to DG, SLEW COMPASS to 0°
14. ENSURE YAW DAMP and AP MAN OFF, autopilot ENGAGED.
15. ENGAGE VOR LOC BUTTON.

bw. ROTATE the NO. 1 course indicator HDG SELECT "BUG" CW and CCW from the aircraft heading. The control wheels SHALL FOLLOW THE "BUG" - RETURN the "BUG" to the AIRCRAFT HEADING.

bx. DISENGAGE the VOR LOC button.

bxA. SLOWLY ROTATE the No. 1 course indicator COURSE SELECT POINTER toward the AIRCRAFT HEADING. When the lateral STEERING BAR is less than one DOT from center, ENGAGE VOR/LOC button, the autopilot shall capture the VOR signal and the control wheels shall ROTATE CW.

by. ROTATE the signal generator COURSE DEVIATION CONTROL to RIGHT and AFT of ON-COURSE. The control wheels shall FOLLOW the signal. Return the COURSE SELECT POINTER TO 0°, ROTATE the HDG "BUG," the aircraft shall not respond.

bz. SET the signal generator GS DEVIATION CONTROL to MAX FLY UP. SET the FREQUENCY CONTROL to the GLIDESCOPE FREQUENCY which corresponds to the LOC FREQUENCY being used.

1. ENGAGE APPR I and ALT on the mode selector.
2. SLOWLY REDUCE the GLIDESCOPE DEVIATION towards ON-COURSE, at approximately on-course signal the mode selector ALT BUTTON shall DISENGAGE.
3. VARY the GLIDESCOPE DEVIATION ABOVE and BELOW the ON-COURSE LEVEL. The control column SHALL FOLLOW.

ca. Doppler (Navigation System Tie-In) (Optional)

ENERGIZE the APPROPRIATE DOPPLER/NAV SYSTEM circuit breakers. On the center console Doppler control, set the function switch to DR. With the SLEW SWITCH set the pilot's DOPPLER INDICATOR DRIFT ANGLE to 0 and GROUND SPEED to 500. On the center console DOPPLER NAV CONTROL set the:

1. MAN AUTO switch to MAN.
2. STAGE A TRACK ANGLE selector to 0000.
3. CROSS TRACK SELECTOR to 0000.
4. OFF-ON switch ON.

(a) DEPRESS STAGE A BUTTON, the annunciator shall indicate ACTIVE.

(b) ALLOW the Doppler system to reach STABLE OPERATING TEMPERATURE.

22.1.2.2 (Cont)

cb. SLEW the NO. 1 compass indicator to 0°. ENGAGE the autopilot - YAW DAMP and AP MAN OFF. ENGAGE NAV mode on FLIGHT MODE SELECTOR.

cc. SET the CROSS TRACK ERROR to 05 miles left. The control wheels shall ROTATE CLOCKWISE. REPEAT for 05 miles right and CCW ROTATION. SET the DOPPLER CONTROL FUNCTION switch to OFF. SET the Doppler NAV CONTROL ON-OFF switch to OFF. DISENGAGE the autopilot.

Note

For a quick self test of the Air Data Computer Type 590D-1, refer to Figure 27-7, page 22-32.

For a complete Dop/Nav ground check refer to Chapter 34.

cd. Altitude Hold Tie-In.

1. Connect the pitot static test set to the static ports. (Refer to Section 34.10.2.6.) Tape opposite side.

CAUTION

If a Pitot/Static crossfeed type test set (VPT10 or equal) is not available, do not exceed 8000 ft. unless the overspeed warning switch is bypassed. Do not exceed 3000 ft. per minute.

2. Evacuate the system to represent 10,000 ft. and 85 KTS airspeed, allow to stabilize.

ce. ENGAGE autopilot (AP MAN and YAW DAMP OFF).

cf. On the flight mode selector, ENGAGE HDG and ALT (Select steering transfer if installed).

1. NOTE, VEE BARS are CENTERED.

2. DECREASE altitude approx. 50 ft. The control column shall MOVE AFT, VEE BARS shall move UP. (Note positions.)

22.1.2.2 (Cont)

3. INCREASE airspeed to 200 KTS, maintain altitude at 9950 ft. Note position of VEE BARS and column, allow system to stabilize. The column and the VEE BARS shall be approximately halfway between center and the position noted in (2).

cfA. VERTICAL SPEED TEST. (Remove the hardware attaching the acceleration sensor.)

1. Adjust PITOT/STATIC for an ascent rate of 1000 ft/minute and ENGAGE VS MODE. INCREASE the rate of ascent and note the CONTROL COLUMN moves FORWARD.
2. Reduce the rate of ascent to 1000 ft/minute. Controls shall return to NEUTRAL.
3. Rotate the ACCELERATION SENSOR upside down, note CONTROL COLUMNS move FORWARD.
4. Restore the ACCELERATION SENSOR to normal. CONTROL COLUMNS return to NEUTRAL.

cfB. IAS TEST.

1. Adjust PITOT/STATIC for an IAS of 150 KTS.
2. ENGAGE IAS mode.
3. INCREASE airspeed to 160 KTS. The CONTROL COLUMNS shall move AFT.
4. REDUCE airspeed to 140 KTS, CONTROL COLUMNS shall move FORWARD.
5. DISENGAGE IAS mode.

cfC. MACH TEST.

1. Adjust the PITOT/STATIC test set for 0.6 MACH.
2. ENGAGE MACH mode.
3. INCREASE pitot pressure to MACH 0.7. The CONTROL COLUMNS shall move AFT.
4. DECREASE pitot pressure to MACH 0.5. The CONTROL COLUMNS shall move FORWARD.
5. DISENGAGE MACH mode.
6. Return PITOT/STATIC system to ambient - slowly.
7. DISENGAGE the autopilot.

cg. Autopilot Servo - Overpower Test (With No. 1 System Boost Pressure On).

Using a spring scale attached to the pilot's or copilot's control wheel, measure the force required to move the controls through approx. 2.0 inches of travel and record.

1. ROLL FORCE.
2. PITCH UP FORCE.
3. PITCH DOWN FORCE.
4. YAW FORCE.

ch. ENGAGE the autopilot in AP MAN mode.

cj. Measure force required to move the controls thru approx. 2 inches of travel. The forces shall exceed the recorded forces within the range listed below:

1. ROLL - 16 to 19 inch pound of torque
2. PITCH-UP - 26 to 40 pound
3. PITCH-DOWN - 21 to 36 pound
4. YAW - 40 to 50 pound

ck. Repeat item cj. three times to determine the average value within limits specified.

cl. OVERPOWER the ROLL CHANNEL in RIGHT ROLL. The aileron trim indicator will DEFLECT CCW.

cm. OVERPOWER the RIGHT RUDDER PEDAL. The rudder trim indicator will DEFLECT LEFT.

cn. OVERPOWER the control column PITCH DOWN and HOLD. The elevator trim indicator will DEFLECT UP. The horizontal stabilizer WILL DRIVE NOSE DOWN. The elevator trim annunciator will ILLUMINATE AFTER 5 SEC. RELEASE the control column and the annunciator WILL EXTINGUISH.

co. OVERPOWER the column PITCH DOWN and operate the trim switches NOSE DOWN. Hold the control column PITCH UP and operate the wheel trim switches NOSE UP. The elevator trim fail annunciator WILL LIGHT.

cp. TURN OFF the systems and RESTORE the airplane to NORMAL. Remove the two (2) jumpers across the aileron and elevator HI/PASS jacks.

22.1.2.3 TROUBLESHOOTING. Refer to Section 22 as a guide. The AP-104 is so integrated with the FD-109 it is not practical to detail specific problems and solutions. The technician is advised to consult the preflight and flight test procedures in the AP-104/FD-109H/FD-109F flight control system maintenance manual, supplied by Collins Radio Co.

22.1.2.4 REMOVALS AND INSTALLATIONS.

22.1.2.4.1 REMOVAL OF AUTOPILOT SERVOS. The rudder, elevator and aileron autopilot servos are located under the cockpit floor between fuselage sta. 199.0 and 213.0. It is necessary to remove the copilot's seat and floor panel to gain access to the rudder and elevator servos. Access to the aileron servo is obtained by removing the pilot's seat and floor

22.1.2.4 (Cont).

panel. The servomotor is removed by disconnecting the electrical connectors and removing the six captive screws around the periphery of the case on, the opposite side to the cable drum. The servomotor may then be withdrawn from the servo mount without disconnecting the control cables. Therefore re-rigging of the control cables is not required. When removing the aileron and rudder servos it may be necessary to remove the servomount attaching bolts and rotate the servo to gain access to the six captive screws. The aileron servo mount is attached to the airplane structure by an adapter plate assembly. The adapter plate is secured to the servomount by two bolts and washers and to the aircraft structure by two bolts and washers. Two additional bolts, nuts and washers attach the mount to the aircraft structure.

22.1.2.4.2 INSTALLATION OF AUTOPILOT SERVOS. Reverse the removal procedure. Inspect cables for proper installation, ensure cables are properly seated in the grooves of the drum. See Chapter 27 for installation and rigging information.

CAUTION

Check the servomotor and/or servo mount identification plate(s) to ensure re-installation of the proper unit in its respective location as follows:

AIL. - 90 in. lbs. torque

ELEV. - 70 in. lbs. torque

RUD. - 125 in. lbs. torque

Use extreme care when sliding the servomotor into the servo mount to avoid damaging the gears.

22.1.2.5 PRIMARY SERVO, SERVO-TORQUE SETTING. Use mating connector MS3106A-10SL-3S (Collins part number 357-4086-00) for application of DC power. Wire A and B mating connector to the DC power source.

a. Apply 27.5 volts DC between pins P2A and P2B. This engages the electromagnetic clutch in the unit.

b. Use a 5/16-inch socket adapter with torque wrench (Snap-On Tools Corp., Kenosha, Wisconsin, 300-0-300 inch-pounds or equivalent).

c. Lock gear train of the servo by inserting the shaft of a Phillips screwdriver through the motor plate hold as shown in figure 22-4.

d. Drive the gears with a torque wrench in either counterclockwise or clockwise rotation. Note indication on the torque wrench dial, and check the appropriate service information letter for proper values.

e. Adjust mechanical clutch tension with spanner wrench, as shown in figure 22-5. Turn the clutch adjusting nut away from the front panel (where the connector is located) to release the tension on the mechanical clutch.

f. Mark the torque setting on the nameplates of the Primary Servo Mount and the Primary Servo.

22.1.2.6 TRIM COUPLER GAIN SETTING. The following paragraphs describe the setup procedure for the Trim Coupler.

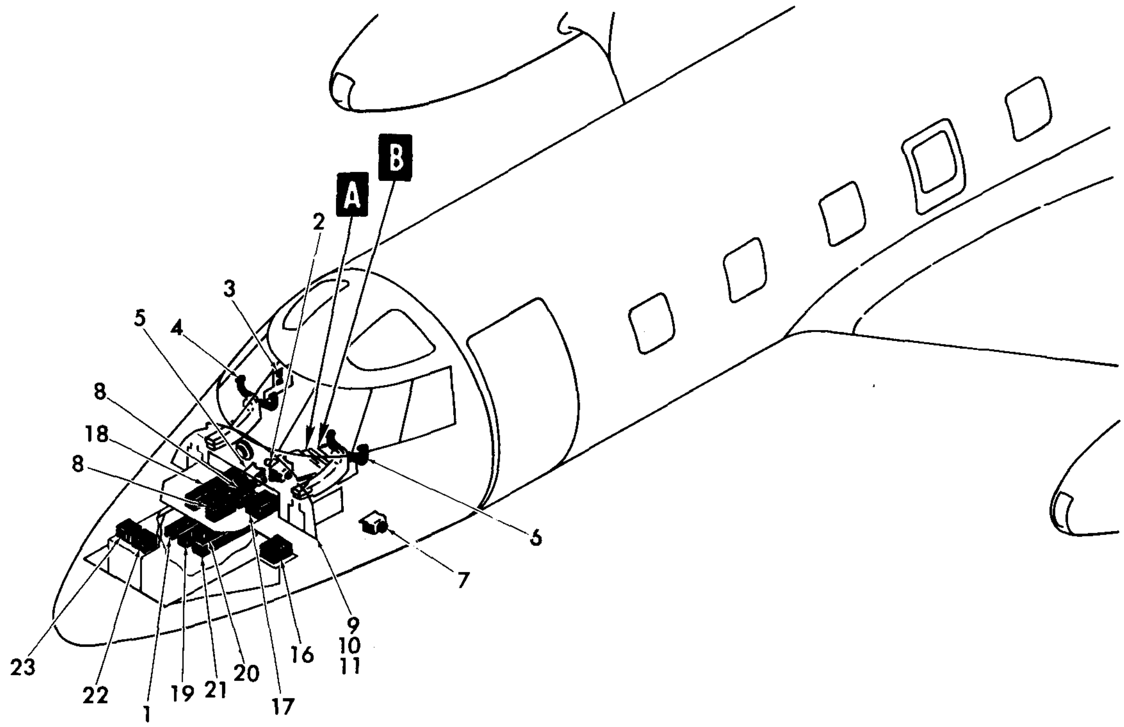
a. Short the aileron and elevator high-pass filters at the front panel jacks on the 562C-4A Autopilot Amplifier. Engage the autopilot and overpower the elevator until the elevator servo clutch slips. The aircraft trim should operate at maximum rate. Degrees per minute is determined by observing aircraft trim indication in conjunction with a stopwatch. Adjust R47 for maximum trim speed as specified by the aircraft type SERVICE INFORMATION LETTER. (Ensure that there is no voltage on P1-9.)

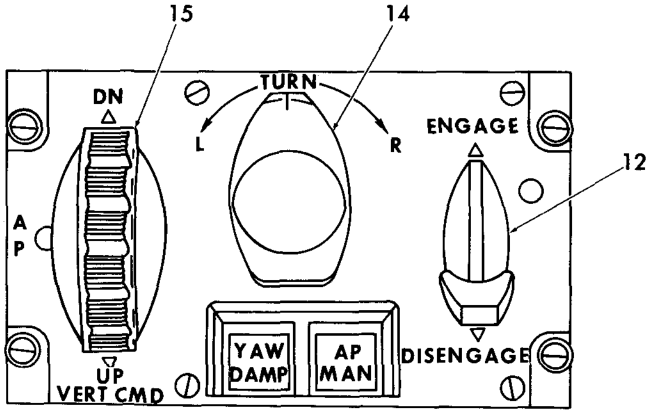
b. To adjust trim-up and trim-down sensitivities, engage the autopilot and overpower the elevator in the pitch-down direction. Observe the voltage present between J2 and J3 with a DC voltmeter when the elevator servo clutch slips. Record this voltage. Remove the trim coupler from the aircraft to complete the adjustment.

c. Connect 115 volts AC 400 Hz to P1-8, 28 volts DC to P1-20, and ground to P1-1. Connect a 28-volt DC light bulb between J10 and J11, and another between J1 and J7. Apply a DC voltage equal to that obtained in step (2) to P1-11 and P1-13, (+ on P1-11). Adjust R2 to the minimum clockwise setting necessary to give 10 pulses on the light connected to J7 for 10 pulses on the light connected to J10.

d. Disconnect the lead at J7 and reconnect the lamp to J8. Apply the DC voltage to P1-22 and P1-24 (+ on P1-22). Adjust R22 as in the preceding step (3).

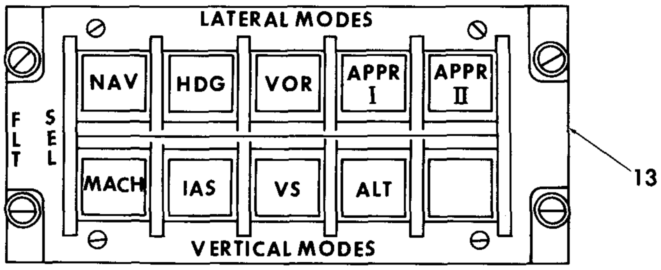
e. The trim coupler may now be reinstalled in the aircraft. After adjusting trim-up and trim-down sensitivities, recheck the R47 adjustment.





AUTOPILOT CONTROLLER

A



FLIGHT MODE SELECTOR

B

1. AUTOPILOT AMPLIFIER
2. RUDDER AUTOPILOT SERVOMOTOR
3. AUTOPILOT TRANSFER RELAY
4. COPILOT'S AUTOPILOT DISCONNECT PUSHBUTTON
5. ELEVATOR AUTOPILOT SERVOMOTOR
6. PILOT'S AUTOPILOT DISCONNECT PUSHBUTTON
7. AILERON AUTOPILOT SERVOMOTOR
8. VERTICAL GYRO
9. RUDDER TRIM METER
10. AILERON TRIM METER
11. ELEVATOR TRIM METER
12. AFCS SWITCH
13. MODE SELECTOR
14. TURN CONTROL
15. PITCH CONTROL
16. AIR DATA COMPUTER
17. RATE SENSING UNIT
18. FLIGHT DIRECTOR COMPUTER
19. AIR DATA COUPLER
20. TRIM COUPLER
21. MODE COUPLER
22. ACCELERATION SENSOR
23. YAW RATE GYRO

Figure 22-3. Autopilot System Components (COLLINS)

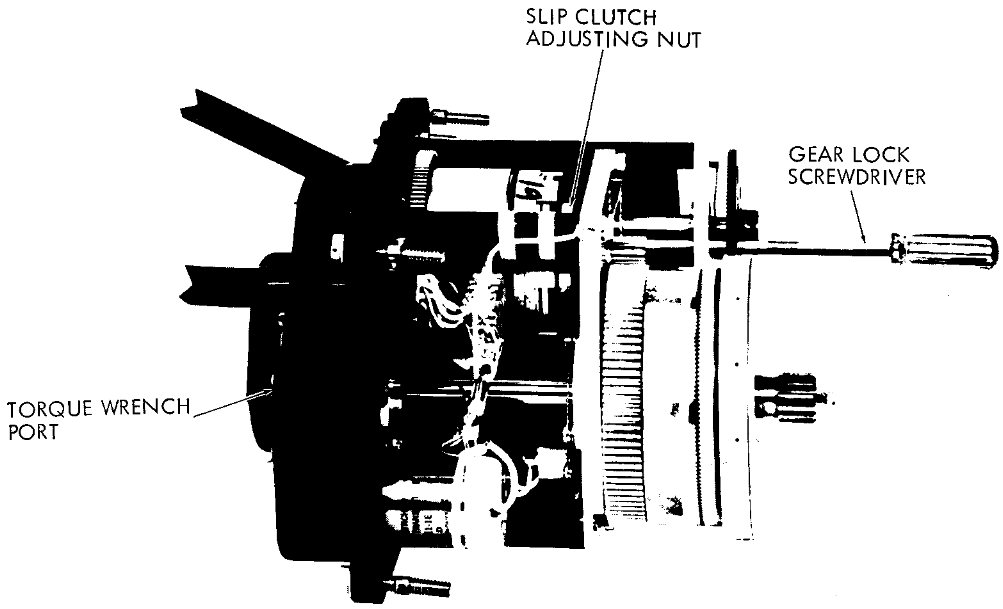
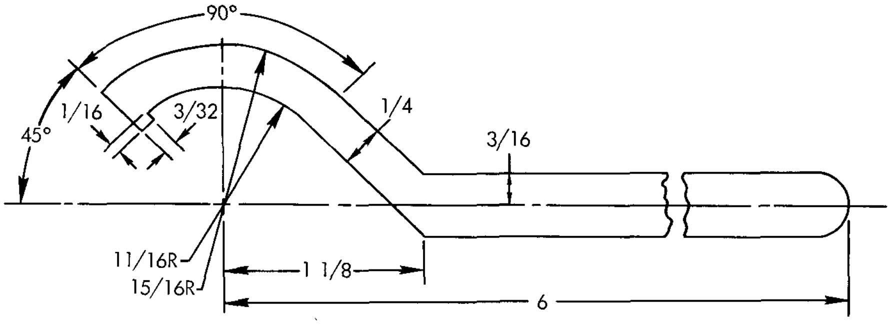


Figure 22-4. Primary Servo, Torque Setting Location Diagram



NOTE

1. MATERIAL: 1/8 FLAT STOCK STEEL
2. ALL DIMENSIONS IN INCHES. FRACTIONAL TOLERANCE ± 1/64.
3. REMOVE BURRS AND SHARP EDGES.

Figure 22-5. Spanner Wrench Fabrication Diagram

POSITION 1	ALTITUDE (AT 29.92) MACH SAT TAS ALL LAMPS WILL BE OFF	FIELD ELEVATION NOT CORRECTED FOR BAROMETER READING OF THE DAY .05 ± .02 AMBIENT ± 3° 42 ± 5 KNOTS
POSITION 2	ALTITUDE MACH SAT TAS LAMPS -- SELF-TEST (RED) - ON	AS IN POSITION 1 ENCODER READOUT LAMPS TO DISPLAY ALTITUDE OF POSITION 1 .05 ± .025 -40° ± 3°C 42 ± 5 KNOTS
POSITION 3	ALTITUDE MACH SAT TAS LAMPS -- SELF-TEST (RED) - ON ALTITUDE (AMBER) WILL COME ON UNTIL ALTITUDE SERVO NULLS.	ALTIMETER READING OF POSITION 1 +750 ± 100 FEET .05 ± .025 -40° ± 3°C 42 ± 5 KNOTS
POSITION 4	ALTITUDE MACH SAT TAS LAMPS -- SELF-TEST (RED) - ON AIRSPEED (AMBER) WILL COME ON UNTIL AIRSPEED SERVO NULLS.	AS IN POSITION 3 .05 ± .025 -40° ± 3°C 70 ± 5 KNOTS
POSITION 5	ALTITUDE MACH SAT TAS LAMPS -- SELF-TEST (RED) - ON MACH (AMBER) WILL COME ON UNTIL MACH SERVO NULLS.	AS IN POSITION 3 .88 ± .02 -71° ± 3°C 490 ± 5 KNOTS

Figure 22-5A. Air Data Computer Self Test (590D-1) (Sheet 1 of 3)

READ-OUT LAMPS ---- X DENOTES LIGHT ON

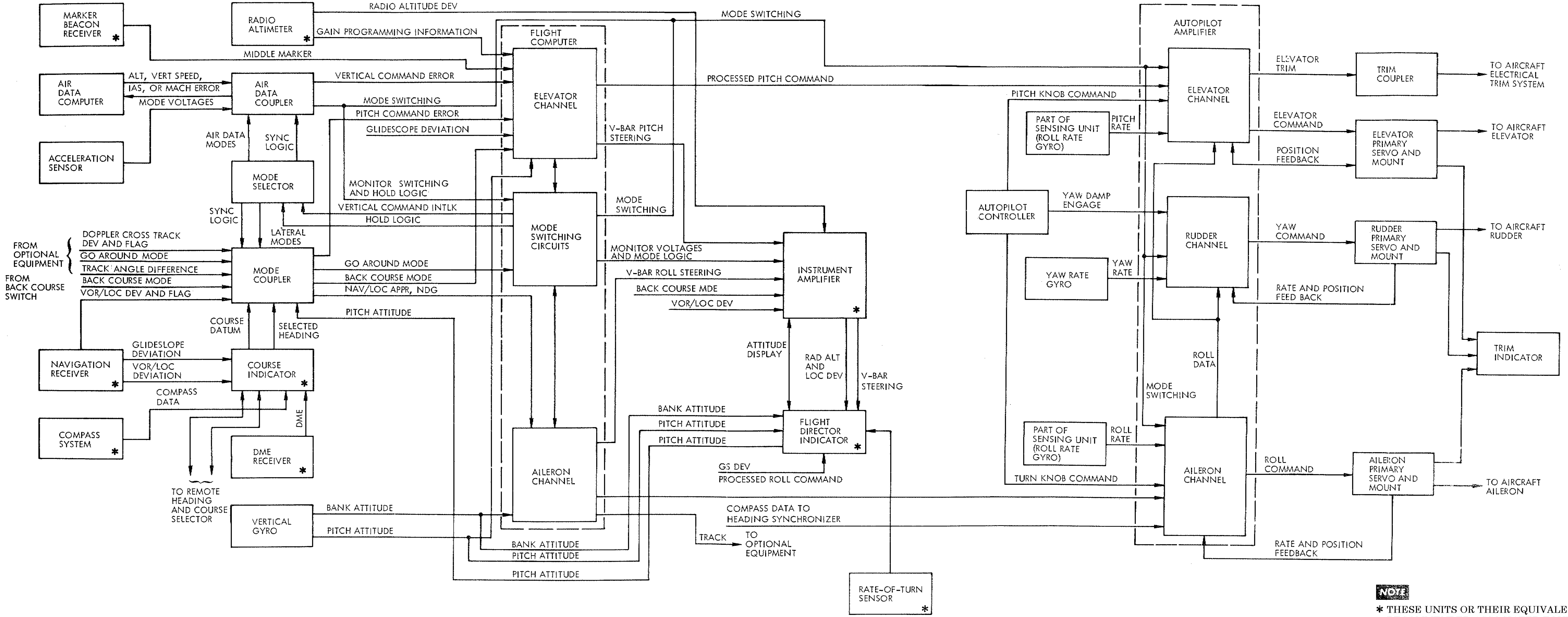
ALTITUDE	D4	A1	A2	A4	B1	B2	B4	C1	C2	C4
-500							X		X	
-400							X		X	X
-300							X			X
-200						X	X			X
-100						X	X		X	X
0						X	X		X	
100						X	X	X	X	
200						X	X	X		
300						X		X		
400						X		X	X	
500						X			X	
600						X			X	X
700						X				X
800					X	X				X
900					X	X			X	X
1000					X	X			X	
1100					X	X		X	X	
1200					X	X		X		
1300					X	X	X	X		
1400					X	X	X	X	X	
1500					X	X	X		X	
1600					X	X	X		X	X
1700					X	X	X			X
1800					X		X			X
1900					X		X		X	X
2000					X		X		X	
2100					X		X	X	X	
2200					X		X	X		
2300					X			X		
2400					X			X	X	
2500					X				X	
2600					X				X	X
2700					X					X
2800				X	X					X
2900				X	X				X	X
3000				X	X				X	
3100				X	X			X	X	
3200				X	X			X		
3300				X	X		X	X		
3400				X	X		X	X	X	
3500				X	X		X		X	
3600				X	X		X		X	X
3700				X	X		X			X
3800				X	X	X	X			X
3900				X	X	X	X		X	X
4000				X	X	X	X		X	
4100				X	X	X	X	X	X	

Figure 22-5A. Air Data Computer Self Test (590D-1) (Sheet 2 of 3)

READ-OUT LAMPS -- "X" DENOTES LIGHT ON

ALTITUDE	D4	A1	A2	A4	B1	B2	B4	C1	C2	C4
4200				X	X	X	X	X		
4300				X	X	X		X		
4400				X	X	X		X	X	
4500				X	X	X			X	
4600				X	X	X			X	X
4700				X	X	X				X
4800				X		X				X
4900				X		X			X	X
5000				X		X			X	
10000			X	X	X		X		X	
15000		X	X						X	
20000		X	X	X	X	X	X		X	
25000		X		X	X	X			X	
30000		X					X		X	
35000	X	X		X	X				X	
40000	X	X	X	X		X	X		X	
45000	X	X	X			X			X	
50000	X		X		X		X		X	

Figure 22-5A. Air Data Computer Self Test (590D-1) (Sheet 3 of 3)



NOTE

* THESE UNITS OR THEIR EQUIVALENTS ARE RECOMMENDED ASSOCIATED EQUIPMENTS, NOT SUPPLIED WITH THIS INSTALLATION.

Figure 22-6. Collins Autopilot System Block Diagram

CHAPTER 23
COMMUNICATIONS

TABLE OF CONTENTS

Section Page

23.0 GENERAL 23-3

23.1 INTERCOMMUNICATION AND PASSENGER ADDRESS SYSTEM 23-5

23.2 VHF COMMUNICATION SYSTEMS 23-19

23.3 HF COMMUNICATION SYSTEMS 23-27

SECTION 23.0

GENERAL

Contents Page

DESCRIPTION 23-3
General Information 23-3
Nose Electronic Equipment Rack 23-3
Cooling Provisions 23-3

REMOVAL AND INSTALLATION 23-3
Removal of the Nose Section 23-3
Removal of Electronic Equipment Boxes 23-3
Installation of Electronic Equipment Boxes 23-3

MINOR REPAIR AND PRESERVATION 23-3

23.0.1 DESCRIPTION.

Note

On airplanes 5046 and up the communication system components are not installed by LOCKHEED when the airplane is manufactured. Complete space provisions for the components and wiring provisions are installed.

23.0.1.1 GENERAL INFORMATION. Two separate VHF communication systems, routed through two AUDIO SELECTOR panels, furnish two-way radio contact with either ground stations or other airplanes. An interphone system and a passenger address system provide communications within the airplane. Interphone facilities are in the cabin and at the pilot's and copilot's stations. The audio heard at either the pilot's or copilot's interphone station can also be heard through an overhead speaker in the pilots' compartment. If the throttles are retarded below minimum cruise position while the landing gear is retracted on airplanes 5001 thru 5065, audio signals to the cockpit speaker are replaced with an audible warning signal. Radio reception during this period of warning may be maintained by using headset earphones. For ground maintenance, interphone connections can be made at two external service interphone jacks. Announcements over passenger address speakers in the cabin area can be made from the pilot's and copilot's microphones. If desired, AM broadcast radio programs can be heard over the passenger address system from one of the two automatic direction finder system receivers.

23.0.1.2 NOSE ELECTRONIC EQUIPMENT RACK. See navigation systems, general, paragraph 34.0.2.3.

23.0.1.3 COOLING PROVISIONS. See navigation systems, general, paragraph 34.0.2.5.

23.0.2 REMOVAL AND INSTALLATION.

23.0.2.1 REMOVAL OF THE NOSE SECTION. See navigation systems, general, paragraph 34.0.2.1.

23.0.2.2 REMOVAL OF ELECTRONIC EQUIPMENT BOXES. See navigation systems, general, paragraph 34.0.2.2 and 34.0.2.3.

23.0.2.3 INSTALLATION OF ELECTRONIC EQUIPMENT BOXES. See navigation systems, general, paragraph 34.0.2.2 and 34.0.2.3.

23.0.3 MINOR REPAIR AND PRESERVATION. See navigation systems, general paragraph 34.0.3.

SECTION 23.1
INTERCOMMUNICATION AND PASSENGER ADDRESS SYSTEM

Contents Page

DESCRIPTION AND OPERATION 23-5
General 23-5
List of Components 23-6
Audio Selector Panels 23-7
PA-Interphone Control Panel 23-7
Interphone Amplifier 23-7
Passenger Address Amplifier 23-7
Overhead Audio Control Panel 23-7
Microphone Pre-amplifier 23-8
Pilots' Compartment Speaker 23-8
Passengers' Compartment Speakers 23-8
Headsets, Microphones, and Handset 23-8
Control Wheel Microphone Switches 23-8
Service Interphone Jacks 23-8
Audible Warning Unit 23-8

OPERATIONAL CHECKOUT 23-8
Intercommunication and Passenger Address Systems 23-8

TROUBLESHOOTING 23-10
Intercommunication and Passenger Address Systems 23-10

REMOVAL AND REPLACEMENT 23-13
Intercommunication and Passenger Address Systems Components 23-13

FIGURES
23-1. Intercommunication and Passenger Address Systems Components Locations 23-14
23-2. Intercommunication and Passenger Address Systems Block Diagram 23-17

23.1.1 DESCRIPTION AND OPERATION.

Note

On airplanes 5046 and up, the intercommunication and passenger address system components are not installed by Lockheed when the airplane is manufactured. Complete space provisions for the components and wiring provisions are installed. See paragraph 23.1.1.2 for components required to make system operative.

23.1.1.1 GENERAL. The intercommunication and passenger address systems (figures 23-1 and 23-2) provide interior communication between the pilot, copilot, and passengers. Provisions are made for the pilot and copilot to monitor all radio communication receivers as well as the VOR-1, VOR-2, DME, ADF, and marker beacon navigation receivers. Provisions are also made for the pilot and copilot to transmit over the VHF communication systems. The intercommunication system is connected to the passengers' PA-interphone control panel providing ADF reception, or communication to the pilot or copilot, as selected on the overhead audio control panel. An overhead speaker, with a built-in amplifier, is provided in the pilots' compartment, and either the pilots' or copilot's selected audio can be broadcast over it. If throttles are retarded below minimum cruise position while the landing gear is retracted on airplanes 5001 thru 5065, audio signals to the cockpit speaker are replaced with an audible warning signal. Radio reception during this period of warning may be maintained by using the headset earphones. Two speakers in the passengers' compartment provide ADF audio, pilot or copilot audio, to be broadcast over them. Two service interphone jacks are provided for communication between the interior and the exterior of the airplane. The system is supplied 28-volt DC power from the essential DC bus through

23.1.1.1 (Cont).

the PA and INPH circuit breakers on the pilot's side circuit breaker panel, and a LG WARNING circuit breaker on the pilot's wainscot circuit breaker panel. The system also receives 28-volt DC power from the main DC bus through the COPILOT ISOLATION AMP circuit breaker on the copilot's wainscot circuit breaker panel.

23.1.1.1 (Cont).

23.1.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Audio Selector Panel (red lighted)	2	Gables	--	G-909-1 [1]
Audio Selector Panel (white lighted)	2	Gables	--	or G1732
PA-Interphone Control Panel	1	Lockheed	--	JR447 [1]
Interphone Amplifier	1	Gables	--	G-912
PA Amplifier	1	Gables	--	G-947 [1]
Microphone Preamplifier	2	Gables	--	G-956
Audible Warning Unit	1	Seaboard Electric	--	1520-2 [7]
Cabin Interphone Switch	1	Microswitch	--	2TP25-21
Cabin Speaker ADF Switch	1	Microswitch	--	1TP 25-1
Cockpit Speaker Switch	1	Microswitch	--	1TP 25-1
Pilots' Compartment Speaker	1	Gables	--	G-515E-11 [5] G-515E-111 [4] G-1565 [6]
Passengers' Cabin Speaker	2	University	--	8501 [1]
Handheld Micorphone	2	Telephonic	RS38E-5	107D000-5
Mask Microphone	2	Roanwell	--	M32B/AIC
Headset	2	Telephonic	--	20002-8
Control Wheel Microphone Switch	2	Licon	--	76-4100 [2]
Control Wheel Microphone Switch	1	Haydon	--	53430-5 [3]
Control Wheel Microphone Switch	1	Haydon	--	53430-6 [3]
Passenger Handset	1	Electro-Voice	--	625TRSKK3226 [1]
Cabin Interphone Light	1	--	--	MS25041-8

[1] On aircraft 5046 and up this equipment is required for completion of system.

[2] Aircraft 5046 and up.

[3] Aircraft 5001 thru 5045 only.

[4] Aircraft 5036 thru 5055.

[5] Aircraft 5001 thru 5035.

[6] Aircraft 5056 and up and preferred spare.

[7] Aircraft 5001 thru 5065.

23.1.1.3 AUDIO SELECTOR PANELS. Two Gables G-909-1 or G-1732 audio selector panels, one on the pilot's side console and one on the copilot's side console, are provided on the airplane. The panel has a VOL knob, an EMER switch, 13 monitor switches, and four microphone selector pushbutton switches. From his particular panel, the pilot or the copilot can select the systems to be heard over his own headset. With the two-position (forward, aft) monitor switch for a given system in the forward position, the receiver audio output of that system enters the audio selector panel, passes through an isolation resistor and through the VOL control rheostat. Next the audio signal is routed through an isolation amplifier within the panel and through the normal position contacts of the EMER switch. The amplified signal is then heard over the corresponding headset, and if desired, over the pilots' compartment speaker, by way of the overhead audio control panel. If the EMER switch is placed in the "EMER" position, the resistors and the isolation amplifier within the panel are bypassed, and the audio signal is routed directly to the headset. In this case, a signal priority sequence is established and the audio output of only one of the selected systems can be heard. The order of preference (beginning with VHF1) is from left to right, row by row from the top (through MKR). To permit routing of transmitted audio as well as received audio through an audio selector panel, the appropriate microphone selector pushbutton must be depressed. When a microphone pushbutton selector switch is engaged, a light within the switch will go on. The light will remain on until released, but only one pushbutton per panel should be depressed at a time. For example, with a VHF microphone selector pushbutton depressed, the received audio of the VHF receiver is routed through the panel, and the transmitter stays on standby. When either the corresponding press-to-talk microphone switch is depressed or the corresponding control wheel interphone switch is held in the "TRANSMIT" position, the transmitter is fully operational and the receiver remains on standby. With an INPH microphone selector pushbutton depressed, the interphone audio output is routed to the audio selector panel concerned. However, if either the related microphone push-to-talk button is pressed or the related control wheel interphone switch is placed in the "INTERCOM" position, audio from the appropriate microphone will be routed to the interphone amplifier.

23.1.1.4 PA-INTERPHONE CONTROL PANEL.

Note

> On aircraft 5001 thru 5045 this is a customer furnished item. The system described is a typical Lockheed recommended system, but not necessarily the system the customer has installed. On aircraft 5046 and up wiring provisions for the particular system are installed.

A Lockheed JR-322 Passenger Address-Interphone control panel is mounted in the passengers' compartment. The panel has a preset VOLUME control and AUDIO SEL switch for the passenger handset and a VOLUME knob and AUDIO SEL switch for the two cabin speakers. The two-position ("OFF," "RADIO,") switch is set to the "RADIO" position to distribute ADF audio over the passenger compartment speakers; the cockpit overhead panel CABIN SPKR ADF switch must also be set to either "ADF1" or "ADF2". The three position ("OFF," "INPH," "CALL") AUDIO SEL switch is used for communication to the pilot and copilot over the passenger handset. For communication between the pilots and the passengers, the AUDIO SEL switch is positioned to "INPH" and the CABIN INPH switch on the cockpit overhead audio control panel is positioned to "ON." The "CALL" position of the AUDIO SEL switch is spring-loaded to the "INPH" position. When held in the "CALL" position, a CALL INPH LIGHT on the copilot's side of the main instrument panel will go on. This announces to the pilots that intercommunications are requested between the pilots and passengers.

23.1.1.5 INTERPHONE AMPLIFIER. A Gables G-912 interphone amplifier is used for amplification of all intercommunication signals. The amplifier is supplied 28-volt DC power from the essential DC bus through the INPH circuit breaker on the pilot's side circuit breaker panel. The amplifier's output is rated at 0.8 watts. Two screwdriver-type volume control adjustments are provided: one for the interphone volume control, and one for the service volume control.

23.1.1.6 PASSENGER ADDRESS AMPLIFIER. A Gables G-947 passenger address (PA) amplifier is used for amplification of the ADF audio and the pilots' announcements. When selected on the PA-interphone control panel, ADF audio can be amplified and routed to the cabin speakers. But if the pilot or copilot depresses the PA microphone selector pushbutton switch on his audio selector panel, then presses his microphone press-to-talk button or holds his control wheel microphone switch in the "TRANSMIT" position, only his announcement will be amplified. Inside the amplifier, a relay is energized which disconnects the signals from the ADF radio and connects the pilot's and copilot's microphones. The amplifier output is routed to the two cabin speakers, and a sidetone signal is directed to the pilot's and copilot's audio selector panels. When the weight of the airplane rests on the landing gear, a squat switch is closed. This energizes a relay within the passenger address amplifier, thereby providing a different and adjustable degree of amplification for ground operation. The amplifier is supplied 28-volt DC power from the essential DC bus through the PA circuit breaker on the pilot's side circuit breaker panel.

23.1.1.7 OVERHEAD AUDIO CONTROL PANEL. The overhead audio control panel provides control for the pilots' compartment speaker and the routing of ADF audio. The two-position ("ON," "OFF") CABIN INPH switch connects the passenger handset to the interphone amplifier through the PA-interphone control panel. The three-position "ADF 1," "OFF" ADF 2

23.1.1.7 (Con'').

CABIN SPKR switch connects the ADF audio to the speakers in the passengers' compartment through the PA-interphone control panel. The three-position ("PILOT," "COPILOT," "OFF") COCKPIT SPKR switch connects the pilot's or copilot's selected audio to the speaker in the pilots' compartment.

23.1.1.8 MICROPHONE PRE-AMPLIFIER. A Gables G956 microphone pre-amplifier is provided between each pilot's M-32B/AIC mask microphone and each audio selector panel. The pre-amplifier is used for increasing the input to the interphone amplifier, communication transmitters, or PA amplifier, when transmitting over the mask microphone.

23.1.1.9 PILOTS' COMPARTMENT SPEAKER. A Gables G-515E-11 (aircraft 5001 thru 5035), G-515E-111 (aircraft 5036 thru 5055) or Gables G-1565 (aircraft 5056 and up) speaker is installed on the overhead control panel in the pilots' compartment. The speaker incorporates an amplifier for amplfication of the audio signal and the landing gear warning tone. A continuous turn potentiometer is installed in the audio input line to the amplifier. The potentiometer is used to vary the audio output level of the speaker, and is adjustable through a hole in the panel with a small screwdriver. The pilot's or copilot's selected audio can be broadcast over the speaker by positioning the COCKPIT SPKR switch on the overhead autio control panel to "PILOT" or "CO-PILOT" and positioning the desired monitor switch on the corresponding audio selector panel forward (on).

23.1.1.10 PASSENGERS' COMPARTMENT SPEAKERS. Two University Loudspeakers 8501 speakers, one in the forward section of the cabin and one in the aft section of the cabin, are used with the passenger address system. When selected, ADF audio or pilot announcements are routed through the passenger address amplifier to the speakers.

23.1.1.11 HEADSETS, MICROPHONES, AND HANDSET. The pilot and copilot each have a headset, a handheld microphone, and a mask microphone. When not in use, the pilot's equipment is normally kept on the forward end of the pilot's wainscot, and the copilot's equipment on the copilot's wainscot. The headset receives the audio selected on the corresponding audio selector panel. With the desired microphone selector pushbutton switch on the audio selector panel depressed, transmission by the corresponding handheld microphone is possible when the press-to-talk button is held in. Also, transmission by the corresponding MASK microphone is possible when the related control wheel microphone switch is held in the "INTERCOM" or "TRANSMIT" position. A handset is provided at the PA-interphone control panel for intercommunication with the pilot and copilot. The handset has a press-to-talk feature for transmitting. Intercommunication receptions can be heard on the handset when selected on the PA-interphone control panel, and by the pilots on the overhead audio control panel.

23.1.1.12 CONTROL WHEEL MICROPHONE SWITCHES. A three-position ("INTERCOM," "OFF," "TRANSMIT") microphone switch is installed on the control wheels. Both the "INTERCOM" and "TRANSMIT" position are spring-loaded to the center-off position. Holding the switch in the "INTERCOM" position allows the pilot to transmit through the mask microphone over the intercommunication system. Holding the switch in the "TRANSMIT" position allows the pilot to transmit through the mask microphone over the system selected on the audio selector panel.

23.1.1.13 SERVICE INTERPHONE JACKS. Two service interphone jacks, one in the nose wheel well and one in the aft equipment compartment, are provided for intercommunication connection points during ground servicing and maintenance. Dust caps are installed over the jacks when not in use.

23.1.1.14 AUDIBLE WARNING UNIT (Aircraft 5001 thru 5065). If an unsafe landing gear condition occurs, a landing gear warning relay is energized. This connects an audible warning unit to the pilots' compartment speaker, which contains its own amplifier, and disconnects any interphone signals which might have been coming over the speaker. Simultaneously, power is supplied to the audible warning unit which generates an interrupted warning tone heard over the pilots' compartment speaker. Since all interphone system signals over the pilots' compartment speaker are disconnected while the warning is sounding, one of the pilots should wear his headset at all times.

23.1.2 OPERATIONAL CHECKOUT.

23.1.2.1 INTERCOMMUNICATION AND PASSENGER ADDRESS SYSTEMS. The operational checkout of the intercommunication and passenger address systems will require two men. Man A will be stationed in the pilots' compartment and man B will move to different stations as directed in the following procedures:

a. Connect a source of 28-volt DC external power to the airplane and energize the AC and DC buses.

CAUTION

Do not operate electronic systems any longer than necessary on the ground, especially when a number of systems are in use at one time. Even then it is preferable to slide the nose radome forward to permit the heat to dissipate more easily. Otherwise, damage due to overheating the equipment boxes can result.

b. Make sure the INPH, PA, and the COPILOT ISOLATION AMP circuit breakers are closed.

c. Man A: On both the pilot's and copilot's audio selector panels, make sure the microphone selector pushbutton switches are out (off), the

23.1.2.1 (Cont)

monitor switches are in the aft (off) position, the EMER switch is in the aft (normal) position, and the VOL knob is approximately mid-way.

d. Man A: On the overhead audio control panel, make sure the CABIN SPKR ADF switch, the COCKPIT SPKR switch, and the CABIN INPH switch are all in the OFF position.

e. Man A: Place the INT monitor switch on the pilot's audio selector panel in the forward (on) position; then place the pilot's control wheel microphone switch in the "INTERCOM" position.
RESULT: When the control wheel microphone switch is operated, a click should be heard over the pilot's headset; but there should be no extraneous, objectionable sounds.

f. Man A: Repeat step (e) except place the copilot's control wheel microphone switch in the "INTERCOM" position.

g. Man A: Repeat steps (e) and (f) reversing the pilot's and copilot's stations.

h. Man A: With the switches positioned according to step (c), connect the pilot's mask microphone.

i. Man A: Press in the INT microphone selector pushbutton switch on the copilot's audio selector panel.

j. Man A: Hold the pilot's control wheel microphone switch in the "INTERCOM" position.

k. Man A: Speak into the pilot's mask microphone at a moderate level of speech.
RESULT: The signal, as heard on the copilot's headset, should be loud and clear with no objectionable distortion.

l. Man A: Press in the PA microphone selector pushbutton switch on the copilot's audio selector panel, and speak into the pilot's mask microphone.
RESULT: No signals should be heard over the copilot's headset.

m. Man A: Repeat steps (h) through (l) reversing the pilot's and copilot's stations.

n. Man B: With the switches positioned according to step (c), connect a microphone-headset to the service interphone jack in the nose wheel well.

o. Man A: Hold the pilot's control wheel microphone switch in the "INTERCOM" position, and speak into the pilot's mask microphone at a moderate level of speech.
RESULT: The speaker's voice should be heard at the nose wheel well station.

p. Man A: Place the INT monitor switch on the pilot's audio selector panel in the forward (on) position.

q. Man B: Speak into the service microphone at the nose wheel well station.
RESULT: The speaker's voice should be heard over the pilot's headset.

r. Men A and B: Repeat steps (n) through (q) for the aft compartment service interphone jack.

s. Man A: Place the VHF 1 monitor switch on the pilot's audio selector panel in the forward (on) position, and select an operating station on the VHF communications system.
RESULT: Audio of the VHF station selected should be heard over the pilot's headset.

t. Man A: Place the VHF 1 monitor switch in the aft (off) position.

u. Man A: Repeat steps (s) and (t) for the rest of the monitor switches on the pilot's audio selector panel in place of the VHF 1 monitor switch.

v. Man A: On the pilot's audio selector panel, place the EMER switch in the "EMER" position. Then, beginning with the MKR monitor switch, one by one place each monitor switch in the forward (on) position. Proceed from right to left, row by row from the bottom of the panel.
RESULT: The audio for the system with priority, or the latest monitor switch turned on, should be heard over the pilot's headset.

w. Man A: Repeat steps (s) through (v) for the copilot's station.

x. Man A: Turn off all the electronic systems except the VHF 1 communication system.

y. Man A: Place the VHF 1 monitor switch on the copilot's audio selector panel in the forward (on) position, and the COCKPIT SPKR switch on the overhead audio control panel in the "COPILOT" position.
RESULT: Audio from the VHF 1 Communications receiver should be heard over the copilot's headset and the pilots' compartment speaker.

Note

If the pilots' compartment speaker volume is not satisfactory, the screwdriver adjustment at the speaker can be set to the desired level.

z. Man A: Select a vacant channel on the VHF 1 communications system, press in the VHF 1 microphone selector pushbutton switch on the pilot's audio selector panel, and speak into the pilot's hand-held microphone.
RESULT: The volume of the pilots' compartment

23.1.2.1 (Cont).

speaker should decrease, and that of the copilot's headset remain unchanged. Sidetone should be heard over the pilot's headset.

aa. Man A: Place the VHF-1 monitor switch on the copilot's audio selector panel in the aft (off) position, and turn off the VHF-1 communications system.

ab. Man B: On the passengers' PA-interphone control panel, hold the handset AUDIO SEL switch in the "CALL" position and then release.
RESULT: The CABIN INPH LIGHT on the main instrument panel should go on, and go out when the switch is released.

ac. Man A: Press in the INT microphone selector pushbutton switch on the pilot's audio selector panel.

ad. Man B: Speak into the passenger handset.
RESULT: No audio should be heard over the pilot's headset, nor should any sidetone be heard over the handset.

ae. Man A: Place the CABIN INPH switch on the overhead audio control panel in the "ON" position.

af. Man B: Speak into the passenger handset.
RESULT: The speaker's voice should be heard over the pilot's headset, and sidetone should be heard over the handset.

ag. Man A: Speak into the pilot's hand-held microphone.
RESULT: The speaker's voice should be heard over the passenger handset, and sidetone should be heard over the pilot's headset.

ah. Man A: Tune in a station on the ADF 1 system, and place the CABIN SPKR ADF switch on the overhead audio control panel in the ADF 1 position.

ai. Man B: Place the cabin speakers AUDIO SEL switch in the "RADIO" position.
RESULT: Audio from the ADF-1 system should be heard over the cabin speakers.

aj. Man B: Operate the appropriate landing gear squat switch.
RESULT: There should be a noticeable change in the volume of the cabin speakers.

ak. Man A: Turn off the ADF 1 system. Tune in a station on the ADF 2 system and place the CABIN SPKR ADF switch on the overhead audio control panel in the ADF 2 position.
RESULT: Audio from the ADF 2 system should be heard over the cabin speakers.

al. Man A: Turn off the ADF 2 system.

am. Man A: Place the CABIN SPKR ADF switch on the overhead audio control panel in the "OFF" position.

an. Man A: Press in the PA microphone selector pushbutton switch on the pilot's audio selector panel, and speak into the pilot's hand-held microphone.
RESULT: The speaker's voice should be heard over the cabin speakers.

ao. Man A: Hold the pilot's control wheel microphone switch in the "TRANSMIT" position, and speak into the pilot's mask microphone.
RESULT: The speaker's voice should be heard over the cabin speakers.

ap. Man A: Release the pilot's control wheel microphone switch, and speak into the pilot's mask microphone.
RESULT: The speaker's voice should not be heard over the cabin speakers.

aq. Man A: Repeat steps (an) through (ap) for the copilot's station.

ar. Man A: Turn off all switches and systems in the pilots' compartment.

as. Disconnect external power by reversing the procedure in Chapter 22.

23.1.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. There are no manuals available for any of the system components. Troubles most often found in the intercommunication and passenger address systems can be traced to switches, amplifiers, selector panels, or cabling between these units. Common troubles, such as motorboating, oscillations, distortion, cross-talk, and weak signals at a particular station can usually be traced to loose cabling, defective selector panel, or defective microphone or headset. Before beginning troubleshooting, test each intercommunication station and note any abnormal symptoms. Assure that the switches for the circuit involved are properly positioned.

PROBABLE CAUSE	ISOLATION PROCEDURE
ENTIRE INTERCOMMUNICATION AND PASSENGER ADDRESS SYSTEMS ARE INOPERATIVE.	
INPH and PA circuit breakers open.	Check circuit breakers.
Defective wiring.	Check wiring.

23.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
INTERCOMMUNICATION SYSTEM INOPERATIVE, BUT PA SYSTEM OPERATIVE.	
INPH circuit breakers open.	Check circuit breakers.
Defective interphone amplifier.	Substitute operable amplifier.
Defective wiring.	Check power wiring to amplifier.
PILOT'S AUDIO SELECTOR PANEL IS INOPERATIVE, BUT COPILOT'S AUDIO SELECTOR PANEL OPERATES NORMALLY.	
Pilot's INPH circuit breaker open.	Check circuit breaker.
Defective audio selector panel isolation amplifier.	Set NORM-EMERG switch to EMERG. If system then operates, the isolation amp is defective.
Defective audio selector panel.	If system does not operate on EMERG, substitute operable control panel.
Defective wiring.	Check wiring to control panel.
COPILOT'S AUDIO SELECTOR PANEL IS INOPERATIVE, BUT PILOT'S AUDIO SELECTOR PANEL OPERATES NORMALLY.	
COPILOT ISOLATION AMP circuit breaker open.	Check circuit breaker.
Defective audio selector panel isolation amplifier.	Set NORM-EMERG switch to EMERG. If system then operates the isolation amp is defective.
Defective audio selector panel.	If system does not operate on EMERG, substitute operable control panel.
Defective wiring.	Check wiring to control panel.
TRANSMISSIONS CANNOT BE MADE THROUGH THE MASK MICROPHONE, BUT CAN BE MADE THROUGH HANDHELD MICROPHONE.	
Defective mask microphone.	Substitute operable microphone.
Defective control wheel microphone switch.	Substitute operable switch.
Defective mask microphone pre-amplifier.	Substitute operable pre-amplifier.
Defective audio selector panel.	Substitute operable panel.
Defective wiring to microphone.	Check wiring.
TRANSMISSIONS CANNOT BE MADE THROUGH HANDHELD MICROPHONE, BUT CAN BE MADE THROUGH MASK MICROPHONE.	
Defective handheld microphone.	Substitute operable microphone.
Defective audio selector panel.	Substitute operable panel.
Defective wiring to microphone.	Check wiring.

23.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
NO AUDIO IS AVAILABLE OVER PILOTS' COMPARTMENT SPEAKER, BUT IS AVAILABLE THROUGH HEADSETS.	
LG WARNING circuit breaker open.	Check circuit breaker.
Defective COCKPIT SPKR switch.	Substitute operable switch.
Defective pilots' compartment speaker.	Substitute operable speaker.
Defective gear warning relay.	Check relay for continuity between contacts D2 and D3.
Defective wiring.	Check wiring.
NO TRANSMISSIONS OR RECEPTIONS CAN BE MADE USING THE PASSENGER HANDSET.	
Defective handset.	Substitute operable handset.
Defective PA-interphone control panel.	Substitute operable panel.
NO AUDIO IS AVAILABLE OVER THE PASSENGERS' COMPARTMENT SPEAKERS.	
PA circuit breaker open.	Check circuit breaker.
Defective fuse in PA amplifier.	Check fuse.
Defective PA-interphone control panel.	Substitute operable panel.
Defective PA amplifier.	Substitute operable amplifier.
Defective speakers.	Substitute operable speaker.
Defective wiring.	Check wiring.
NO INTERCOMMUNICATIONS AVAILABLE AT THE PASSENGER HANDSET.	
Defective passenger handset.	Substitute operable handset.
Defective CABIN INPH switch.	Substitute operable switch.
Defective PA-interphone control panel.	Substitute operable panel.
Defective interphone amplifier.	Substitute operable amplifier.
Defective wiring.	Check wiring.
NO ADF AUDIO AVAILABLE OVER THE PASSENGERS' COMPARTMENT SPEAKERS.	
Defective CABIN SPKR ADF switch.	Substitute operable switch.
Defective PA-interphone control panel.	Substitute operable panel.
NO ADF AUDIO AVAILABLE OVER THE PASSENGERS' COMPARTMENT SPEAKERS (Cont).	
Defective PA-amplifier.	Substitute operable amplifier.
Defective wiring.	Check wiring.

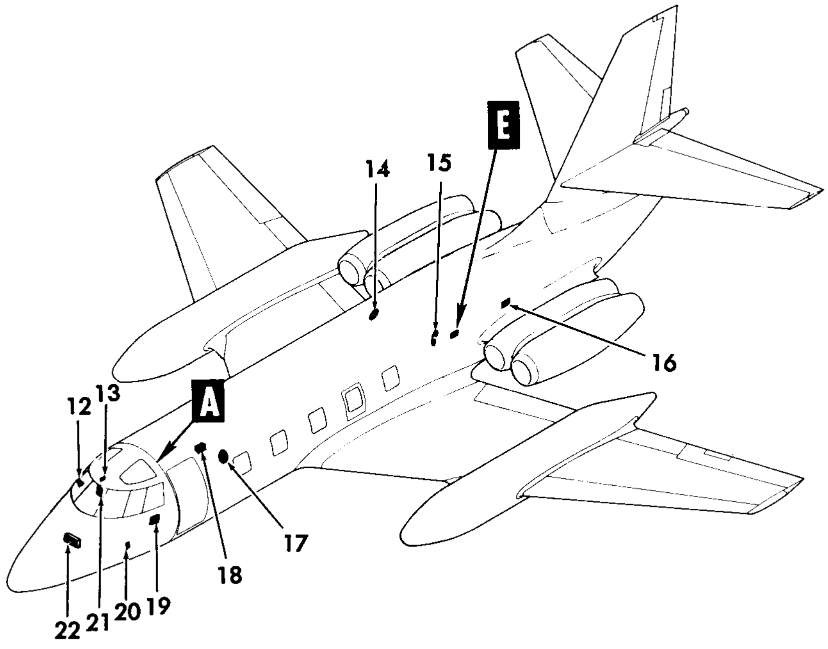
23.1.4 REMOVAL AND REPLACEMENT.

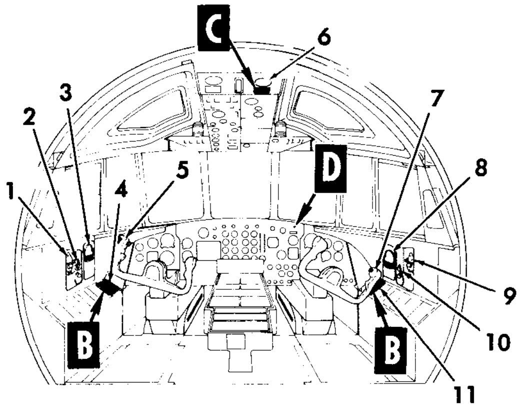
23.1.4.1 INTERCOMMUNICATION AND PASSENGER ADDRESS SYSTEMS COMPONENTS. Removal and replacement of the intercommunication and passenger address systems components require no special maintenance techniques, with the exception of the following instructions:

a. Before the removal of system components, disconnect electrical power and disengage circuit breakers as required.

b. After the replacement of system components, connect electrical power and engage circuit breakers.

c. Perform an operational checkout of the system.



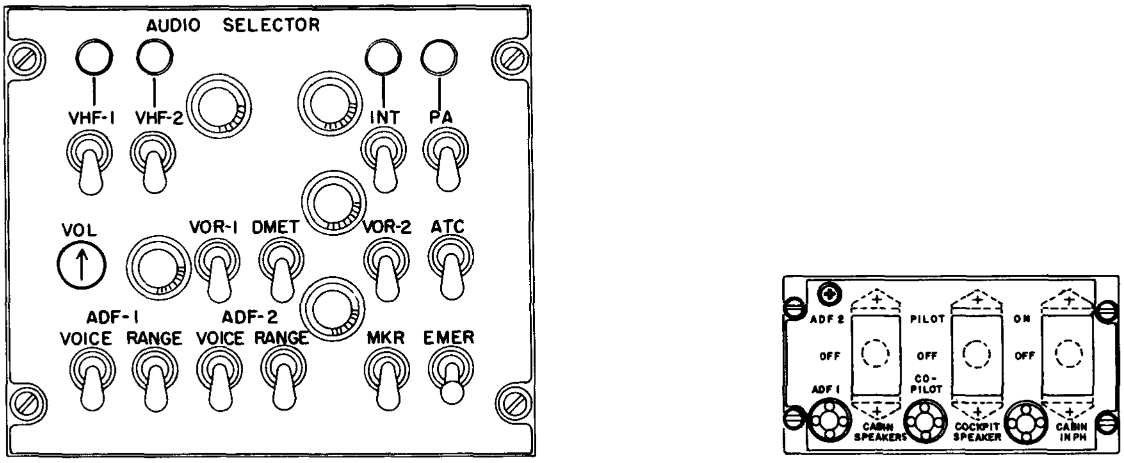


(LOOKING FORWARD)

A

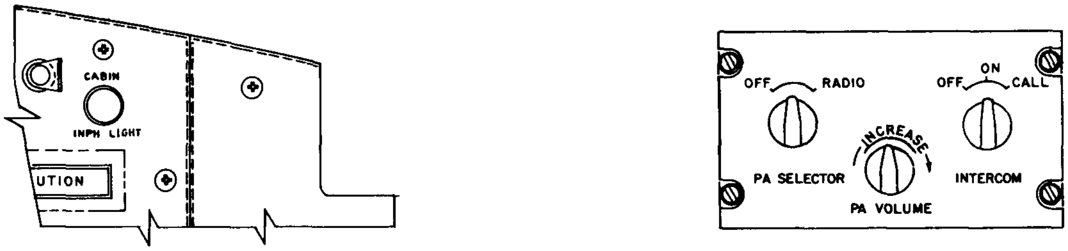
1. PILOT'S MASK MICROPHONE
2. PILOT'S HAND-HELD MICROPHONE
3. PILOT'S HEADSET
4. PILOT'S AUDIO SELECTOR PANEL
5. PILOT'S CONTROL WHEEL INTERPHONE SWITCH
6. PILOT'S COMPARTMENT SPEAKER
7. COPILOT'S CONTROL WHEEL INTERPHONE SWITCH
8. COPILOT'S HEADSET
9. COPILOT'S MASK MICROPHONE
10. COPILOT'S HAND-HELD MICROPHONE
11. COPILOT'S AUDIO SELECTOR PANEL
12. COPILOT'S MASK MICROPHONE PRE-AMP
13. LANDING GEAR WARNING RELAY
14. AFT CABIN SPEAKER
15. CABIN HEADSET
16. AFT COMPARTMENT SERVICE INTERPHONE JACK
17. FORWARD CABIN SPEAKER
18. INTERPHONE AMPLIFIER
19. PILOT'S MASK MICROPHONE PRE-AMP
20. NOSE WHEEL WELL SERVICE INTERPHONE JACK
21. AUDIBLE WARNING UNIT (AIRPLANES 5001 THRU 5065)
22. PASSENGER ADDRESS AMPLIFIER

Figure 23-1. Intercommunication and Passenger Address Systems Components Locations (Sheet 1 of 2)



AUDIO SELECTOR PANEL (TYPICAL)
(LOOKING OUTBOARD)
B

OVERHEAD AUDIO CONTROL PANEL
(LOOKING UP)
C

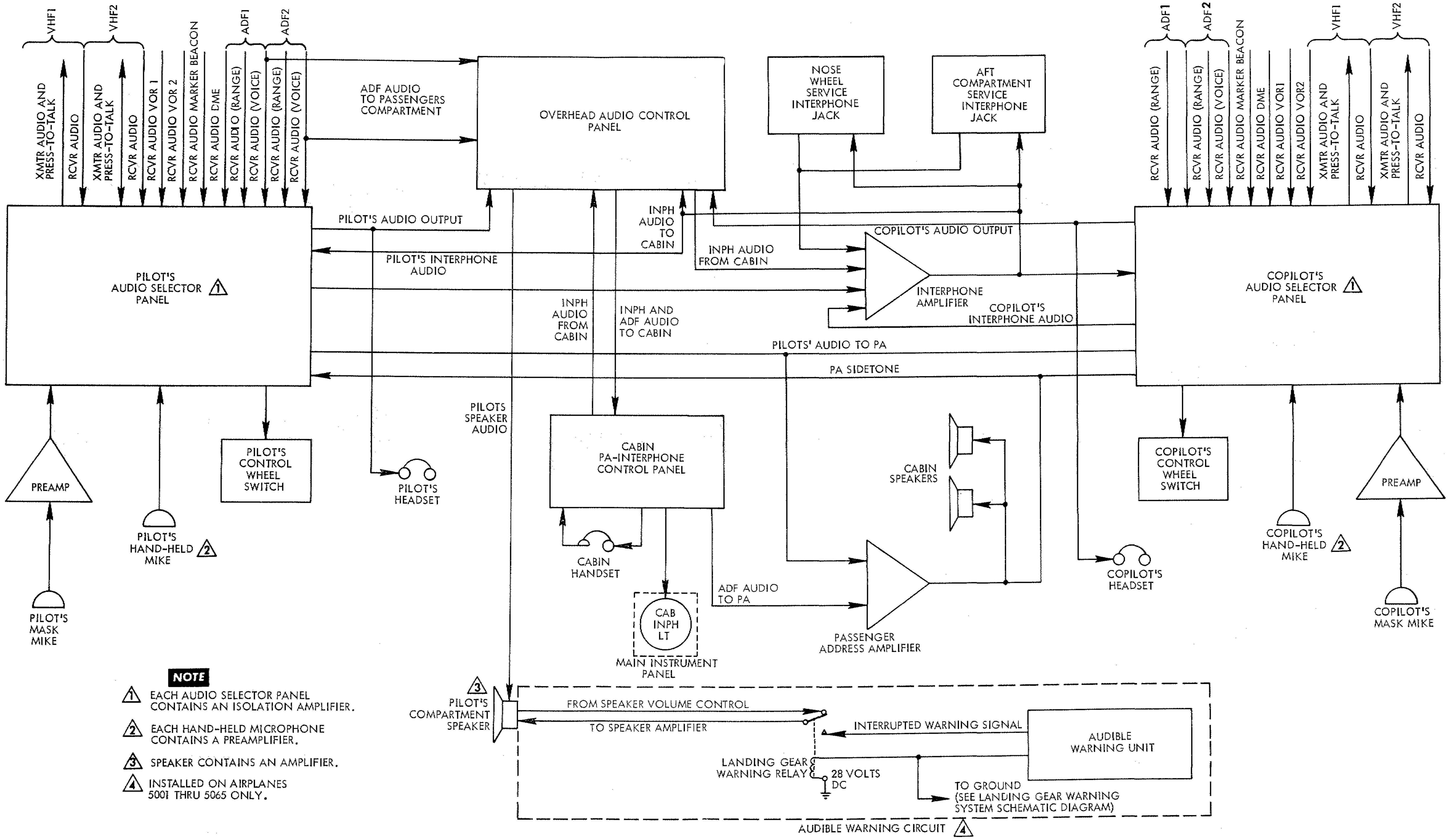


(LOOKING FORWARD)
D

CABIN PA-INTERPHONE CONTROL PANEL
(LOOKING OUTBOARD TO THE LEFT)
E

JS-2-2-2304-2

Figure 23-1. Intercommunication and Passenger Address Systems Components Locations (Sheet 2 of 2)



NOTE

1. EACH AUDIO SELECTOR PANEL CONTAINS AN ISOLATION AMPLIFIER.
2. EACH HAND-HELD MICROPHONE CONTAINS A PREAMPLIFIER.
3. SPEAKER CONTAINS AN AMPLIFIER.
4. INSTALLED ON AIRPLANES 5001 THRU 5065 ONLY.

Figure 23-2. Intercommunication and Passenger Address Systems Block Diagram

Changed 1 December 1965

SECTION 23.2
VHF COMMUNICATION SYSTEMS

Contents Page

DESCRIPTION AND OPERATION 23-19
General 23-19
List of Components 23-19
Modes of Operation 23-20
Control Panels 23-20
Transmitters 23-20
Receivers 23-20
Antennas 23-21

OPERATIONAL CHECKOUT 23-21

TROUBLESHOOTING 23-22

REMOVAL AND REPLACEMENT 23-23
Removal of Antennas 23-23
Replacement of Antennas 23-23

FIGURES
23-3. VHF Communication Systems Components 23-24
23-4. VHF Communication Systems Block Diagram 23-26

23.2.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060, and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

23.2.1.1 GENERAL. Two independent VHF communication systems provide two-way plane-to-ground or plane-to-plane communications in the frequency range of 118.00 to 135.95 megacycles in increments of 0.05 megacycles. Control of the two VHF systems is by two panels on the center console: a VORTAC-VHF panel for VHF No. 1 system and a VOR-VHF panel for VHF No. 2 system. The VHF No. 1 system is supplied 28-volt DC from the essential DC bus through a VHF No. 1 REC and a VHF No. 1 XMTR circuit breaker on the pilot's side circuit breaker panel; the No. 2 VHF system receives main DC bus power through a VHF No. 2 REC and VHF No. 2 XMTR circuit breaker on the copilot's wainscot circuit breaker panel. Each of the two VHF systems contains a Collins 51X-2 receiver, a Collins 17L-7 transmitter, a control panel, a power relay, and a Collins 37R2 antenna. All transmission and reception is routed through one of the AUDIO SELECTOR panels.

23.2.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
VORTAC/VHF Control Panel	1	Gables	--	G-906V [1]
VORTAC/VHF Control Panel	1	Gables	--	G-906VA [3] [2]
VOR/VHF Control Panel	1	Gables	--	G-905V [1]

NOTE: [1] Aircraft 5001 thru 5038 only.
[2] On airplanes 5046 thru 5049, 5052, 5056 and 5059 this equipment is required for completion of system.
[3] Aircraft 5039 thru 5049, 5052, 5056 and 5059.

23.2.1.2 LIST OF COMPONENTS (Cont).

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
VOR/VHF Control Panel	1	Gables	--	G-905VA △3 △2
Transmitter	2	Collins	17L-7	5220899024 △2
Receiver	2	Collins	51X-2	5220833054 △2
Antenna	2	Collins	37R-2	5221135012
Power Relay	2	--	--	AN3311-2
Antenna	2	Dorne and Margolin	37R-2	DMC50-2 △4

NOTE: △1 Aircraft 5001 thru 5038 only.
△2 On aircraft 5046 and up this equipment is required for completion of system.
△3 Aircraft 5039 and up.
△4 Aircraft 5121 and up and preferred spare.

23.2.1.3 MODES OF OPERATION. Two modes of operation can be used with the VHF system. The modes are: the single channel simplex (SCS) and double channel simplex (DCS). In SCS operation, which is the more common, the transmitter and receiver are tuned to the same frequency, and the receiver is disabled during operation of the transmitter. Operation is thereby restricted to either transmission or reception on the assigned channel. The DCS operation differs from the SCS operation in that the transmitter is tuned to a frequency 6 megacycles above the frequency of the receiver. The DCS operation is permitted in the transmitter frequency range of 124.00 to 126.95 and 133.00 to 135.95 megacycles. In DCS operation the receiver is also disabled during operation of the transmitter, and the frequency indicated on the control panel is always that of the receiver.

23.2.1.4 CONTROL PANELS. The VORTAC-VHF panel on the pilot's side of the center console controls the VHF No. 1 system; the VOR-VHF panel on the copilot's side controls the VHF No. 2 system. Each panel has a VHF communications frequency selector, VHF VOL control, squelch control and mode selector switch. When a VHF system is turned on with its volume control knob, a corresponding VHF communications power relay is energized. This relay then connects the 28-volt DC input to both the receiver and the transmitter. Turning the VHF VOL control knob clockwise increases the volume of the receiver audio output. The two-position ("DC," "SC") mode selector switch selects either DCS or SCS operation. On airplanes 5001 thru 5038: the two-position ("SQ," "OFF") squelch switch, when placed in the "SQ" position, eliminates noise below a preset signal level. On airplanes 5039 and up; A squelch control potentiometer enables the squelch to be set at the control panel. The two frequency selector knobs are mounted concentrically with the on-off/VOL knob. The knob closer to the panel selects the frequency in megacycle increments (the numbers to the left of the decimal point), and the other knob selects the frequency in 50-kilocycle increments (the numbers to the right of the decimal point).

23.2.1.5 TRANSMITTERS. Two 25-watt VHF transmitters are installed in the nose electronic equipment compartment. Each transmitter is built in modular form and contains the following module assemblies: R-F Module, Modulator Module and Power Supply Module. The R-F Module contains the components of the electromechanical auto-positioner system which enables the transmitter to be tuned remotely from the control panel. The transmitter contains 38 crystals, which control the usable 560 channels. A test meter and associated test switch on the front panel of the transmitter provide a monitoring point for certain important stages of the equipment. The nine positions of the test switch are: "OFF," "ANT," "PA-G," "DR-G," "MOD-E," "BAT," "LV," "RV," and "PA-P." The test meter is a 0- to 1-milliampere ammeter, which is wired to show the following normal indications:

TEST METER Switch Position	Typical TEST METER Indications with 25 Watts Output
ANT	0.4 - 0.6
PA-G	0.4 - 1.0
DR-G	0.3 - 1.0
MOD-E	0 - 0.8*
BAT	0.7
LV	0.7
HV	0.7
PA-P	0.7

*This indication depends on the percentage of modulation.

Frequency indicator dials on the transmitter front panel provide an indication of the frequency to which the transmitter is tuned.

23.2.1.6 RECEIVERS. Two VHF receivers are mounted adjacent to the transmitter units, and are installed in the nose electronic equipment compartment. Like the transmitter, the receivers are built in modular form and contain an electro-mechanical autopositioner system. Each receiver contains the

23.2.1.6 (Cont).

following module assemblies: R-F and Variable I-F Amplifier Module, Power Supply Module, and I-F and Audio Module. The receiver's autopositioning system is installed on the R-F and Variable I-F amplifier module and is controlled from the control panel. There

23.2.1.6 (Cont).

are 560 channels available for selection by the control unit. The front panel consists of a frequency indicator, a "PHONES" jack, a SQUELCH adjustment screw, and a pushbutton switch. The frequency selected by the control panel appears in the window of the frequency indicator. The "PHONES" jack permits the output of the receiver to be monitored. The SQUELCH adjustment screw is used to set the signal level (threshold) below which the squelch is effective. The SQUELCH pushbutton switch will override the squelch circuit, when depressed and held.

23.2.1.7 ANTENNAS. Each of the two VHF communications systems has its own vertically-polarized, blade-type antenna used for both transmitting and receiving. The VHF 1 antenna is mounted on the bottom centerline of the fuselage aft of the nose wheel. The VHF 2 antenna is mounted on the top centerline of the fuselage aft of the windshield.

23.2.1.7.1 On aircraft 5121 and up, the fiberglas VHF antennas have been replaced with all metal antennas that give improved rain erosion performance, carry lightning strikes safely to ground, and have a significantly reduced level of P static interference.

23.2.2 OPERATIONAL CHECKOUT. The operational checkout of the VHF communication systems will require two men. Man A will be stationed in the pilot's seat and man B will be stationed at the transmitter and receiver. When intercommunications or VHF transmissions are being made, the appropriate microphone selector pushbutton switch on the pilot's audio selector panel must be depressed.

Note

Reception and transmission on the VHF systems are dependent on the intercommunication system parts. Therefore, perform this check during or after the operational checkout of the intercommunication system.

a. Man A: Turn off all switches in the pilot's compartment.

b. Man A and B: Connect a source of 28-volt, DC external power to the airplane and energize the AC and DC buses.

c. Man A: Make sure the VHF 1 REC, VHF 1 XMTR and INPH circuit breakers on the pilot's side circuit breaker panel are closed.

d. Man B: Connect the service interphone cord to the nose wheel well interphone receptacle and connect a suitable headset-microphone to the cord. Extend the nose section as described in Section 34.0.2.1.

e. Man A: Place the VHF 1 switch on the pilot's audio selector panel to "VHF 1" and adjust the VOL knob as desired.

f. Man A: Turn the VHF-1 system on by turning the on-off/VOL knob on the VHF 1 control panel clockwise. Allow a 3-minute warmup.

On aircraft 5001 thru 5038, proceed as outlined in steps (g), (h), and (i):

g. Man A: Place the SQ/OFF switch on the VHF 1 control panel to the "SQ" position.

h. Man A: Select several frequencies that are known to be active and are within the reception capabilities of the receiver.

i. Man B: Set the SQUELCH adjust control on the front panel of the receiver so it just opens the receiver on the weakest of the signals found in step (h).
RESULT: The audio output of the receiver should be sufficient for good reception.

On aircraft 5039 thru 5049, 5052, 5056 and 5059, proceed as outlined in steps (j), (k), and (l):

j. Man B: Set the SQUELCH adjust control on the front panel of the receiver fully counterclockwise.

k. Man A: Select several frequencies that are known to be active and are within the reception capabilities of the receiver.

l. Man A: Check the operation of the SQUELCH knob on the VHF-1 control panel.
RESULT: The audio output of the receiver should be sufficient for good reception.

m. Man A: Select a strong, active station within the reception capabilities of the receiver.
RESULT: The audio output level should be loud and clear without objectionable distortion, and the background noise should be greatly reduced. This indicates that the automatic gain control is functioning properly.

n. Man A and B: Rotate the large knob on the control panel through all the megacycle channels, and the small knob through all the 50-kilocycle channels. Place the DC/SC switch in the "SC" position. Observe the frequency windows on the control panel and on the front panel of the receiver and transmitter.
RESULT: The frequency indications on the control panel should agree with those on the front panel of the receiver and transmitter.

o. Man A: Place the DC/SC switch on the control panel to the "DC" position. Set the frequency indicator to 118.00 megacycles.

p. Man B: Observe the frequency window on the front panel of the receiver and transmitter.
RESULT: The frequency of the receiver should read 118.00 megacycles and frequency of the transmitter should read 124.00 megacycles.

q. Man A: Place the DC/SC switch in the "SC" position.

r. Man A: Select a frequency that may be used for testing transmissions and contacting the local control tower.
RESULT: Speech (sidetone) should be heard in the pilot's headset during transmission. Reception should be clear.

23.2.2 (Cont).

s. Man B: While man A is transmitting (on a test frequency), turn the meter selector switch to each of its positions while observing the test meter.
RESULT: As the switch is turned to its various positions, the meter should indicate the readings shown in paragraph 23.2.1.5.

t. Man A: Turn the on-off/VOL knob fully counterclockwise.

u. Repeat steps (c) through (t) for the copilot's VHF 2 system.

v. Turn off all switches in the pilot's compartment.

w. Man B: Disconnect and stow the service interphone cord and the head-microphone. Secure the nose section by reversing the procedure in Chapter 34.

x. Man B: Disconnect external power.

23.2.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component refer to the following handbooks: transmitter manuals, Collins part numbers 5230021006, 5230021004, and 5230021003; receiver manuals, Collins part numbers 5230022006, 5230022004, and 5230022003. There are no manuals available for the Gables control panels; these should be returned to the manufacturer for repair.

PROBABLE CAUSE	ISOLATION PROCEDURE
SYSTEM COMPLETELY INOPERATIVE.	
VHF XTMR and REC circuit breakers open.	Check circuit breakers.
VHF power relay inoperative.	Using a 0-50 volt DC voltmeter, measure the voltage between terminal A1 and ground and B1 and ground with the on-off/VOL switch fully counterclockwise and then in the clockwise position. The measured voltages should be 0 volts in the fully counterclockwise position and approximately 28 volts in the clockwise position.
Defective control panel.	Substitute operable control panel.
Defective wiring.	Check wiring.
SYSTEM OPERATING, SIDETONE NORMAL, BUT NO TRANSMISSION OR RECEPTION.	
Defective antenna.	Substitute known good antenna.
Defective co-axial cable.	Check cable for damage or loose connectors.
TRANSMISSION NORMAL, BUT NO RECEPTION OR SIDETONE.	
VHF REC circuit breaker open.	Check circuit breaker.
Fault in the intercommunication system.	Check reception at the receiver using the PHONES jack on the front panel.
Defective control panel.	Substitute operable control panel.
Defective receiver.	Substitute operable receiver.
Defective wiring.	Check wiring.
AUDIO OUTPUT INADEQUATE.	
SQUELCH control on the receiver set incorrectly.	Check setting of SQUELCH control.
Fault in the intercommunication system.	Check reception at the receiver using the PHONES jack on the front panel.

23.2.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
AUDIO OUTPUT INADEQUATE (Cont).	
Aural level adjust (RIB) in the receiver set too low.	Remove the receiver from its mount and advance the aural level control. Replace the receiver in its mount.
Defective receiver.	Substitute operable receiver.
TRANSMITTER TEST METER READINGS DO NOT FALL WITHIN THE ALLOWABLE TOLERANCES.	
Defective transmitter.	Substitute operable transmitter.
RECEPTION NORMAL, BUT NO TRANSMISSIONS CAN BE MADE.	
VHF XMTR circuit breaker open.	Check circuit breaker.
Fault in the intercommunication system.	Check transmission at the transmitter using the MIKE jack on the front panel.
Defective transmitter.	Substitute operable transmitter.
Defective control panel.	Substitute operable control panel.
Defective wiring.	Check wiring.

23.2.4 REMOVAL AND REPLACEMENT.

23.2.4 REMOVAL OF ANTENNAS. The antennas are installed in pressurized sections of the airplane and have been sealed to the fuselage skin to prevent the loss of cabin air pressure. When an antenna is replaced these seals must be renewed. Use the procedure in paragraph 23.2.4.2 to install and seal a new antenna. Use the following procedure to remove an antenna:

a. Upper antenna: Remove the headliner material overhead below the antenna at the entrance to the pilots compartment.

b. Lower antenna: Remove the floorboard aft of the inverter compartment.

c. Disconnect the coaxial cable connector at the base of the antenna.

d. From the outside of the airplane, remove the six screws which hold the antenna to the fuselage.

e. Remove the antenna from the fuselage.

23.2.4.2 REPLACEMENT OF ANTENNAS. The contacting surface of the new antenna must be bonded to the fuselage skin. Before mounting the antenna, clean a small area around each screw hole on both the antenna and the fuselage. Use a fine flat file or a rotary stainless-steel brush. After filing, wipe the area with a grease-free cloth moistened in Specification TT-N-95 aliphatic naphtha. Remove all grease, oil, fingerprints, or other foreign substances. When both surfaces are properly cleaned, use the following procedure to install and seal the antenna.

a. Place a new rubber seal and the antenna in the proper mounting position on the fuselage with the tip of the antenna pointing aft.

b. Secure the antenna to the fuselage with six mounting screws.

c. Apply a coating of Minnesota Mining and Mfg. Co. EC-1239, Class A4 or equal, sealing compound around the opening where the antenna receptacle enters the fuselage.

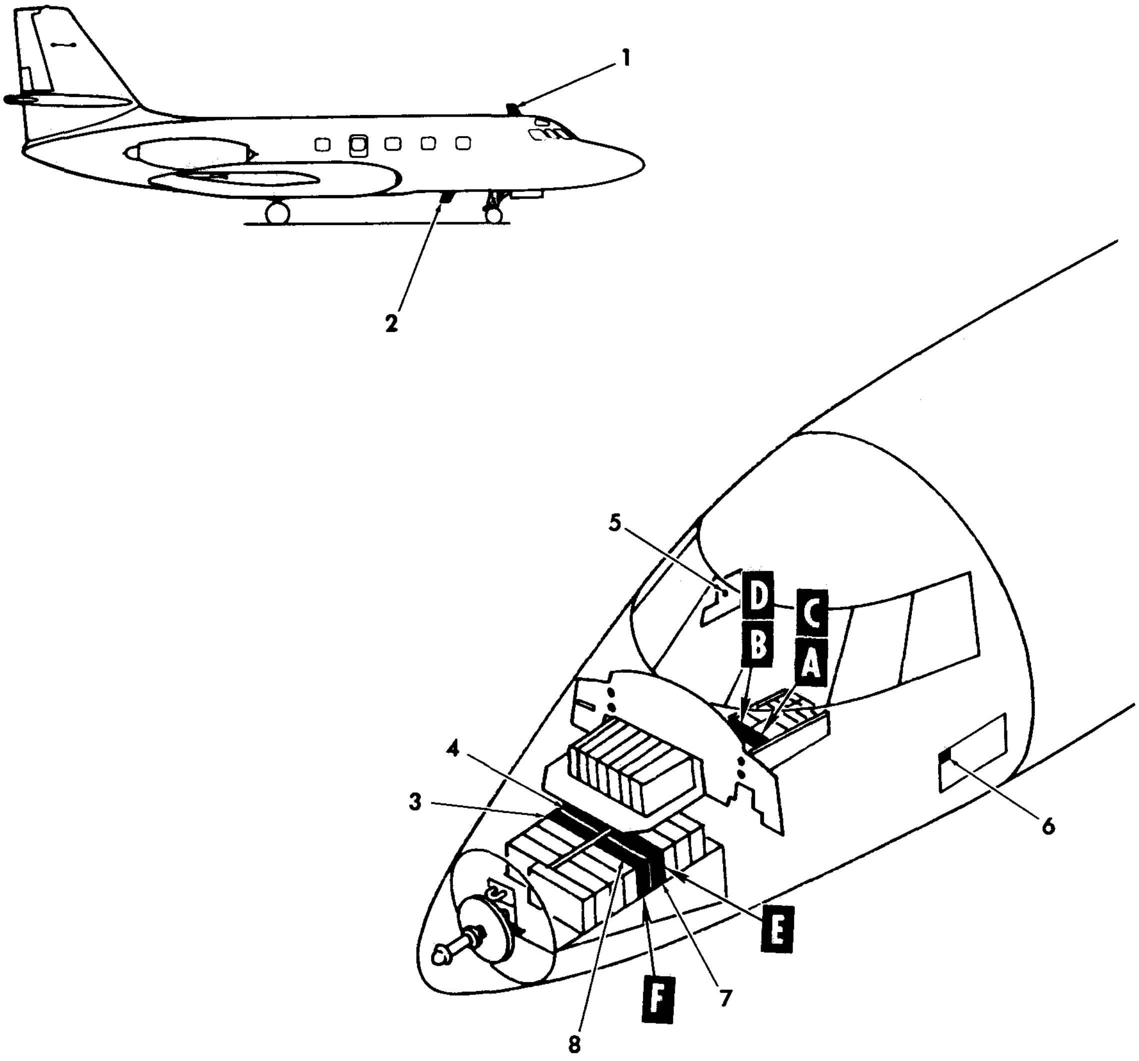
d. Connect the coaxial cable connector to the antenna.

e. Perform an operational checkout of the affected VHF communication system.

f. Replace the headliner or the floorboard that was removed during removal procedures.

g. Allow the antenna sealant to cure for about 36 hours at a temperature of 25°C(77°F) before pressurizing the airplane.

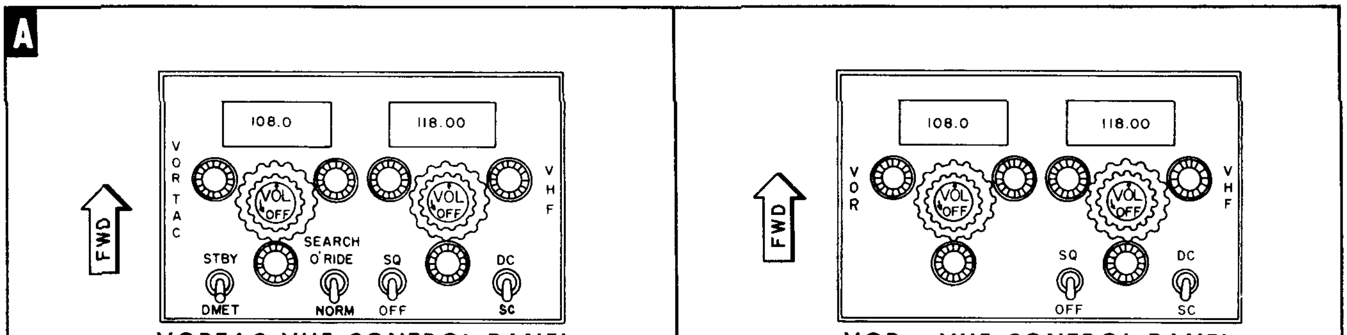
h. Perform a pressurization leak check.



1. NO. 2 VHF COMMUNICATIONS ANTENNA
2. NO. 1 VHF COMMUNICATIONS ANTENNA
3. NO. 2 VHF COMMUNICATIONS RECEIVER
4. NO. 2 VHF COMMUNICATION TRANSMITTER
5. NO. 2 VHF COMMUNICATION POWER RELAY
6. NO. 1 VHF COMMUNICATION POWER RELAY
7. NO. 1 VHF COMMUNICATION TRANSMITTER
8. NO. 1 VHF COMMUNICATION RECEIVER

JS-2-2-2303-1

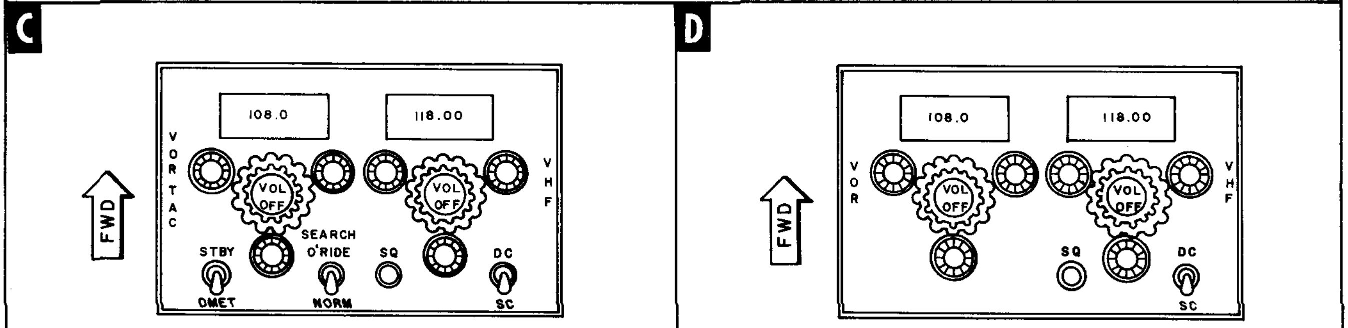
Figure 23-3. VHF Communication Systems Components (Sheet 1 of 2)



VORTAC-VHF CONTROL PANEL

VOR - VHF CONTROL PANEL

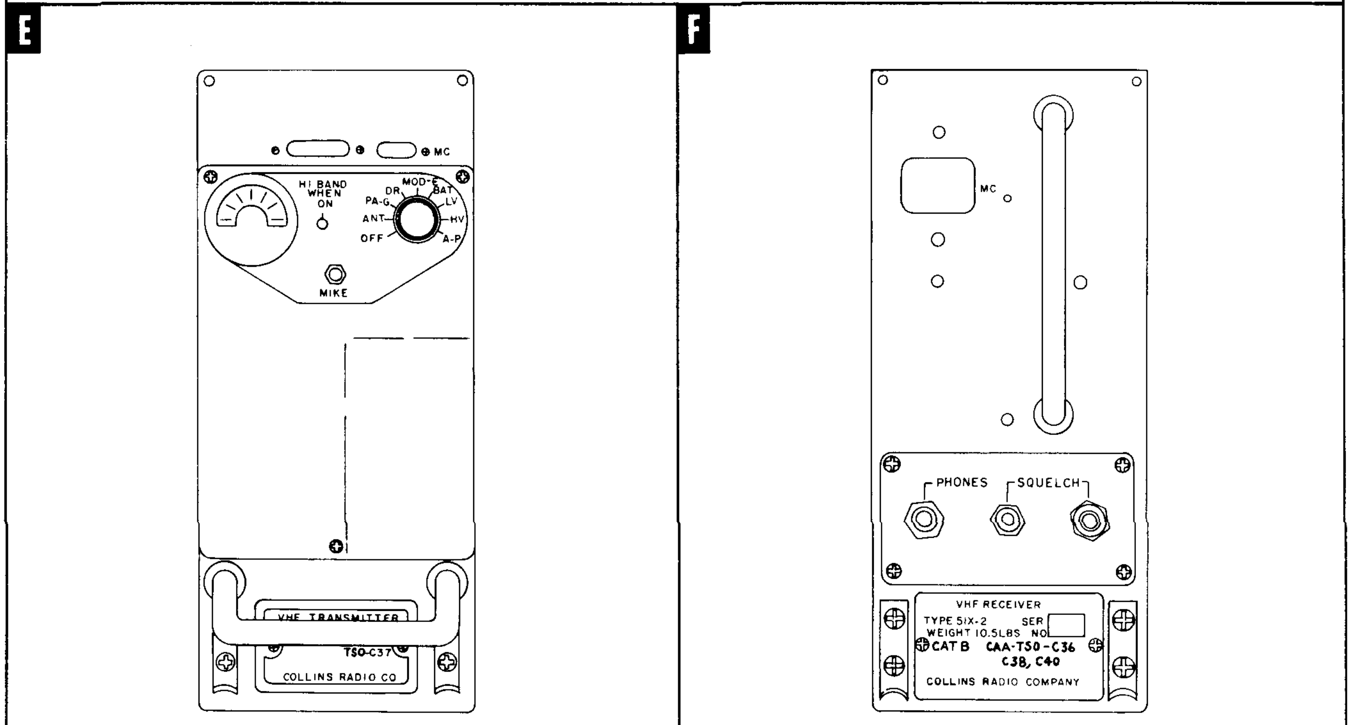
USED ON AIRCRAFT 5001 THRU 5038



VORTAC-VHF CONTROL PANEL

VOR - VHF CONTROL PANEL

USED ON AIRCRAFT 5039 THRU 5049, 5052, 5056, AND 5059



COLLINS 17L-7 TRANSMITTER
(LOOKING INBOARD)

COLLINS 51X-2 RECEIVER
(LOOKING INBOARD)

JS-2-3-2303-2

Figure 23-3. VHF Communication Systems Components (Sheet 2 of 2)

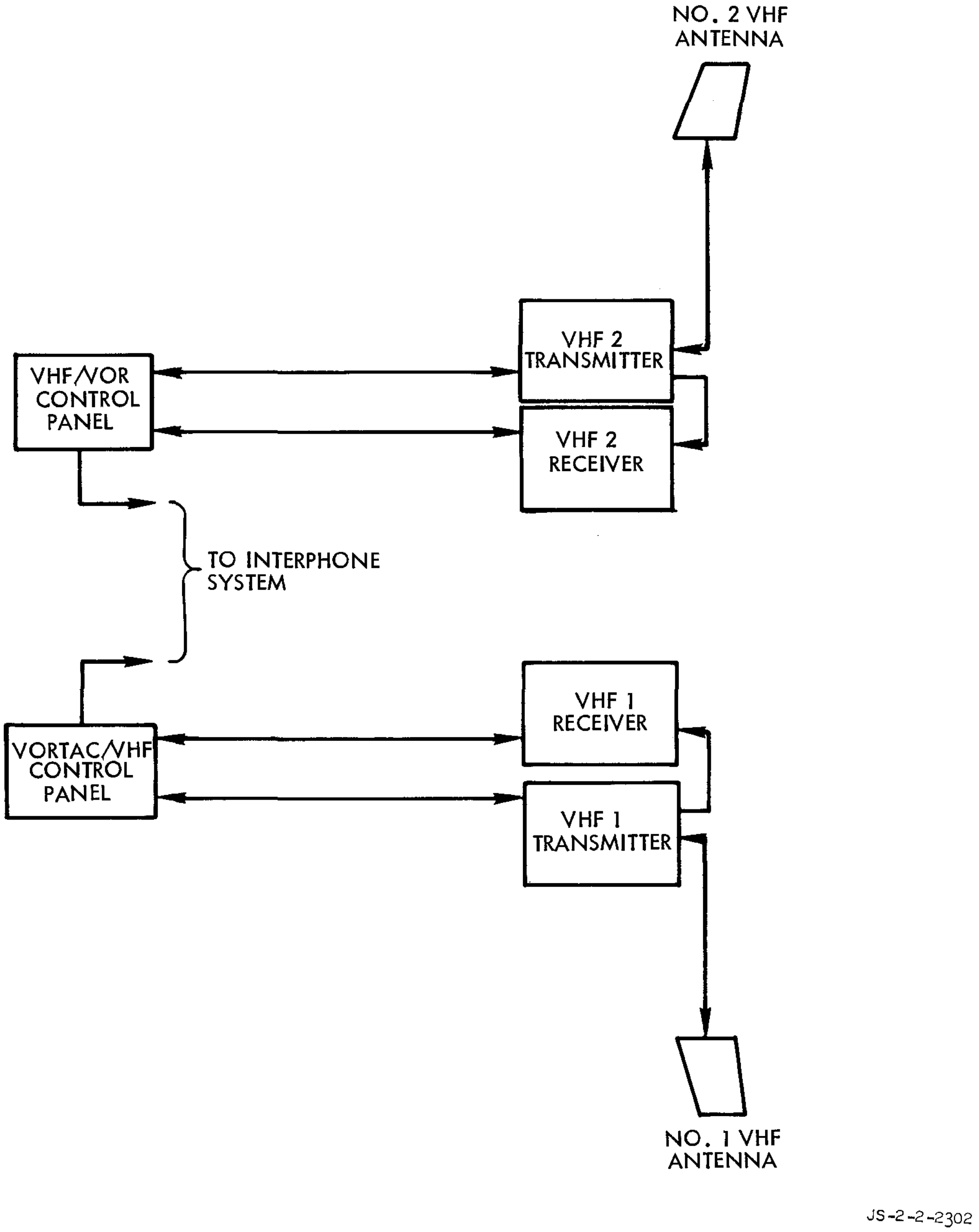


Figure 23-4. VHF Communication Systems Block Diagram

SECTION 23.3

HF COMMUNICATION SYSTEMS

Note

The Collins HF System is installed on some airplanes as optional equipment. Refer to this section for information if the complete system, provisions for the system, or a partial system is installed.

Contents Page

DESCRIPTION AND OPERATION 23-27
General 23-27
List of Components 23-28
Transceiver 23-28
Control Panel 23-28
Antenna Tuner 23-28
Antenna 23-28

OPERATIONAL CHECKOUT 23-28
HF Communication System 23-28

TROUBLESHOOTING 23-29
HF Communication System 23-29

REMOVAL AND REPLACEMENT 23-30
Antenna 23-30

FIGURES
23-5. Collins HF Communication System Components 23-33
23-6. The HF Antenna 23-34
23-7. HF Communication System Block Diagram 23-35

23.3.1 DESCRIPTION AND OPERATION.

23.3.1.1 GENERAL. (See figures 23-5 and 23-7.) Two-way plane-to-ground and plane-to-plane communication is provided by the Collins HF communication system, with operation in any one of 28,000 directly selectable frequencies in the frequency range of 2 to 30 megacycles. This system operates either in the single sideband mode with 400 watts peak envelope power, or in the AM mode with 125 watts of carrier power. The 115-volt AC and 28-volt DC required to operate the system are received from the essential AC and DC buses through two HF circuit breakers on the pilot's side circuit breaker panel. The system consists of a transceiver, an HF control panel, an HF antenna, and an antenna tuner. All transmission and reception is routed through an AUDIO SELECTOR panel.

23.3.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Control Panel	1	Collins	714E-2	522 2213 00 △1 522 2213 016 △2
Transceiver	1	Collins	618T-3	522 1660 00
Antenna	1	Lockheed		JR 460-1 △3 JR 460-16 △4
Antenna Tuner	1	Aerocom	5-JTR	AR-6166B

△1 Aircraft 5001 thru 5065.

△2 Aircraft 5066 and up.

△3 Aircraft 5001 thru 5074.

△4 Aircraft 5075 and up and preferred spare.

23.3.1.3 CONTROL PANEL. The remote control panel for the transceiver is mounted on the copilot's side console. The panel contains the necessary controls for turning the system on or off and for controlling the system when in operation. Controls on the panel include a function selector switch, a RF SENS knob, and a frequency selector. The four-position function selector switch has "OFF," "USB," "LSB," and "AM" positions. Placing the switch in either "USB," "LSB," or "AM" position turns the equipment on and selects the indicated type of reception and transmission. Placing the switch in the "OFF" position removes all power from the system. The frequency selector consists of four knobs and a rectangular window for displaying the operating frequency. The window covers five numbered wheels which are separated by a decimal point between the second and third numbered wheels. The knob on the extreme left controls the first two numbered wheels, which indicate the unit frequency (2 to 30 megacycles). The knob on the lower left controls the third numbered wheel, which indicates the frequency in tenths of megacycles. The knob on the lower right controls the fourth numbered wheel, which indicates the frequency in hundredths of megacycles. The knob on the extreme right controls the fifth numbered wheel, which indicates the frequency in thousandths of megacycles.

23.3.1.4 TRANSCEIVER. The Collins 618T-3 is attached to a shockmounted rack in the nose electronic equipment compartment. The transceiver, which combines the HF communication transmitter and receiver in one case, provides reception and transmission on any of 28,000 selectable frequencies in the 2- to 30-megacycle range. A built-in power supply unit is contained within the transceiver for providing low and high operating voltages. Control of the transceiver is from the remote control panel on the copilot's console. Two jacks, labelled "PHONE" and "MIC," are provided on the front panel of the transceiver for accommodation of a headset and a microphone. These jacks permit transmission and reception for use during operational checkout. A test meter and a meter selector switch, also on the front panel of the transceiver, provide a means for checking operating voltages and current in the transceiver. When the meter selector switch is in the "CAL TONE" position, the frequency of the transceiver can be checked with WWV. Transceiver components are cooled by a blower motor mounted directly behind the front panel. The transceiver automatically tunes to the channel selected on the HF control panel. When an HF microphone selector switch on one of the audio selector panels is positioned to "HF," the transmit-receive operation is controlled by either the corresponding microphone press-to-test button or the control wheel microphone switch.

23.3.1.5 ANTENNA. The Lockheed JR 460-1 or JR 460-16 antenna consists of a mast which contains an internally attached coaxial cable for connecting to the antenna tuner, and a single external wire antenna. The mast is attached to the top of the left wing, and the antenna wire is attached to a tension regulator on the No. 1 engine nacelle. Construction of the tension regulator permits the antenna wire to be fed through the regulator, taking up any slack in the wire. An internal spring in the regulator maintains constant tension on the antenna wire. The antenna forms a loop which couples the airframe to the transceiver.

23.3.1.6 ANTENNA TUNER. The Aerocom AR-6166B antenna tuner is attached to a shockmounted rack in the lower left side of the lavatory area. The tuner is used to automatically tune the antenna to match the impedance of the transceiver. The tuner is placed in operation when the transceiver is keyed. During the tuning operation of the antenna tuner, a tuning tone is audible through the headsets; this tone ceases when tuning is complete. During transmission, the tuner breaks the receiving circuit to prevent high-voltage feedback into the receiver. During reception, the tuner does not tune. A VSWR (Voltage Standing Wave Ratio) meter on the front of the tuner indicates the accuracy of antenna tuning for any particular operating frequency. During transmission, the indication on the VSWR meter should read better than 1.3 to 1.

23.3.2 OPERATIONAL CHECKOUT. An operational checkout of the Collins HF communication system is given in the following paragraphs. See the Airborne SSB Transceiver 618T-1, 618T-2, and 618T-3 Installation Manual, Publication Number 520-5970-006 for a bench checkout of the system.

23.3.2.1 OPERATIONAL CHECKOUT OF THE COLLINS HF COMMUNICATION SYSTEM. A source of 28-volt DC external power is needed to perform this test.

WARNING

During ground operation of the HF communication system, keep all personnel and equipment at least several inches away from the airplane exterior, particularly the HF antenna. Arcing can occur at most points on the airplane, depending on the operating frequency of the HF system. This arcing can cause penetrating burns to personnel and fire or explosion in ground equipment.

a. Connect external electrical power to the airplane according to the instructions in Chapter 24, and energize the AC buses.

b. Engage HF and INPH circuit breakers.

c. On one of the AUDIO SELECTOR panels, position the HF selector switch on; on the HF control panel, place the function selector switch in the "AM" position. Allow 15 minutes for the system to warm up.

d. Make sure that the blower in the HF transceiver is functioning.

e. Select eight different AM stations which are representative of the entire selector band.

Note

Do not press the press-to-talk button.

RESULT: Each time a station is selected, the transceiver should tune to the power frequency.

CAUTION

If the transceiver does not tune within 1 minute, reselect the same frequency. If the transceiver again does not tune within 1 minute, turn the system off until the trouble can be corrected.

f. On the HF control panel, rotate the RF SENS knob clockwise.
RESULT: The background noise in the receiver should increase in volume.

g. On the transceiver, position the meter selector switch to the "130V" position.
RESULT: The meter should indicate in the red area of the scale.

h. On the transceiver, position the meter selector switch to the "28V" position.
RESULT: The meter should indicate in the red area of the scale.

i. On the HF control panel, select an authorized frequency that can be used for test purposes.

j. On the transceiver, position the meter selector switch to the "PA MA" position, and key the system.
RESULT: The meter should indicate approximately 300 MA.

CAUTION

The blower in the transceiver should speed up when the HF system is keyed. If it does not, unkey the system immediately and check the blower.

k. On the transceiver, position the meter selector switch to the "150V" position, and key the system.
RESULT: The meter should indicate in the red area of the scale.

l. Select an authorized test frequency; after the transceiver has tuned, depress the press-to-talk button. When the tuning tone ceases, get a report on the quality of transmission.

CAUTION

If the antenna tuner does not tune within 1 minute, turn the system off until the trouble is corrected.

m. Turn the HF system off, and remove external power from the airplane.

23.3.3 TROUBLESHOOTING.

23.3.3.1 HF COMMUNICATION SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE
HF COMMUNICATION SYSTEM APPEARS TO BE INOPERATIVE.	
HF circuit breaker open.	Check circuit breaker.
Fault in the intercommunication system.	Check reception and transmission at the transceiver, using the PHONE and MIC jacks provided.
Defective transceiver.	Substitute operable transceiver.
Defective antenna tuner.	Substitute operable antenna tuner.
Defective control panel.	Substitute operable control panel.
Damaged antenna.	Inspect antenna.

23.3.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
RECEPTION NORMAL BUT NO TRANSMISSION CAN BE MADE.	
Fault in intercommunication system.	Check transmission by transmitting through the MIC jack on the transceiver.
Defective transceiver.	Check transceiver test meter for correct readings. Select other operating frequencies to see if the set will transmit on other frequencies.
Defective antenna tuner.	Substitute operable antenna tuner.
TRANSMISSION NORMAL BUT NO RECEPTION.	
Fault in the intercommunication system.	Check reception at the transceiver, using the PHONE jack.
Defective transceiver.	Substitute operable transceiver.
Defective antenna tuner.	Substitute operable antenna tuner.
TUNING TONE DOES NOT STOP WITHIN ONE MINUTE.	
Defective antenna tuner.	Switch off immediately, and substitute an operable antenna tuner.

23.3.4 REMOVAL AND REPLACEMENT.

23.3.4.1 ANTENNA. (See figure 23-6.) The HF antenna is furnished as a complete kit for replacement of a damaged or inoperative antenna. The kit antenna is completely assembled to the mast and coaxial cable and has all fittings installed on the antenna wire except the terminals, chafing sleeve, and ball terminal at the tension regulator. To remove the old antenna and install the new antenna proceed as follows:

a. Disconnect the coaxial cable connector of the old antenna from the antenna tuner, and remove from coaxial cable.

b. Attach 12 feet of strong cord to the coaxial cable.

c. Loosen the clamps securing the coaxial cable to the fuselage and wing.

d. Remove the nine screws and washers from the antenna mast.

e. Remove the antenna mast from the wing, and withdraw the coaxial cable with it, leaving the cord in the airplane.

f. Detach the cord from the coaxial cable.

g. To gain access to the tension regulator, remove the No. 1 engine nose cowl lower access panel.

h. Disconnect the antenna wire from the tension regulator pulley by disassembling the guard from the regulator, removing the cable guard pin adjacent to the fairlead roller, and disconnecting the two terminals, one from the center bolt of the regulator and one from the screw through the fairlead roller.

i. Measure and note length of coaxial cable attached to new antenna. Install the new coax cable in the airplane, leading it into position with the cord previously left in the airplane.

j. Cut coax cable so that the length between antenna base and end of connector shall be 107.0 (±1.0) inches, and install coax connector. Connect cable to antenna tuner.

k. Install the new antenna mast on the wing with the nine screws and washers.

l. Secure the coaxial cable clamps on the wing and in the fuselage. Reseal coax cable where it passes through grommet in pressure skin. (See Section 51.2.)

m. Rotate the pulley of the tension regulator to align the scribe marks on the pulley and on the housing. Hold or secure the pulley in this position during the following step.

n. Position the new antenna wire, over the fairlead roller (in the pulley groove), making sure there is no slack in the antenna wire between the pulley and the antenna mast. The antenna wire should wrap almost one complete turn around the pulley. Using tape, mark on the antenna wire the location of the pulley cutout which will be engaged by the ball terminal to be installed on the antenna wire.

o. Remove the marked antenna wire from the tension regulator, and swage the AN664C4 ball terminal (supplied with the antenna kit) on the antenna wire at the location previously marked with tape, using a suitable swaging tool. The shank of the ball terminal should face towards the antenna mast.

p. Install the JR 460-14 terminal (supplied with the antenna kit) on the antenna wire at a location 1.75 (±0.1) inches from the ball terminal.

q. Slide the JR 460-13 sleeve (supplied with the antenna kit) on the antenna wire.

r. Install the MS25036-12 terminal (supplied with the antenna kit) on the antenna wire 5.1 (±0.1) inches from the ball terminal. Cut off the surplus antenna wire.

s. Install the antenna wire over the fairlead roller and on the tension regulator pulley, with the pulley positioned as in step (m) and the antenna wire positioned as in step (n), with the ball terminal inserted in the pulley cutout.

t. Check that the antenna wire has positive tension and that the regulator pulley rotates freely when the antenna wire is deflected and released.

u. Secure the JR 460-14 terminal (supplied with the antenna kit) to the bolt in the center of the tension regulator pulley with one nut and one washer which were removed in step (h).

v. Secure the MS25036-12 terminal to the screw through the fairlead roller with one nut and one washer which were removed in step (h).

w. Reinstall the tension regulator pulley guard and the fairlead roller guard pin.

x. Close the nose cowl access panel.

y. Perform an operational checkout of the HF communication system.

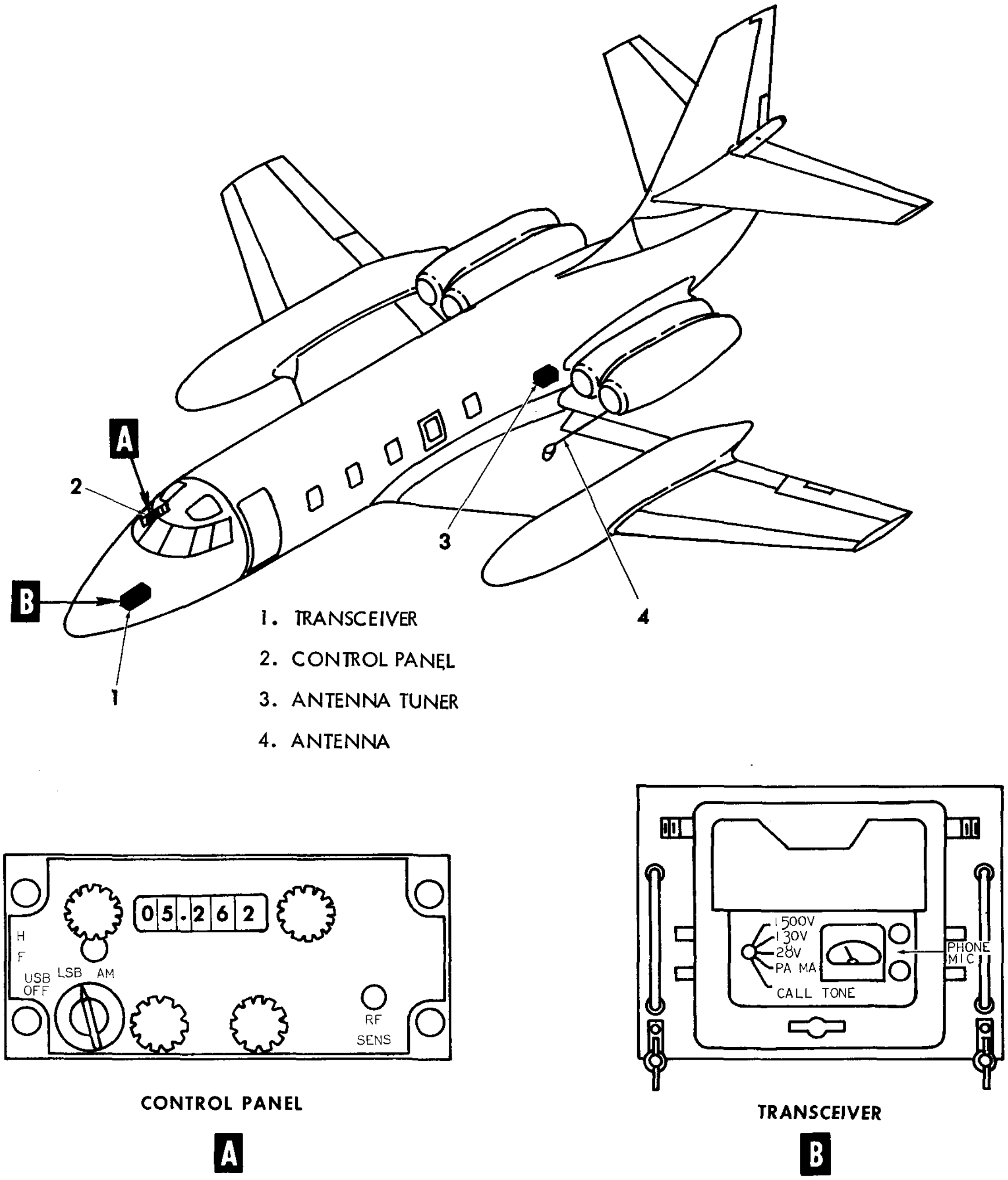
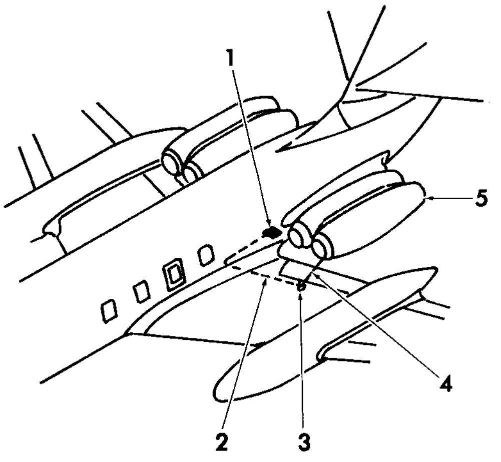
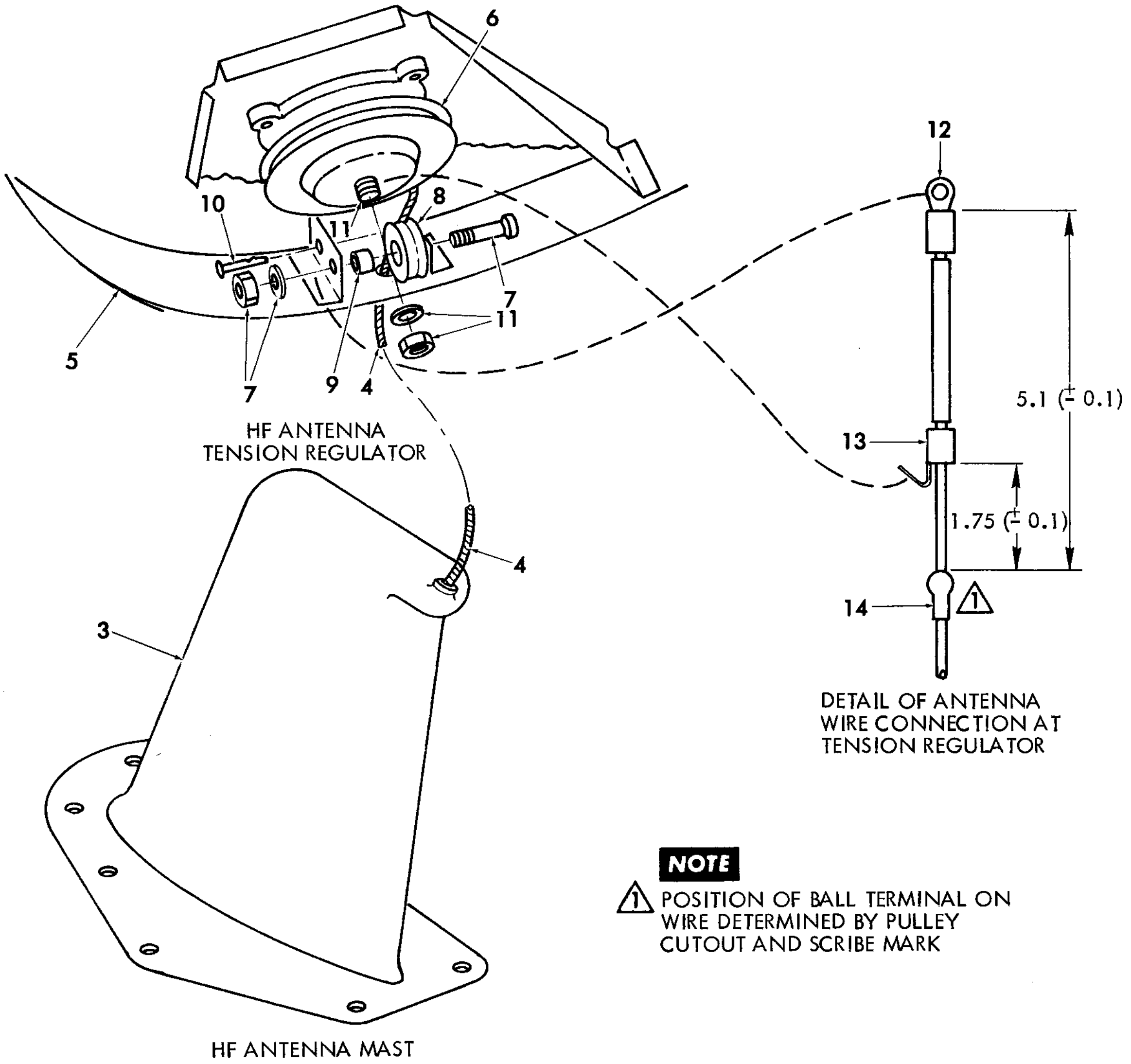


Figure 23-5. Collins HF Communication System Components



1. HF ANTENNA TUNER
2. COAXIAL CABLE
3. HF ANTENNA MAST
4. HF ANTENNA WIRE
5. NO. 1 NACELLE
6. TENSION REGULATOR PULLEY
7. SCREW, WASHER, AND NUT
8. FAIRLEAD ROLLER
9. SPACER
10. GUARD PIN
11. BOLT, WASHER, AND NUT
12. MS25036-12 TERMINAL
13. JR460-14 TERMINAL
14. AN664C4 BALL TERMINAL


6
10
8
11
7
11
5
7
9
4
HF ANTENNA TENSION REGULATOR
4
3
HF ANTENNA MAST
12
5.1 (± 0.1)
13
1.75 (± 0.1)
14
1
DETAIL OF ANTENNA WIRE CONNECTION AT TENSION REGULATOR

NOTE

⚠1 POSITION OF BALL TERMINAL ON WIRE DETERMINED BY PULLEY CUTOUT AND SCRIBE MARK

Figure 23-6. The HF Antenna

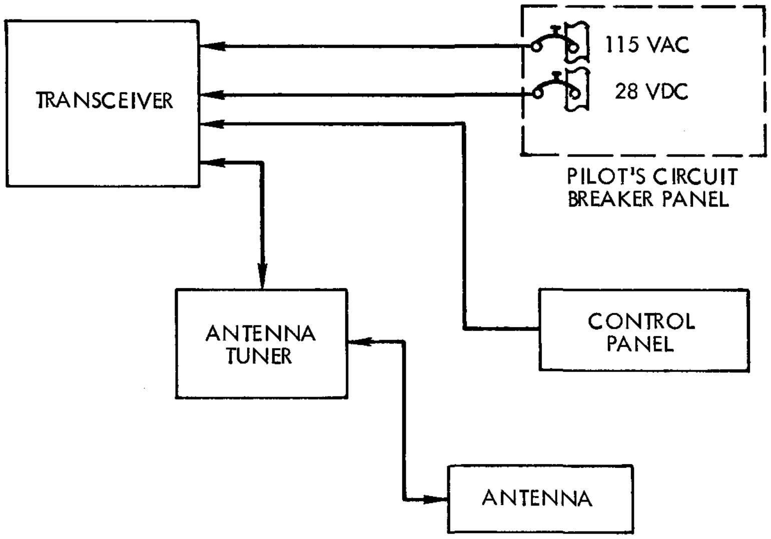


Figure 23-7. HF Communication System Block Diagram

CHAPTER 24
ELECTRICAL POWER

TABLE OF CONTENTS

Section Page

24.0 DC POWER SYSTEM 24-3

24.1 AC POWER SYSTEM (AIRPLANES 5001, 5002, 5004, THROUGH 5009) 24-35

24.2 AC POWER SYSTEM (AIRPLANES 5003, 5005, 5010 AND UP). 24-49

SECTION 24.0
DC POWER SYSTEM

Contents Page

DESCRIPTION AND OPERATION 24-4
  General 24-4
  DC Starter-Generators 24-4
  DC Generator Control Panels 24-4
  Equalizer Relays 24-5
  Equalizer Circuit (Airplanes Modified by Service Bulletin 329-294) 24-5
  Overvoltage Relays 24-5
  Overvoltage Circuit (Airplanes Modified by Service Bulletin 329-294) 24-5
  DC Voltage Regulators 24-5
  DC Voltage Regulator (Airplanes Modified by Service Bulletin 329-294) 24-5
  Ground Fault Relays 24-5
  Ground Fault Circuit (Airplanes Modified by Service Bulletin 329-294) 24-6
  Ground Fault Annual Check 24-6
  Field Relays 24-6
  Differential Relays 24-6
  Overvoltage Lockout Disconnect Relay 24-6
  Generator Out Relays 24-6
  Generator Contactors 24-6A
  Generator Contactors (Airplanes Modified by Service Bulletin 329-294) 24-6A
  Generator Transfer Relays 24-6A
  Bus Tie Transfer Relay 24-6A
  Normal Reverse Current Cutout 24-6A
  Emergency Reverse Current Cutout 24-6B
  Starter Relays 24-6B
  DC Control Panel 24-6B
  Generator Control Switches 24-6B
  Field Trip-Reset Switches 24-6B
  DC Generator Out Lights 24-7
  DC Generator Loadmeters 24-7
  DC Voltmeter 24-8
  Emergency Generator Selector Switch 24-8
  Generator Overheat Warning 24-8
  Battery System 24-8
  Batteries 24-8
  Battery Switch 24-8
  Battery Relay 24-8
  Battery Relay Rectifiers 24-8
  DC Power Distribution 24-8
  Main DC Bus 24-8
  Essential DC Bus 24-8
  Ignition Bus 24-8
OPERATIONAL CHECKOUT 24-9

TROUBLESHOOTING 24-10
  Starter-Generator 24-10
  Batteries 24-12

MAINTENANCE DC ELECTRICAL SYSTEM 24-12
  DC Starter-Generator 24-12
  Brush Inspection 24-13
  Power Shield Fasteners 24-15
  Battery Disconnect 24-15

PRESERVATION AND DEPRESERVATION 24-15
  Preservation of Electrical Equipment 24-15
  Depreservation of Electrical Equipment 24-16

Contents Page

FIGURES

24-1. External Power Connections 24-19
24-2. DC Power Supply Components 24-20
24-3. DC Power Distribution Diagram 24-23
24-4. Normal DC Power Schematic Diagram (Airplanes 5001, 5002, 5004, and 5006) 24-25
24-5. Normal DC Power Schematic Diagram (Airplanes 5003, 5005, 5007, and Up) 24-27
24-5A. Normal DC Power Schematic Diagram (Airplanes Modified by Service Bulletin 329-294) 24-28A
24-6. Emergency DC Power Schematic Diagram (Airplanes 5001, 5002, 5004, and 5006) 24-29
24-7. Emergency DC Power Schematic Diagram (Airplanes 5003, 5005, 5007, and Up) 24-30
24-7A. Emergency DC Power Schematic Diagram (Airplanes Modified by Service Bulletin 329-294). . . 24-30A
24-8. Generator Control Panel Schematic Diagram (Airplanes Not Modified by Service Bulletin 329-294) 24-31
24-9. Battery System Schematic Diagram 24-32
24-9A. Generator Out Lights 24-32A
24-9B. Essential-Main Bus Tie (Airplanes 5066 and Up) 24-32B
24-10. Circuit Breaker Panel and Fuses 24-33
24-11. Voltmeter-Loadmeter Schematic Diagram 24-34
24-11A. Bus Tie Warning Light (Airplanes 5059 thru 5085) 24-34A
24-11B. Battery Connector Inspection Gauge 24-34A

24.0.1 DESCRIPTION AND OPERATION.

24.0.1.1 GENERAL. Four starter-generators supply the DC power for the airplane. This electrical power is distributed from an electrical power shield and from circuit breaker panels in the pilots' compartment. (See figures 24-2 and 24-3.) Two 24-volt batteries store electrical power.

a. Normally, the four engine-driven starter-generators supply 28 volts DC, through respective generator contactors in the electrical power shield, to the main DC bus, which provides 28-volt power for the essential DC bus through the normal reverse current cutout. Power from the essential DC bus charges the batteries. Two starter-generators can supply sufficient power for ordinary requirements.

b. If power is lost to the essential bus, the output of a selected starter-generator can be routed to the essential DC bus through the circuit which includes a starter relay, the starter bus, and the emergency reverse current cutout; in this way the main DC bus is bypassed. If none of the starter-generators is operable, battery power can supply the essential DC bus, but the main DC bus cannot be energized. During ground operations, an external source of 28-volt, DC power can be connected to the main DC bus, and in turn to the essential DC bus.

c. Two electrical power receptacles are located side-by-side in a recessed compartment on the right side of the forward fuselage. (See figure 24-1.) The start receptacle provides a source of DC power for operating the engine starter-generators during the engine starting period. The service receptacle provides a source of DC power for engine ignition, inverters, ground checkout of equipment, lighting, and other miscellaneous service functions.

d. Routing of 28-volt, DC power to the starter, main, ignition, and essential buses is automatic, once the plug from the external power source is connected to the external power receptacles. Two static grounding receptacles, one on the right side of the forward fuselage and the other on the aft fuselage, are provided for grounding the airplane when using an external power source.

24.0.1.2 DC STARTER-GENERATORS. (See figures 24-4 thru 24-7.) A DC starter-generator is mounted on a drive pad on each of the four engines. Rated output of each generator is 300 amperes at 30 volts when running at a rated speed of 3,000 to 8,000 RPM. The continuous operating speed is 6,500 RPM. Voltage at a rated speed is maintained by the shunt field current controlled by a carbon pile voltage regulator in a DC generator control panel. During flight, forced air cools the generator.

24.0.1.3 DC GENERATOR CONTROL PANELS. (See figures 24-2 and 24-8.) The operation of each starter-generator is automatically governed by a generator control panel on an electrical equipment rack located in the aft compartment. The functions of the generator control panels include regulating output voltages, equalizing loads, protecting against ground faults and overvoltage, and allowing starter-generators to be connected to the main DC bus only when output voltages are within the proper limits. During engine

24.0.1.3 (Cont).

starting, the overvoltage protection is suspended to accommodate increased voltage, such as that of the batteries in series; the voltage regulator is bypassed to permit the shunt field to build up unrestricted. Each generator control panel consists of an equalizer relay, ballast lamps, a DC voltage regulator, an overvoltage relay, an interlocking relay, a differential relay, and a starting control relay.

24.0.1.3.1 Airplanes modified by Service Bulletin 329-294 have lightweight solid state generator control panels in lieu of the carbon pile control panels. During engine starting, the generator control panel is bypassed, thereby eliminating the need for overvoltage protection. (See figures 24-5A and 24-7A).

24.0.1.4 EQUALIZER RELAYS. The equalizer circuit connects between terminal D of all four generators, and the generator control panel equalizer relays to an equalizing bus, balancing loads and equalizing output voltages of the starter-generators in use during normal operation. If a generator is not accepting its equal share of the total load, its potential across E and D terminals of the generator will be different from those of the other generators. This difference in potential causes a current to flow through the equalizer circuit. As this current flows through the equalizer parallel coils, it aids or opposes (depending on the direction of current flow) the regulator coils of the voltage regulators, thereby raising or lowering the resistance of the field circuit, which results in changing the terminal voltage until the current outputs of all generators are equal. The generators are designed to have a 2-volt drop between terminal D and E at rated load. If a generator will not share the load, the series field winding may be deteriorated. This deterioration will cause the resistance of the series winding to increase, which will increase the voltage drop for a given load.

24.0.1.4.1 EQUALIZER CIRCUIT (AIRPLANES MODIFIED BY SERVICE BULLETIN 329-294). The equalizer system of a generator control panel uses an integrated circuit to detect the difference between the associated generator interpole voltage and the "equalizer circuit" voltage. This voltage difference is amplified and coupled through weighted inputs to the "summing junction" of the voltage regulator circuit and the overvoltage circuit. The equalizer circuit functions to raise or lower the generator voltage to balance the output of the generators. The differential voltage is sensed between terminal "D" of the generator and the equalizer circuit which connects pins 24/25 of all the control panels together. When a generator is transfered to EMERGENCY, it is disconnected from the normal equalizer circuit.

24.0.1.5 OVERVOLTAGE RELAYS. If the output voltage of a starter-generator is excessive (35 volts DC without aid of bias coil), an overvoltage relay in the affected generator control panel opens the shunt field circuit. Each of the four overvoltage relays has two coils, an overvoltage biasing coil connected to the equalizer bus, and an overvoltage relay coil energized by the output voltage of the corresponding starter-generator. In an overvoltage condition, the overvoltage relay contacts are closed, the field trip relay coil is energized, and the shunt field of the starter-generator concerned is opened. Overvoltage protection is suspended during ground engine starting. Overvoltage ground fault test switches are located adjacent to the DC control panels in the aft compartment. When the starter-generator is "on the line" and the TEST switch is depressed to the "OVERVOLTAGE" position, the starter-generator will be tripped "off the line." The GEN OUT light will be illuminated and the voltage will drop to 0-5 volts.

24.0.1.5.1 OVERVOLTAGE CIRCUIT (AIRPLANES MODIFIED BY SERVICE BULLETIN 329-294). The overvoltage protection circuitry consists of integrated circuits that sense the generator output voltage and will cause the generator to be de-energized and removed from the bus if an overvoltage condition exists.

24.0.1.6 DC VOLTAGE REGULATORS. A carbon pile voltage regulator, one mounted on each of the four DC generator control panels, varies the shunt field current of each starter-generator. Resistance, and consecuently shunt field current and voltage, is altered by either the compression or expansion of the carbon pile according to the extent of actuation of the two controlling coils. The output voltage is regulated by the varying shunt field current. The voltage regulator operating coil is energized by the starter-generator output voltage, and the voltage regulator biasing coil by the equalizing circuit voltage. In this manner, the output voltage of each starter-generator is held at approximately 28 volts. Minor adjustments can be made by a rheostat in the base of each voltage regulator. During ground engine starting, shunt field voltage regulation is bypassed for the selected starter-generator.

24.0.1.6.1 DC VOLTAGE REGULATOR (AIRPLANES MODIFIED BY SERVICE BULLETIN 329-294). A solid state regulator varies the shunt field current of each starter-generator. The output voltage is regulated by the varying shunt field current. Minor adjustments can be made by a rheostat in each voltage regulator.

24.0.1.7 GROUND FAULT RELAYS. Two torus-type transformers, one circling the feeder line and the other the ground line, operate in series to detect ground faults in each of the four starter-generator circuits. Normally, feeder line current for a starter-generator is the same as the ground line current. Consequently, any line current changes induce equal but opposed voltages in the transformers of the related ground fault circuits. But, if a fault occurs in a starter-generator in either its feeder or ground line, voltages are induced in the transformers. This results in a flow of current in the ground fault circuit, and the ground fault relay in the affected generator control panel is actuated. The field relay trip coil is then energized, thereby removing shunt field excitation from the faulty starter-generator. If a ground fault exists during voltage build-up, it is possible the voltage and current would not change rapidly

24.0.1.7 (Cont).

enough to be sensed by the ground fault transformers. To preclude this, the generator contactor in the open position provides a ground circuit through terminal B3, which senses a voltage drop (caused by current flow in the starter-generator) between terminal D of the starter-generator and ground, thereby allowing sufficient current to flow through the ground fault relay coils, through the rectifier to terminal D of the starter-generator. This causes the field-trip relay to energize and remove shunt field excitation from the respective starter-generator. In each of the four ground fault circuits, the line from terminal D to the ground fault relay coils is opened by the start control relay to prevent current during starting from flowing from the starter-generator through the ground fault circuit to ground. Otherwise, the field relay would trip during each start. Overvoltage-ground fault test switches are located on the DC control panel. (See figure 24-2.) When the starter-generator is "on the line" and the TEST switch is depressed to the "GROUND FAULT" position the starter-generator will be tripped "off the line," the GEN OUT light will be illuminated, and the voltage will be decreased. A test to confirm correct operation should be run on all four starter-generators every 100/300 hours.

24.0.1.7.1 GROUND FAULT CIRCUIT (AIRPLANES MODIFIED BY SERVICE BULLETIN 329-294). Two torus-type transformers, one circling the feeder line and the other the ground line, operate in series to detect ground faults in each of the four starter-generator circuits. Normally, feeder line current for a starter-generator is the same as the ground line current. Consequently, any line current changes induce equal but opposed voltages in the transformers of the related ground fault circuits. But, if a fault occurs in a starter-generator in either its feeder or ground line, unequal voltages are induced in the transformers. This results in a flow of current in the ground fault circuit, and the ground fault circuit in the affected generator control panel is actuated. The field relay trip coil is then energized, thereby removing shunt field excitation from the faulty starter-generator. If a ground fault exists during voltage build-up, it is possible the voltage and current would not change rapidly enough to be sensed by the ground fault transformers. To preclude this, a self-sustaining circuit, within the protection panel, measures the voltage across the generator interpole winding before the line contactor has originally closed. If the interpole voltage represents a current flow equal to one-half or more, of the generator's rating, the line contactor circuit will prevent the line contactor from closing and an input to the overvoltage integrator will cause the circuit to de-energize the generator. Overvoltage-ground fault test switches are located on the aft compartment electrical equipment rack. (See figure 24-2.) When the starter-generator is "on the line" and the TEST switch is depressed to the "GROUND FAULT" position, the starter-generator will be tripped "off the line," the GEN OUT light will be illuminated, and the voltage will be decreased.

24.0.1.7.2 GROUND FAULT ANNUAL CHECK.

a. The following test should be accomplished annually to ensure positive ground fault protection.

b. Stop engines and remove all power from the aircraft.

c. Connect external loadbank to terminal "B" of the No. 1 starter-generator and set for a 200-ampere load.

Note

Avtron Model K576, P/N D17521 or similar loadbank should be used. Contact: Avtron Manufacturing, Inc., 10409 Meech Avenue Cleveland, OH 44105, Phone 216-642-1230.

Note

The total loading capability of the AVTRON Model K576 is 500 amps at 52 VDC. The 28 VDC airplane system must be taken into consideration when performing this check. AVTRON states: "Lower voltages may be applied to the load circuit of the load bank, however, amperage with a lower voltage is computed using the following formula:"

$$\text{Actual Load (Amps Applied)} = \text{Selected Load (Amps Rated)} \times \frac{\text{Applied Voltage}}{\text{Rated Voltage}}$$

Therefore, to achieve 200 amp actual load from a 28 VDC generator, a selected load of 375 amperes must be selected on the AVTRON Model K576 load bank.

When using other load banks, the selected load may be different based on the electrical characteristics of that unit.

d. Start No. 1 engine.
RESULT: No. 1 generator will not come on the line.

e. Turn loadbank "OFF" and reset generator.
RESULT: Generator will come on bus.

f. With No. 1 generator on bus, suddenly apply 200-ampere external load.
RESULT: Generator will trip off line.

g. Repeat steps b, c, d, e, and f for engines 2, 3, and 4.

24.0.1.8 FIELD RELAYS. When an overvoltage or a ground fault condition occurs, the field trip coil in the appropriate generator control panel actuates its field relay. The shunt field is then opened, and the corresponding generator contactor disconnects the affected starter-generator from the main DC bus. A field relay can also be tripped by a corresponding TRIP-RESET switch on a DC control panel or a corresponding fire emergency handle. Once a field relay has been tripped, the relay must be reset by placing its TRIP-RESET switch in the "RESET" position.

This energizes the reset coil and the field relay closes; this restores the shunt field excitation and allows the starter-generator to build up again. An interlocking relay keeps the field relay from cycling if it is tripped again while the momentary contact TRIP-RESET switch is held in the "RESET" position. On the ground, the field relay can be reset manually on carbon pile generator control panels by pressing the reset lever on the front of the panel.

24.0.1.9 DIFFERENTIAL RELAYS. Power for energizing each of the four generator contactors is supplied through their differential relays. Each relay has two coils: the differential voltage relay coil and the reverse current coil. The differential voltage relay coil actuates the relay to close the contacts when the output voltage of the corresponding starter-generator exceeds the main DC bus voltage by 0.4 to 0.6 volt. These relay contacts remain closed unless a current of 18 to 35 amperes flows from the main DC bus to the starter-generator. Then sufficient current flows through the related reverse current coil to actuate the differential relay and open the contacts. The corresponding generator contactor is then de-energized, and the affected starter-generator disconnected from the main DC bus. Airplanes that have Service Bulletin 329-294 incorporated do not have differential relays. (Refer to paragraph 24.0.1.12.)

24.0.1.10 OVERVOLTAGE LOCKOUT DISCONNECT RELAY.

Note

The following explains the use of the Disconnect Relay on Aircraft 5001, 5002, 5004 and 5006 only. Refer to Chapter 80 for information regarding the Disconnect Relay on all other aircraft.

During emergency generator operation the Overvoltage Lockout Disconnect Relay is energized to open the circuit between terminal C of the starter relay and terminal 1 of the overvoltage lockout relay. This holds open the circuit connections between terminal R of the Starter Relay, and Terminal X1 of the Overvoltage Lockout Relay, and ground Overvoltage protection is eliminated if these circuits are not open. (See figure 24-6.)

24.0.1.10.1 On airplanes modified by Service Bulletin 329-294, overvoltage lockout protection is not required. Refer to Chapter 80 for use of the disconnect relay in the starting system.

24.0.1.11 GENERATOR OUT RELAYS. Within the power shield are four generator out relays, one for each starter-generator. When a generator contactor is closed, the appropriate generator out relay is energized, thereby completing the loadmeter circuit for the corresponding starter-generator. During a battery start condition the Generator Out Relay completes a ground circuit for the

24.0.1.11 (Cont).

Generator On Sense Relay which opens the ignition bus transfer relay circuit. (Refer to Chapter 80.) If a generator contactor opens, the related generator out relay is de-energized, the corresponding loadmeter circuit is opened, and an appropriate GEN OUT light goes on. (See figure 24-9A.) Simultaneously, a GENERATOR OUT module on the annunciator panel goes on.

24.0.1.12 GENERATOR CONTACTORS. When the DC control panel ON-OFF-EMER generator switches are turned "ON," and all engines and generators are operating, the generator-contactors are energized. This connects the starter-generators to the main DC bus, in parallel. Individual generator loads are indicated by the loadmeters. All loads are equalized by the load equalizer circuits. The four generator-contactor relays are in the main power shield. When energized, the relays connect their respective generators to the main DC bus. The main contacts of the relay of each generator-contactor are the heavy duty type. The main coil of the relay energizes when the input voltage rises to 18 volts. When the main coil energizes, the auxiliary coil will be energized, adding resistance in series with the main coil. This creates an economizer circuit which reduces the amount of current flow required to hold the contactor closed. If the main bus has voltage, the generator control circuits prevent the main contactor from being energized until the generator voltage exceeds the main DC voltage by 0.4 to 0.6 volts. Once the main contacts are closed, a current of 18 to 35 amperes flowing to the starter-generator from the main DC bus causes the reverse current relay in the generator control panel to open. This, in turn, de-energizes the generator contactor and causes it to open. For each generator contactor, these voltage limits and reverse currents are detected by the differential relays within the respective generator control panels.

24.0.1.12.1 GENERATOR CONTACTORS (AIRPLANES MODIFIED BY SERVICE BULLETIN 329-294). Generator contactor operation is the same as described in paragraph 24.0.1.12 except that if the main bus has voltage, the generator control circuits prevent the main contactor from being energized until the generator voltage equals the main DC bus voltage. Once the main contacts are closed, a current of approximately 50 amperes flowing to the starter-generator from the main DC bus causes the reverse current circuit in the generator control panel to de-energize the generator contactor and causes it to open. For each generator contactor, these voltage limits and reverse currents are detected by protective circuits within the respective generator control panels.

24.0.1.13 GENERATOR TRANSFER RELAYS. Four generator transfer relays, one for each starter-generator, are mounted on the generator transfer relay panel. The primary function of the generator transfer relays is during emergency generator operation. When emergency generator operation for a particular starter-generator is selected, its corresponding generator-transfer relay is energized completing the following circuits:

a. Starter relay is energized, connecting starter bus to generator output.

b. Bus-tie transfer relay is energized connecting starter bus to essential bus through emergency reverse current relay.

c. The source of power for controlling the generator-out relay is transferred from the generator contactor to generator transfer relay. This provides correct generator-out indication even though the generator contactor is open.

d. Overvoltage lockout disconnect relay is energized and functions as described in paragraph 24.0.1.4 (airplanes 5001, 5002, 5004, and 5006 only).

e. The ground circuit through normally open generator contacts for ground fault circuit is open. (Refer to 24.0.1.7.)

During normal generator operation all the generator transfer relays remain in the de-energized position. In this position the generator transfer relay completes:

a. Generator out relay power circuit.

b. Ground circuit for ground fault protection. (Refer to 24.0.1.7.)

24.0.1.14 BUS TIE TRANSFER RELAY. Located on the generator transfer relay panel, a bus tie transfer relay is normally de-energized. In this position, control power from the main DC bus is supplied through the bus tie transfer relay to the SW terminal of a normal reverse current cutout. When one of the generator transfer relays is energized, the bus tie transfer relay is also energized. Power then passes from the starter bus through the bus tie transfer relay to the SW terminal of the emergency reverse current cutout. This action separates the essential DC bus from the main DC bus, and connects it to the starter bus.

24.0.1.15 NORMAL REVERSE CURRENT CUTOUT. Electrical current is allowed to flow from the main DC bus to the essential DC bus through a normal reverse current cutout in the power shield, but current flow exceeding 18 to 35 amps in the opposite direction is prevented. When the bus tie relay is de-energized, power from the NORM BUS TIE RELAY circuit breaker energizes both the voltage relay coil and the biasing coil in the cutout. With the voltage relay contacts closed, current can flow from the main DC bus through the reverse current coil and differential voltage coil of the cutout. The current relay then closes, allowing the main contactor coil to be energized. The main

24.0.1.15 (Cont).

contacts remain closed until power is removed from the SW terminal of the cutout, or current begins to flow from the essential DC bus to the main DC bus, in which case both the reverse current coil and the differential coil are energized in the opposite direction. The reverse current relay opens, the main contactor is de-energized, and the main contacts are opened.

On airplanes 5059 and up (see figure 24-11) a NORM BUS TIE OPEN light has been provided which indicates when the normal reverse current relay has opened. For airplanes 5059 through 5085, this light is located on the overhead generator control panel. For airplanes 5086 and up, this light is on the annunciator panel. The purpose of the light is to warn the crew that the relay has failed and that the batteries are supplying all the power for the essential bus loads. Therefore, when this condition exists, the loads in the essential bus should be monitored to preclude complete battery discharge. As long as the relay is functioning correctly (closed), the bus tie warning relay will remain energized through the IND terminal of the normal reverse current relay. If the relay opens, the bus tie warning relay de-energizes and provides a ground circuit, closing the master caution relay. This action causes the MASTER CAUTION lights and NORM BUS TIE OPEN light on the overhead electrical control panel to illuminate. Depressing the MASTER CAUTION light to reset extinguishes the MASTER CAUTION light but not the NORM BUS TIE OPEN light.

For airplanes 5066 and up (see figure 24-9B) an ESS-MN BUS TIE switch is provided which connects the essential bus to the main bus to allow ground checkout of the main bus systems, using battery power. By placing the ESS-MN BUS TIE switch, located at the pilot's side (fuselage station 232 bulkhead) in the closed position, power is provided to the normal reverse current cutout relay APP terminal through the normally open (energized position) contacts of the touchdown relay. The touchdown relay is energized through the nose gear scissor switch whenever the aircraft is on the ground and the essential bus is "hot." The ESS-MN BUS TIE switch is guarded to prevent inadvertent actuation of the system. In addition, an engine start cannot be initiated until the ESS-MN BUS TIE switch is placed in the guarded open position. The purpose for this is to ensure, during battery start, that the essential and main buses are disconnected.

24.0.1.16 EMERGENCY REVERSE CURRENT CUTOUT. During an emergency, power is routed from a selected starter-generator through the start relay, start bus, and the emergency reverse-current cutout to the essential DC bus. For this to occur, the bus tie transfer relay is energized, and power from an EMER BUS TIE RELAY circuit breaker energizes the voltage relay coil and the biasing coil in the emergency reverse-current cutout. Once the voltage coil relay contacts are closed, current from the starter bus can pass through the reverse-current coil and the differential voltage coil, thereby closing the current relay contacts. This permits the main contactor coil to be energized; then current flows through the main contactor to the essential DC bus. If current begins to flow through the cutout from the essential DC bus, the main contactor coil is de-energized and the circuit is opened.

24.0.1.17 STARTER RELAYS. When, during emergency generator operation, power is routed through the emergency reverse-current cutout to the essential DC bus, the current must pass through a starter relay corresponding to the selected starter-generator. To complete this path, the starter relay concerned is energized through the corresponding generator transfer relay when the selected starter-generator is switched to emergency service. (Refer to 24.0.1.15.)

24.0.1.18 DC CONTROL PANEL. Most of the switches and indicators for the DC power supply system are on a DC control panel, on the left forward corner of the overhead panel console. This panel contains a loadmeter, a GEN OUT light, a generator control switch, and a guarded FLD TRIP-RESET switch for each of the four starter-generators. A DC voltmeter with a VOLTMETER SELECTOR switch is also included on this panel.

24.0.1.19 GENERATOR CONTROL SWITCHES. Four generator control switches, which have three positions ("ON," "OFF," and "EMER") are on the DC control panel. (See figure 24-2.) If a generator control switch is placed in the "ON" position, the corresponding starter-generator, when operating properly, is connected to the main DC bus through the generator contactor. Also, the equalizer bus relay, in the corresponding generator control panel, is connected to the equalizer bus. If a generator control switch is placed in the "EMER" position, the output of that starter-generator is eligible for selection on an EMER GEN SEL SW for routing to the essential DC bus through the emergency reverse-current cutout. The EMER GEN SEL SW is mounted on the overhead audio control panel.

24.0.1.20 FIELD TRIP-RESET SWITCHES. Four FLD TRIP-RESET switches on the DC control panel provide pilots' compartment control over the field relays in each of the four generator control panels. Besides a normal position, each switch has two momentary contact positions, "FLD TRIP" and "RESET." If a FLD TRIP-RESET switch is placed in the "FLD TRIP" position, the field relay in the corresponding generator control panel is tripped, thereby removing shunt field excitation from the related starter-generator. Also, the equalizer bus relay is de-energized (disconnecting the generator control panel from the equalizer bus), and the generator contactor is opened.

If a field relay has been tripped, the appropriate FLD TRIP-RESET switch must be placed in the "RESET" position to restore the corresponding starter-generator to operation. This closes the field relay and permits normal operation to resume, provided the starter-generator is functioning properly.

24.0.1.21 DC GENERATOR OUT LIGHTS. If a generator contactor disconnects a starter-generator from the main DC bus during normal operation, a corresponding GEN OUT light on the DC control panel goes on. When the generator contactor opens, the corresponding generator out relay is de-energized, thereby grounding the GEN OUT light. Similarly, if a generator transfer relay de-energizes during emergency generator operation, thereby causing the affected starter-generator to be disconnected from the essential DC bus, the corresponding generator out relay is de-energized also. This provides a ground to the related GEN OUT light and as it goes on, a GENERATOR OUT module on the annunciator panel also goes on. Power for the four GEN OUT lights is furnished by the essential DC bus, through a GEN OUT LT circuit breaker.

24.01.22 DC GENERATOR LOADMETERS. (See figure 24-11.) The percent of full load current output (300 amperes) for each starter-generator is continuously indicated on the four loadmeters on the DC control panel. Each loadmeter is connected in parallel with the corresponding shunt. These shunts are mounted in the power shield and are linked to the input terminals of their generator contactors. Two 1-ampere fuses in the power shield protect each of the four loadmeter circuits. The loadmeter for each starter-generator is open until its GEN OUT relay is energized. This is to prevent

24.0.1.22 (Cont).

damage to the loadmeters when high currents flow during the start cycle.

24.0.1.23 DC VOLTMETER. Measurement of voltage output of any of the four starter-generators, or of the batteries in parallel, can be obtained on a DC voltmeter, which is mounted in the DC control panel (see figure 24-2) along with a VOLTMETER SELECTOR switch which permits the selection of the voltage to be indicated. The two 1-ampere fuses in each of the loadmeter shunt circuits in the power shield also protect the voltmeter circuits.

24.0.1.24 EMERGENCY GENERATOR SELECTOR SWITCH. (See figure 24-2.) If it is necessary to route DC power directly to the essential DC bus, thus bypassing the main DC bus, an EMER GEN SEL SW is provided on the overhead audio control panel. This switch allows the selection of any one of the four starter-generators for the essential DC bus. To effect the transfer, the generator transfer-relay and the starter relay for the selected starter-generator are energized when the corresponding generator control switch is placed in the "EMER" position.

24.0.1.25 GENERATOR OVERHEAT WARNING. (Refer to Chapter 33.) If any of the starter-generators overheats, a corresponding GEN OVERHEAT module on the annunciator warning panel goes on, along with the two master caution lights. When the critical temperature is exceeded, an overheat detector thermoswitch inside the case of the affected starter-generator closes. A ground is thus provided to the warning circuit which turns the appropriate GEN OVERHEAT module on. The starter-generator, however, continues to run, since this is merely a warning device. Consequently, the appropriate FLD TRIP-RESET switch should momentarily be placed in "FLD TRIP" position, and the corresponding generator control switch should be placed in the "OFF" position. This permits the overheated starter-generator to cool, thereby reducing the possibility of fire or further damage to the starter-generator.

24.0.1.26 BATTERY SYSTEM. (See figures 24-2 and 24-9.) Two batteries, a BATTERY switch, a battery relay, two battery relay rectifiers, and a battery series relay are included in the battery system. Connected in series by the battery series relay, these batteries supply power for starting the engines. The batteries can be charged with power from the essential DC bus when it is energized. If starter-generator power or external power is not available, battery power can supply the essential DC bus requirements.

24.0.1.27 BATTERIES. Two 24-volt, 36 ampere-hour, nickel-cadmium batteries, under the cabin baggage compartment, chemically store electrical power for engine starting, or for inflight emergency use or ground servicing requirements. These batteries are also capable of discharging a high current, such as demanded during an engine start. For battery maintenance refer to Section 12.5.

24.0.1.28 BATTERY SWITCH. The battery switch is located on the overhead starting control panel. When this switch is placed in the "ON" position, a ground is applied to the battery relay, thereby connecting the batteries to the essential DC bus.

24.0.1.29 BATTERY RELAY. Current from either the essential DC bus or the batteries can energize the battery relay when the relay is grounded by the BATTERY switch. If there is no power on the essential DC bus, battery current can be used to energize the battery relay. If the batteries are discharged, current from the essential DC bus can be used. When closed, the battery relay connects the batteries with the essential DC bus. If 28-volt DC is available on the essential DC bus, the batteries are charged according to their need. If no voltage is available on the essential DC bus, the batteries supply 24-volt DC to the bus.

24.0.1.30 BATTERY RELAY RECTIFIERS. Since circuits from both the essential DC bus and the batteries connect to the battery relay coil, two battery relay rectifiers are necessary to prevent these circuits from creating a closed circuit between the bus and the batteries.

24.0.1.31 DC POWER DISTRIBUTION. (See figures 24-3 and 24-10.) Normally, the output of all four starter-generators is routed through the four generator contactors to the main DC bus in the electrical power shield. The main DC bus in turn supplies the essential DC bus through the normal reverse current cutout. Electrical power from both buses is distributed through current limiters to the circuit breaker panels and directly to certain loads.

24.0.1.32 MAIN DC BUS. 28-volt, DC power from the main DC bus is supplied through 11 current limiters in the electrical power shield to the pilot's wainscot, the copilot's wainscot, and the copilot's side circuit breaker panels. Main DC bus power is also supplied through four current limiters in the electrical power shield to the galley, the normal stabilization loads, the windshield and the No. 1 main inverters.

24.0.1.33 ESSENTIAL DC BUS. 28-volt, DC power from the essential DC bus is supplied through four current limiters in the electrical power shield to the pilot's upper, the pilot's and copilot's side, and the pilot's and copilot's wainscot circuit breaker panels. Essential DC bus power is also supplied through three current limiters in the electrical power shield to the auxiliary hydraulic pump, the emergency stabilization loads, and an inverter. The vestibule lights are supplied through a circuit breaker in the electrical power shield.

24.0.1.34 IGNITION BUS. To maintain a certain minimum voltage for igniter circuits during a battery start, an ignition bus is provided. The ignition bus is usually fed by the essential DC bus, but is supplied from the starter bus when an ignition bus transfer relay is energized.

24.0.2 OPERATIONAL CHECKOUT.

a. BUS VOLTAGE SETTING. The generator output voltage, as monitored with a precision voltmeter at no load, at the JR1068 generator control panel voltage jacks, with a generator switch in the "OFF" position, should be 28 (+0.5 -0.0) volts. If the voltage is not within this tolerance, the voltage should be adjusted according to the following procedure.

(1) Start all four engines, set at IDLE and put generators on the line.

(2) Allow at least 15 minutes running time before attempting to set the bus voltage. This is to allow the carbon pile regulator temperature to stabilize.

(3) Remove from the bus the generator which is to be adjusted, by placing the generator switch in the "OFF" position.

(4) Adjust the voltage for each generator with the voltage adjustment screw on the JR1068 generator control panel until the output voltage for the generator being adjusted is 28 (+0.5, -0.0) volts. A precision voltmeter is required for this check.

Note

The cockpit meter should not be used for this setting; the dial graduations, indicating tolerances, and the fact that the needle setting can be externally adjusted makes it unsatisfactory for setting the generator output voltages.

When the bus output voltage has to be adjusted according to the above procedure, the cockpit meter should be externally adjusted to indicate 28 volts.

b. Start all four engines according to the prescribed procedure in the FAA-Approved Flight Manual.

c. Make sure the following circuit breakers are closed: GEN OUT LT, GENERATOR CONT NO. 1, 2, 3, and 4, START, ENG IGNITER PWR NO. 1, 2, 3, and 4, GEN TRIP & RESET, BUS TIE RELAYS, NORM & EMER, and BAT SERIES PWR.

d. Place the VOLTMETER SELECTOR switch in the "GEN 1" position. Trip the No. 1 generator FLD TRIP-RESET switch by momentarily holding the switch in the "FLD TRIP" position.
RESULT: The NO. 1 GEN OUT light should go on, indicating that starter-generator No. 1 is disconnected from the main DC bus, also voltage should decrease to something less than 5 volts.

e. Reset the No. 1 generator FLD TRIP-RESET switch.
RESULT: The DC voltmeter should read 28 (±1) volts and the NO. 1 GEN OUT light should go out.

f. Place the No. 1 generator control switch in the "EMER" position and the EMER GEN SEL SW in the "GEN 1" position. Then place the VOLTMETER SELECTOR switch in the "BAT" position.
RESULT: The DC voltmeter should read 28 (±1) volts.

g. Place the No. 1 generator control switch in the "ON" position and the EMER GEN SW in the "NORM" position.
RESULT: The generator out light should go out.

h. Repeat steps (d) through (g) for starter-generators 2, 3, and 4.
RESULT: The corresponding results should be obtained.

i. With external power off, place all four generator control switches in the "ON" position. Apply a load of 600 amperes by airplane loads or a loadbank to the main DC bus.
RESULT: Each of the four generator loadmeters should indicate between 45 and 55 percent.

j. With all four generators connected to the main DC bus, to which approximately 600 amperes is being supplied, select any one of the four "GEN" positions on the VOLTMETER SELECTOR switch.
RESULT: The DC voltmeter should read 28 (±1) volts.

k. Trip starter-generators 1 and 2.
RESULT: The generator loadmeters for starter-generators 3 and 4 should indicate between 95 and 105 percent.

CAUTION

Owing to insufficient cooling during ground operation, do not apply a 600-ampere load to two starter-generators for more than 2 minutes.

l. Successively place the VOLTMETER SELECTOR switch in each of the four generator positions.
RESULT: A voltage of 28 (±1) volts DC should be indicated for each starter-generator (engines No. 3 and No. 4 only). Not more than five volts should be indicated for the starter-generators of engines No. 1 and No. 2.

m. Repeat steps (k) and (l), but trip generators 3 and 4 instead.
RESULT: The corresponding results should be obtained.

n. Remove the loadbank from the main DC bus. Place all four generator control switches in the "ON" position, and select any one of the four "GEN" positions on the VOLTMETER SELECTOR switch.
RESULT: The DC voltmeter should read 28 (±1) volts.

24.0.2 (Cont).

o. Trip all four starter-generators, stop all the engines, and restore the airplane to normal.

p. Check for "Normal Reverse Current Cutout" (operation of engines not required):

(1) Connect a light or voltmeter to the "IND" terminal of the Reverse Current relay.

(2) Check to insure all power circuit breakers are in and there are essential and main bus loads.

(3) Connect external power to the aircraft service receptacle.
RESULT: The light should illuminate or voltmeter should indicate voltage.

(4) Battery switch "ON."

(5) Decrease external power supply voltage slowly until it reaches a value less than 24 V DC.
RESULT: The reverse current relay should open as indicated by the light extinguishing or voltmeter reading zero.

q. Check for emergency reverse current cutout operation:

(1) Connect a light or voltmeter to the "IND" terminal of emergency reverse current cutout relay.

(2) Check to insure all power system circuit breakers are "in" and there are essential bus loads.

(3) Start all four engines according to the prescribed procedure in the FAA-Approved Flight Manual. (Remove external start and service receptacle power if connected.)

(4) Position No. 1 generator control switch from "ON" to "EMERG" and the emergency selector switch from "NORM" to "GEN 1."
RESULT: Emergency reverse current cutout should close, as indicated by the light illuminating or voltmeter indicating voltage.

Note

Repeat step (4) for engines 2, 3, and 4.

(5) Position battery switch to "ON".

(6) Place engine No. 1 throttle in the "CUTOFF" position.
RESULT: Emergency reverse current cutout should open as indicated by light diminishing or voltmeter reading being zero.

r. Check for operation of differential relay reverse current coil (located in the generator control panel, JR 1068).

(1) Check to insure all power system circuit breakers are "in."

(2) Start all four engines according to the prescribed procedure in the FAA-Approved Flight Manual.

(3) Disconnect external power source, if used.

(4) Leaving No. 1 generator switch in "ON" position, place No. 1 throttle lever in the "CUTOFF" position.
RESULT: No. 1 generator contactor should open as indicated by the generator out light illuminating. If light does not illuminate, engine will start to "motor", indicating malfunction of the differential relay reverse current coil.

(5) Repeat steps (1) through (4) for engines 2, 3, and 4.

s. Check overvoltage and ground fault circuits per paragraphs 24.0.1.5 and 24.0.1.7 respectively.

24.0.3 TROUBLESHOOTING.

24.0.3.1 STARTER-GENERATOR.

TROUBLE	PROBABLE CAUSE
No output from DC starter-generator.	Starter-generator defective.
	Fire handle pulled.
	Generator control panel defective.
	Control wiring open.
	Low voltage.

24.0.3.1 STARTER-GENERATOR (Cont).

TROUBLE	PROBABLE CAUSE
Starter-generator will not connect to main DC bus.	Generator contactor defective.
	Generator control panel defective.
	Generator control switch defective.
	Low voltage.
	Wiring open.
Starter-generator output not available on essential DC bus under normal operating conditions.	Normal reverse current cutout defective.
	Norm bus tie relay circuit breaker open.
	Bus tie relay defective.
	Control wiring open.
Starter-generator output not available on essential DC bus under emergency operating conditions.	Emergency reverse current cutout defective.
	Bus tie relay defective.
	EMER GEN SEL SW defective.
	Starter relay defective.
	Generator transfer relay defective.
	Wiring open.
	Emergency bus tie circuit breaker open.

24.0.3.1 STARTER-GENERATOR (Cont).

TROUBLE	PROBABLE CAUSE
Starter-generator will not share load.	Series winding burned.
	Defective control panel.
	Damaged wire insulation on wires from terminal "D of starter-generator to control panel.
	Incorrect voltage setting.
	Poor ground connection between generator ground pad and nacelle structure. **Note** Check for this by using the following procedure: a. With engines running, turn on enough electrical load to total approximately 300 amperes (100% load for one generator). Do not vary this load during the remainder of the test. b. Leaving one generator "ON," turn the battery and other three generators "OFF" and with a 0-5 volt voltmeter, measure the voltage drop between aircraft structure and terminal D of the operating generator (Terminal D is minus). This voltage should be 1-1/2 to 2-1/2 volts. Record reading. c. Repeat step b for each generator. If this ground connection is bad, the generator that will not parallel should reflect a slightly higher voltage drop than the other three generators. Removal of the ground pad and cleaning of the mating surfaces between pad and nacelle will correct this condition.
Batteries will not connect to essential DC bus.	BATTERY switch defective.
	Battery relay defective.
	Battery relay rectifiers defective.
	Wiring open.
	BAT relay fuse defective.
Generator drops off bus when "ENG SEL" switch is placed in the corresponding generator position.	Faulty auxiliary contacts in battery series relay.
	Failed battery series control relay which maintains voltage to energize battery series relay. (The above faults cause the corresponding start control relay, which is located in the generator control panel, to be energized when the engine selector switch is positioned to the corresponding engine. The start control relay, when energized, interrupts the generator contactor circuit.)

24.0.3.2 BATTERIES.

TROUBLE	PROBABLE CAUSE	REMEDY
Loss of capacity (fading). This condition can be detected by sluggish engine starts, charging current returning to trickle more rapidly than normal, or frequant need to add water to the cells. To make a positive check on the battery condition, measure the capacity by discharging the battery into a known load. To do this, remove the connector from the battery not under test, and remove all load from the essential bus. Reconnect a load to the bus, consisting of the No. 2 and No. 3 fuel boost pumps and the No. 2 main inverter. Allow the battery to discharge into this load for 15 minutes while monitoring the battery voltage. The inverter must not be loaded during this test. If the voltage decays below 19 volts within the time interval, refer to remedy.	Electrochemical reaction within the cell when charging is done exclusively by the constant-potential method.	Charge the battery by the constant-current method.
Complete failure to operate.	Faulty electrical connection.	Correct wiring, quick-disconnect, and intercell connections within the battery as necessary.
Excessive spewing, resulting in a large quantity of white crystalline deposit on the top of the cells and on the case cover, and collection of liquid in the bottom of the battery case.	Voltage from aircraft power system or GPU too high during charge. Over-filling with water.	Clean the battery, adjust the liquid level, and check for lost capacity. Check system voltage per 24.0.2 (a).
Distortion of cell cases.	Overcharging.	Replace cells.
Cells appear grey, black, or brown in color, and will not respond to normal charging procedures.	Foreign material in the cell case caused by addition of impure water or water containing acid.	Replace cells.
	Overheat	Replace cells.
Corrosion of top hardware.	Batteries being subjected to corrosive fumes, sprays, or chemical moisture.	Clean the batteries.

24.0.4 MAINTENANCE OF DC ELECTRICAL SYSTEM.

24.0.4.1 REMOVING DC STARTER-GENERATOR.

a. Place all switches to "off" or "airplane-parked" position.

b. Disconnect external power from the airplane.

c. Open appropriate nacelle lower access door.

d. Unclamp and detach the cooling duct at the forward end of the starter-generator.

24.0.4.1 (Cont).

e. Detach all electrical leads to the starter-generator.

f. At the aft end of the starter-generator, remove the four screws and clamp securing the shroud, then slide it forward.

g. Loosen the six nuts holding the starter-generator to the engine accessory drive pad.

CAUTION

The starter-generator should never be suspended by its shaft on installation or removal as its weight (approximately 61 pounds) may damage the shaft. Failure to observe this precaution may result in premature shaft failures.

h. Rotate the starter-generator counterclockwise, and lift and pull free.

24.0.4.2 INSTALLING DC STARTER-GENERATOR. Each time a starter-generator is removed or replaced for any reason, check that the wiring to the two large generator studs has not been reversed inadvertently. A reverse connection of the starter-generator will be noted by the fact that the generator will not come on the line when the engine is started.

CAUTION

Damage to the generator control panel will occur within approximately one minute after an engine is started if the connections are reversed.

a. Initial failure will be in the differential reverse-current relay coil in the generator control panel. The reverse-current relay coil should be checked each time the wiring to the starter-generator is removed for any reason, and the starter-generator will not come on the line during the subsequent engine run-up.

b. Check for a failed reverse current relay coil by removing the generator control panel and measuring the resistance between pins 16 and 21 on the control panel. The resistance should be 5 (+0.5) ohms. A defect is indicated if the resistance is high or infinite. If a defective reverse-current relay is indicated, replace the generator control panel to prevent damage to the starter-generator.

CAUTION

Whenever tightening the nut on the stud of a generator overheat detector sensor, excessive torque must not be applied. The proper limits are 25 to 30 inch-pounds. If the maximum torque limit is exceeded, the sensor can be permanently damaged.

c. Disconnect all external electrical power from airplane.

d. Guide the starter-generator into position on the engine accessory drive pad aligning the splined shaft and slipping over the securing nuts as necessary.

CAUTION

The starter-generator should never be suspended by its shaft on installation or removal as its weight (approximately 61 pounds) may damage the shaft. Failure to observe this precaution may result in premature shaft failure.

e. Rotate the starter-generator clockwise, and tighten the six (6) securing nuts.

f. Reconnect the shroud at the aft end of the starter-generator.

g. Connect the proper electrical lead to the stud of the generator overheat detector sensor and secure with a nut. Apply a torque of 25 to 30 inch-pounds.

h. Reconnect the starter-generator leads. Safety-wire the feeder and ground line nuts separately.

CAUTION

The ground and feeder lines must not be reversed, since damage to the generator control panel and perhaps to the starter-generator will result. Also, the securing nuts must not be safety-wired together, for this would short-circuit the ground and feeder lines.

i. Replace and secure the cooling duct to the forward end of the starter-generator.

j. Close the nacelle lower access door.

24.0.4.3 BRUSH INSPECTION. Each starter-generator has four brush holders, each of which contains a set of four brushes. To inspect for wear, it is necessary to remove the brushes from the holder. A set of brushes should not be used after wear has reached the 0.03-inch-wide groove notched across the outside edges. For a quick inspection of a starter-generator, only one set of brushes need be checked; comparable wear is then assumed for the other three sets of brushes. This quick inspection avoids the necessity of removing the starter-generator from the drive pad, as required for a complete inspection. Determine remaining brush life using the formula in step

24.0.4.3 (Cont).

(c) below. The complete inspection of the brush wear for any of the starter-generators can be accomplished in the following manner:

a. Refer to paragraph 24.0.4.1 for removal of starter-generator.

b. At a convenient workplace, unclamp and remove the commutator inspection cover.

c. Remove and inspect each of the four sets of brushes individually. Excessive wear is indicated when the wear limit mark on the outside edge of any brush is reached or closely approached. This mark is a 0.03-inch-wide groove notched across the edge of the brush.

Note

The remaining brush life can be obtained by measuring the remaining brush length and then calculating the remaining brush life using the following formula:

$$\frac{T}{0.65 - B_L} = \frac{R_B}{B_L}$$

Where: T = Operating time since brushes were installed

R_B = Remaining brush life

B_L = Remaining usable brush length (measured from the short side of the brush to the wear limit mark)

For a new brush the remaining usable brush length is 0.65 inch.

d. Replace the worn brushes with the new ones as necessary; reinstall brushes in all four brush holders.

Note

When replacement brushes are required, replace with Lear-Siegler Part No. 23035-1110 brushes or Lear-Siegler Part No. 303001024 brushes. Lockheed-Georgia Company recommends installation of the Lear-Siegler Part No. 23035-1110 brushes as a means of reducing wear on the commutator.

CAUTION

If the commutator is grooved any discernible amount, new brushes should not be installed without first dressing down the commutator. This is necessary to obtain proper brush seating.

e. When new brushes are installed in a starter-generator, it is necessary to run-in the brushes. The following procedure should be used:

1. Secure the generator so that power can be applied to its terminals.
2. Place a strip of 5/0 sandpaper, slightly wider than the brush area, over the commutator with the sand side out. Overlap the ends of the sandpaper and secure it in position with adhesive tape.
3. Carefully lower the brushes in place against the sandpaper and rotate the armature by hand in the normal direction of rotation until a partial seat has been obtained.
4. Remove the sandpaper and carefully release the brushes against the commutator.
5. Connect terminal "B" through a 0.1 ohm resistor and switch of suitable rating to the positive terminal of a d-c source. Connect terminal "E" to the negative terminal of the d-c source.
6. Connect the "A" terminal to the generator side of the switch through a 10 ohm, 15 ampere rheostat.
7. Rotate the rheostat control to the minimum resistance reading and close the circuit by means of the switch. The generator should start and run as a motor in a clockwise direction, when viewed from the commutator end. If rotation is counterclockwise, check both internal and external connections to the generator.
8. Adjust the rheostat until the generator armature revolves at approximately 4,000 to 4,500 rpm.
9. Allow the generator to operate as a motor until the brushes show a minimum seating of 90 percent in an axial direction and 100 percent in direction of rotation.
10. Remove all carbon dust with filtered compressed air.

f. Replace the commutator inspection cover.

g. Refer to paragraph 24.0.4.2 for installation of starter-generator.

Note

Perform a "quick" inspection of starter-generator brushes every 400 hours.

Perform a "complete" inspection of starter-generator brushes at each engine change period or when a "quick" inspection reveals that the remaining brush life is less than 500 hours.

24.0.4.4 POWER SHIELD FASTENERS. The torque values shown below are recommended in securing bolted connections. The bolts are identified on their heads.

BOLT DESIGNATION	BOLT SIZE	TORQUE RANGE (in Inch-Pounds)
NAS428-4-6	1/4-28	70-100
NAS428-4-10	1/4-28	70-100
NAS428-4-12	1/4-28	70-100
NAS428-6-12	3/8-24	190-390
NAS428-6-14	3/8-24	190-390

24.0.4.5 BATTERY DISCONNECT. The receptacle in the Elcon battery quick disconnect becomes weak in service allowing a resistance to build up. The disconnect should be inspected in accordance with the following procedure.

24.0.4.5.1 EQUIPMENT. The following equipment is required:

a. Inspection gauge No. 029, available from Elcon. (See figure 24-11B.)

b. Dial calipers, capable of reading to 0.001 inch.

24.0.4.5.2 INSPECTION OF QUICK DISCONNECT. Disengage the battery disconnect from the mating receptacle and inspect for the following:

Note

Any evidence found in a, b, or c below or failure to meet d or e shall be cause for rework or replacement. Check fit of replacement part per d and e before installing.

a. Evidence of corrosion or pitting of the helix.

b. Excessive free play in the handwheel worm assembly.

c. Evidence of arcing or burn marks on the helix. This is caused when the disconnect is removed under electrical load.

d. Insert the 0.385 inch diameter end of the inspection gauge into each helix to a depth of 0.437 inches. The fit shall be snug with a force to remove greater than one pound. This is to test the resiliency of the helix to an oversized pin.

e. Insert the 0.370 inch diameter end of the inspection gauge into each helix to a depth of 0.437 inches. The fit shall be snug with a nominal force to remove of one pound. This will ensure proper contact to a worn or undersized pin.

24.0.4.5.3 INSPECTION OF BATTERY RECEPTACLE. With the mating disconnect removed, inspect the receptacle for the following:

Note

Any evidence found in a, b, or c below or failure to meet d shall be cause for rework or replacement. Check fit of replacement part per d before installing.

a. Evidence of corrosion or pitting on the contact pins.

b. Evidence of arcing or burn marks on the contact pins. This is caused when the disconnect is removed under electrical load.

c. Evidence of battery electrolyte leakage through the receptacle and/or the contact pins.

Note

Electrolyte leakage can be detected by a discoloration of the receptacle body with the glass fibers exposed.

d. Diameter of each contact pin using the dial calipers. The diameter shall be 0.375 (±0.005) inches.

24.0.5 PRESERVATION AND DEPRESERVATION.

24.0.5.1 PRESERVATION OF ELECTRICAL EQUIPMENT DURING PROLONGED INACTIVITY. If an airplane is to be inactive for a prolonged period, observe the following precautions to protect the electrical equipment from deterioration and keep it clean.

a. Where possible, cover the electrical equipment with grease-proof paper, MIL-B-121 Grade A, Class 2, Type II (Sherman Paper Prod. Corp., "V-29"). Enclose a desiccant within the paper covering to absorb moisture and keep the equipment dry.

b. Enclose all disconnected electrical fittings and connectors in polyethylene bags and tie securely.

Note

Make a small hole in the bottom of the bag to allow moisture to escape.

c. All electrical equipment not properly covered should be operated 30 minutes every 30 days, but the starter-generators should not be operated.

d. Keep the batteries in the airplane and maintain them at the fully charged condition.

e. Neutralize any spilled battery electrolyte with either a 2 or a 3 percent solution of boric acid.

f. Inspect the electrical equipment once a week. If corrosion is found, remove the corrosion and operate the equipment until it is thoroughly dry.

24.0.5.2 DEPRESERVATION OF ELECTRICAL EQUIPMENT AFTER PROLONGED INACTIVITY. Before returning the electrical equipment to service, remove all protective covers, remove all corrosion, reconnect all connections, and remove temporary desiccants. Components with permanently provided desiccants should be checked, and desiccant replenished, if necessary.

Page 24-17/24-18 Deleted.

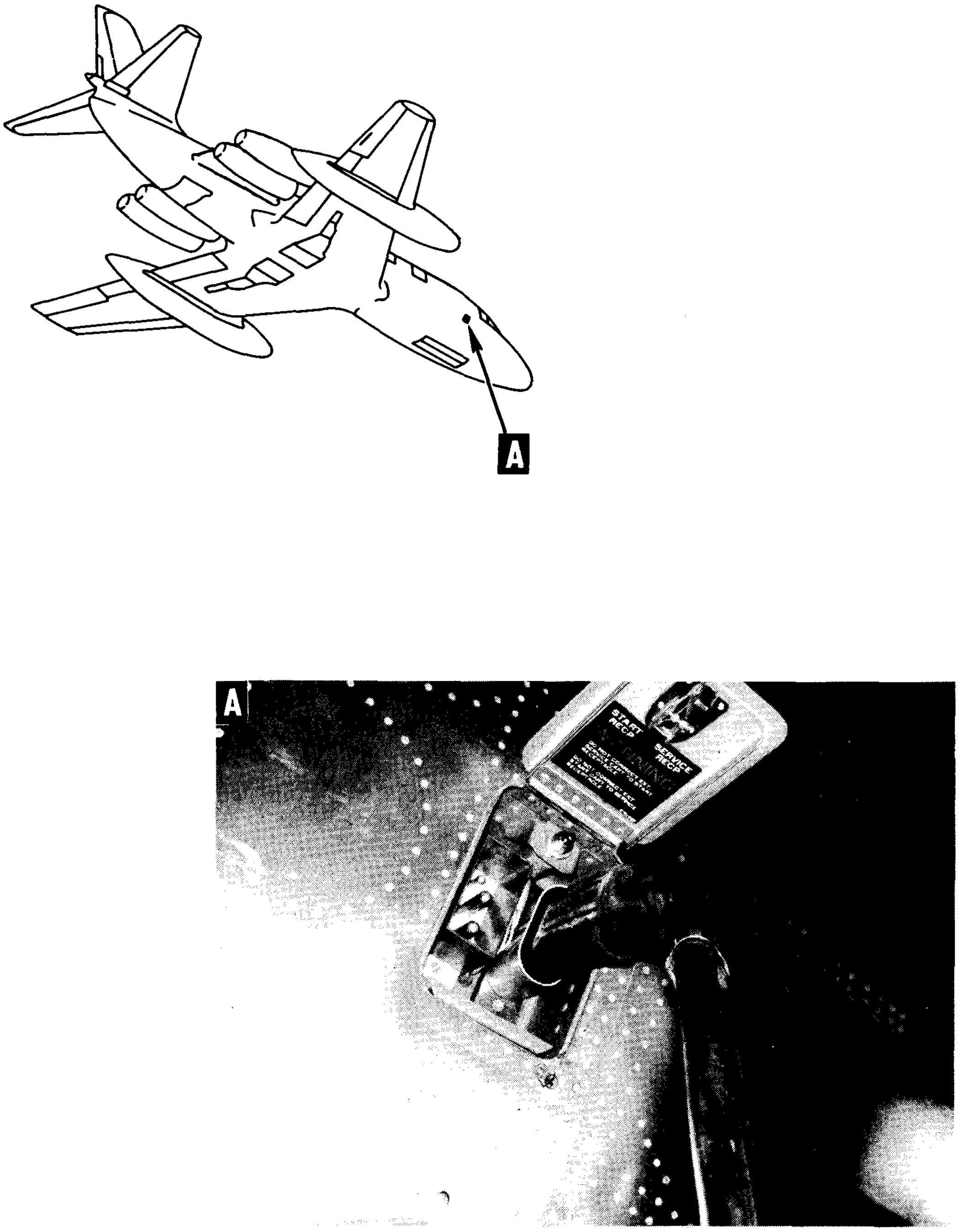
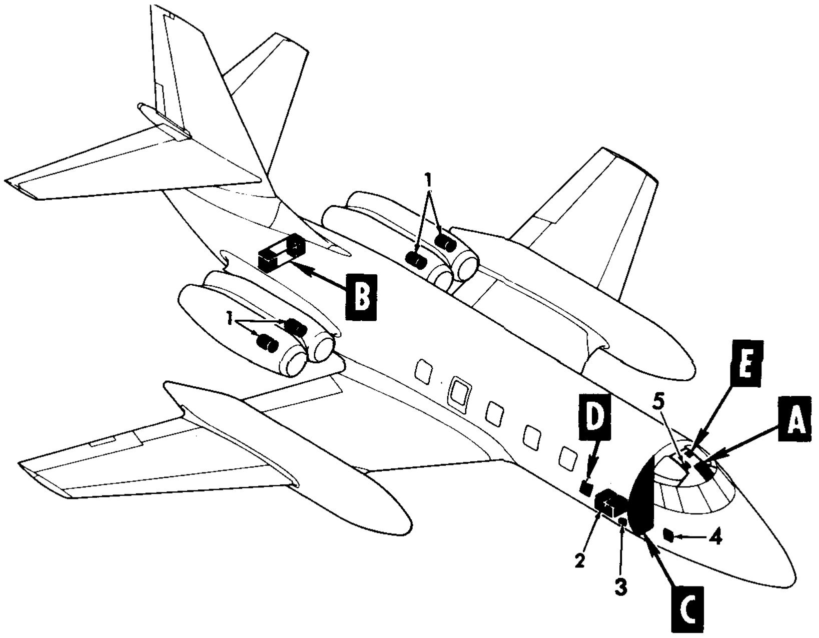
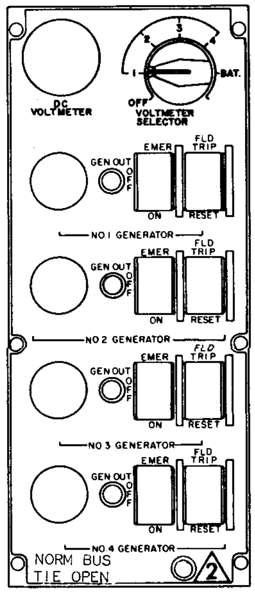


Figure 24-1. External Power Connection





A **DC CONTROL PANEL**
(LOOKING UP)

1. STARTER GENERATORS
2. BATTERIES
3. EXTERNAL POWER RECEPTACLES
4. TOUCH DOWN RELAY
5. BUS-TIE SWITCH
6. NO. 4 GENERATOR CONTROL PANEL
7. NO. 3 & NO. 4 OVERVOLTAGE LOCKOUT RELAYS
8. NO. 2 OVERVOLTAGE LOCKOUT RELAY
9. NO. 1 OVERVOLTAGE LOCKOUT RELAY
10. NO. 2 GENERATOR CONTROL PANEL
11. NO. 1 GENERATOR CONTROL PANEL
12. OVERVOLTAGE-GROUND FAULT TEST SWITCHES
13. OVERVOLTAGE LOCKOUT DISCONNECT RELAY
14. NO. 3 GENERATOR CONTROL PANEL

△1 15. NO. 3 & NO. 4 OVERVOLTAGE REGULATOR LOCKOUT RELAYS

△1 16. NO. 1 & NO. 2 OVERVOLTAGE REGULATOR LOCKOUT RELAYS

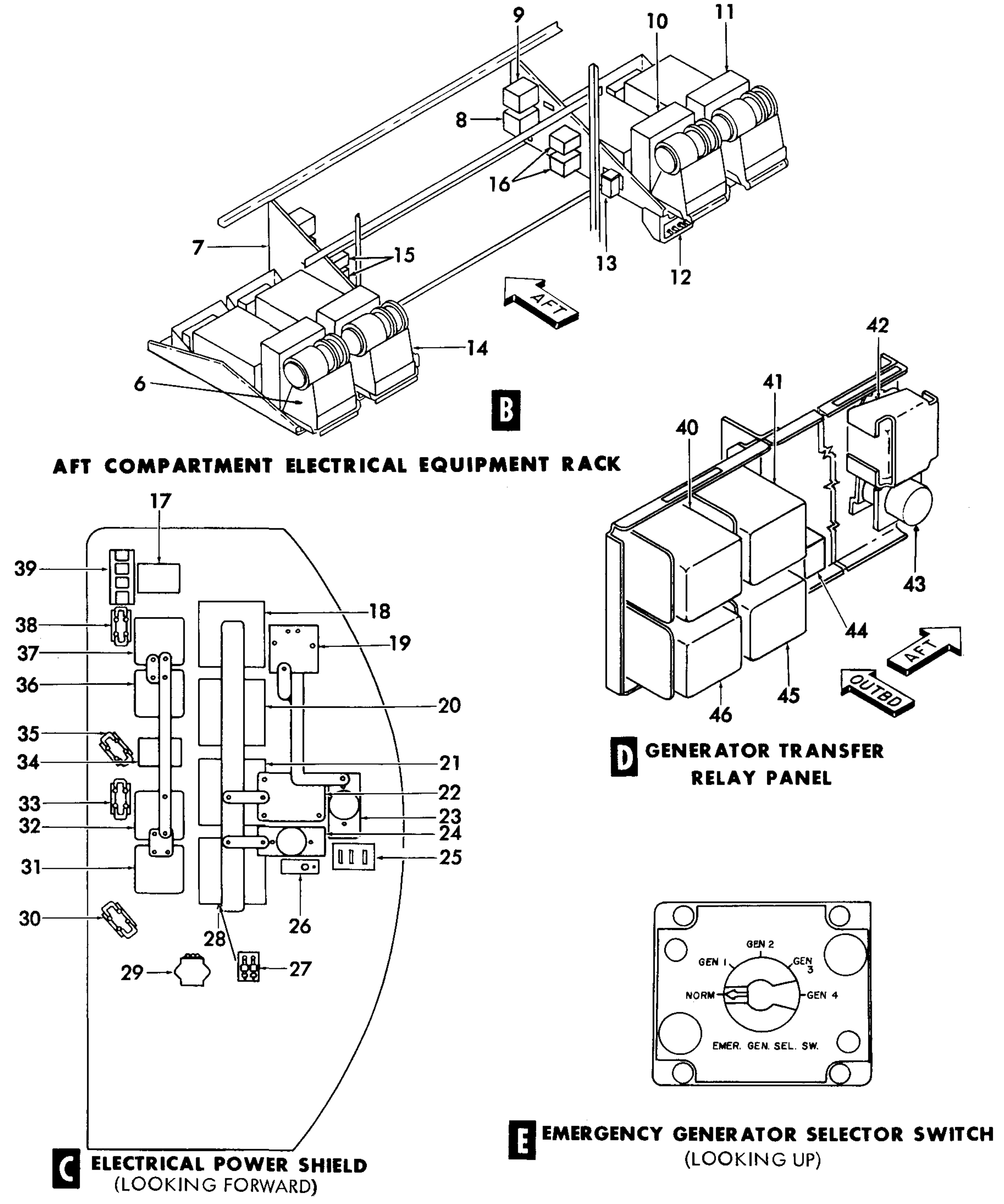
17. LOADMETER FUSES, NOS. 3 & 4
18. NO. 4 GENERATOR CONTACTOR
19. EMERGENCY REVERSE CURRENT CUTOUT
20. NO. 3 GENERATOR CONTACTOR
21. NO. 2 GENERATOR CONTACTOR
22. NORMAL REVERSE CURRENT CUTOUT
23. BATTERY RELAY
24. EXTERNAL POWER RELAY
25. BATTERY VOLTMETER FUSE
26. VESTIBULE LIGHT CIRCUIT BREAKER
27. BATTERY RELAY RECTIFIERS, NOS. 1 & 2
28. NO. 1 GENERATOR CONTACTOR
29. BATTERY SERIES CONTROL RELAY
30. NO. 1 GROUND FAULT TRANSFORMER
31. NO. 1 STARTER RELAY
32. NO. 2 STARTER RELAY
33. NO. 2 GROUND FAULT TRANSFORMER
34. LOADMETER FUSES, NOS. 1 & 2
35. NO. 3 GROUND FAULT TRANSFORMER
36. NO. 3 STARTER RELAY
37. NO. 4 STARTER RELAY
38. NO. 4 GROUND FAULT TRANSFORMER
39. GENERATOR OUT RELAYS
40. NO. 1 GENERATOR TRANSFER RELAY
41. NO. 3 GENERATOR TRANSFER RELAY
42. BUS TIE TRANSFER RELAY
43. IGNITION BUS TRANSFER RELAY
44. BUS-TIE WARNING RELAY
45. NO. 4 GENERATOR TRANSFER RELAY
46. NO. 2 GENERATOR TRANSFER RELAY

NOTE

△1 AIRPLANES 5001, 5002, 5004 & 5006.

△2 AIRPLANES 5059 THRU 5085

Figure 24-2. DC Power Supply Components (Sheet 1 of 2)



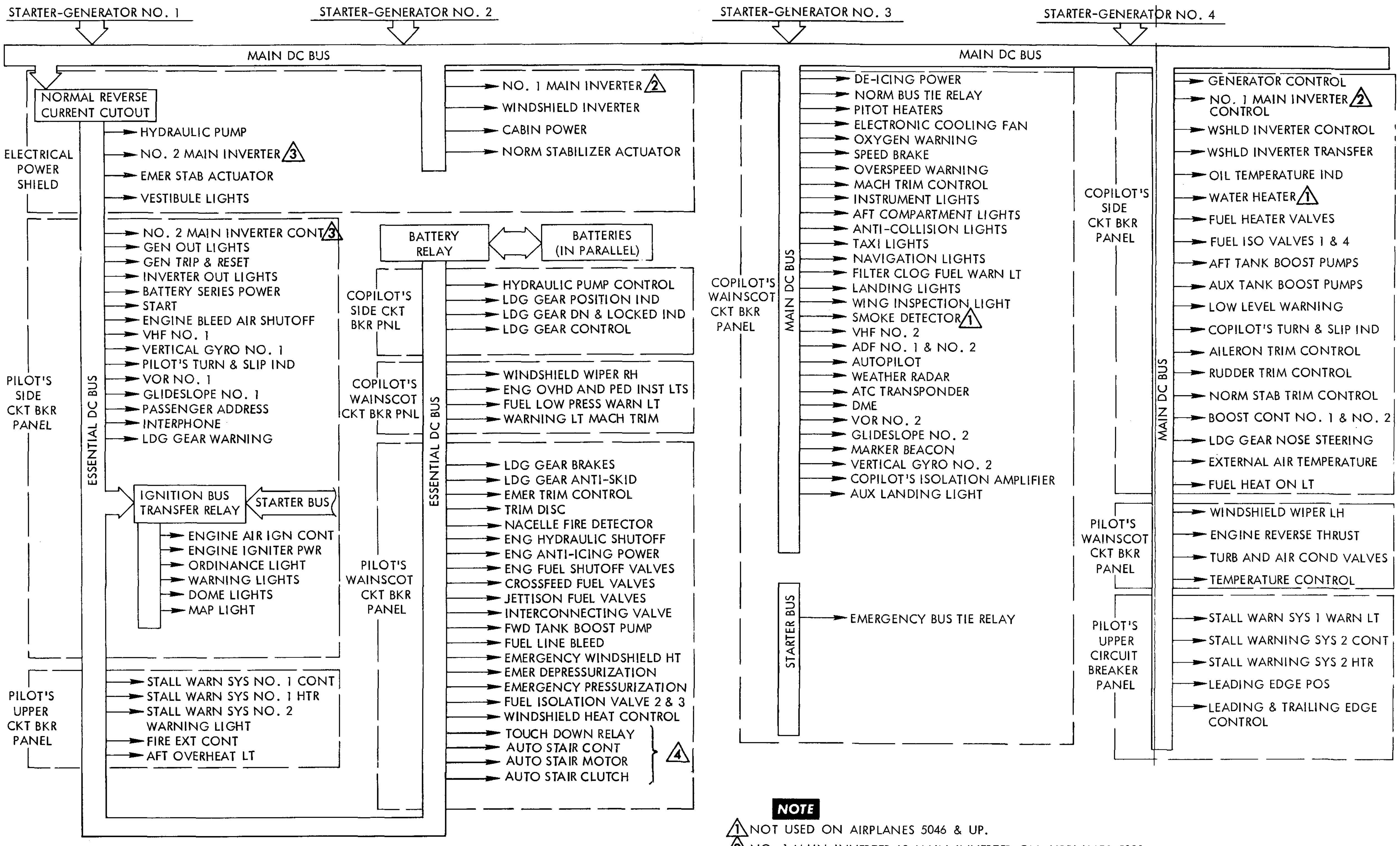
AFT COMPARTMENT ELECTRICAL EQUIPMENT RACK

GENERATOR TRANSFER RELAY PANEL

EMERGENCY GENERATOR SELECTOR SWITCH (LOOKING UP)

ELECTRICAL POWER SHIELD (LOOKING FORWARD)

Figure 24-2. DC Power Supply Components (Sheet 2 of 2)



NOTE

△1 NOT USED ON AIRPLANES 5046 & UP.

△2 NO. 1 MAIN INVERTER IS MAIN INVERTER ON AIRPLANES 5001, 5002, 5004, 5006 - 5009.

△3 NO. 2 MAIN INVERTER IS STANDBY INVERTER ON AIRPLANES 5001, 5002, 5004, 5006 - 5009.

△4 ON AIRPLANES 5066 AND UP

Figure 24-3. DC Power Distribution Diagram

Changed 1 December 1965

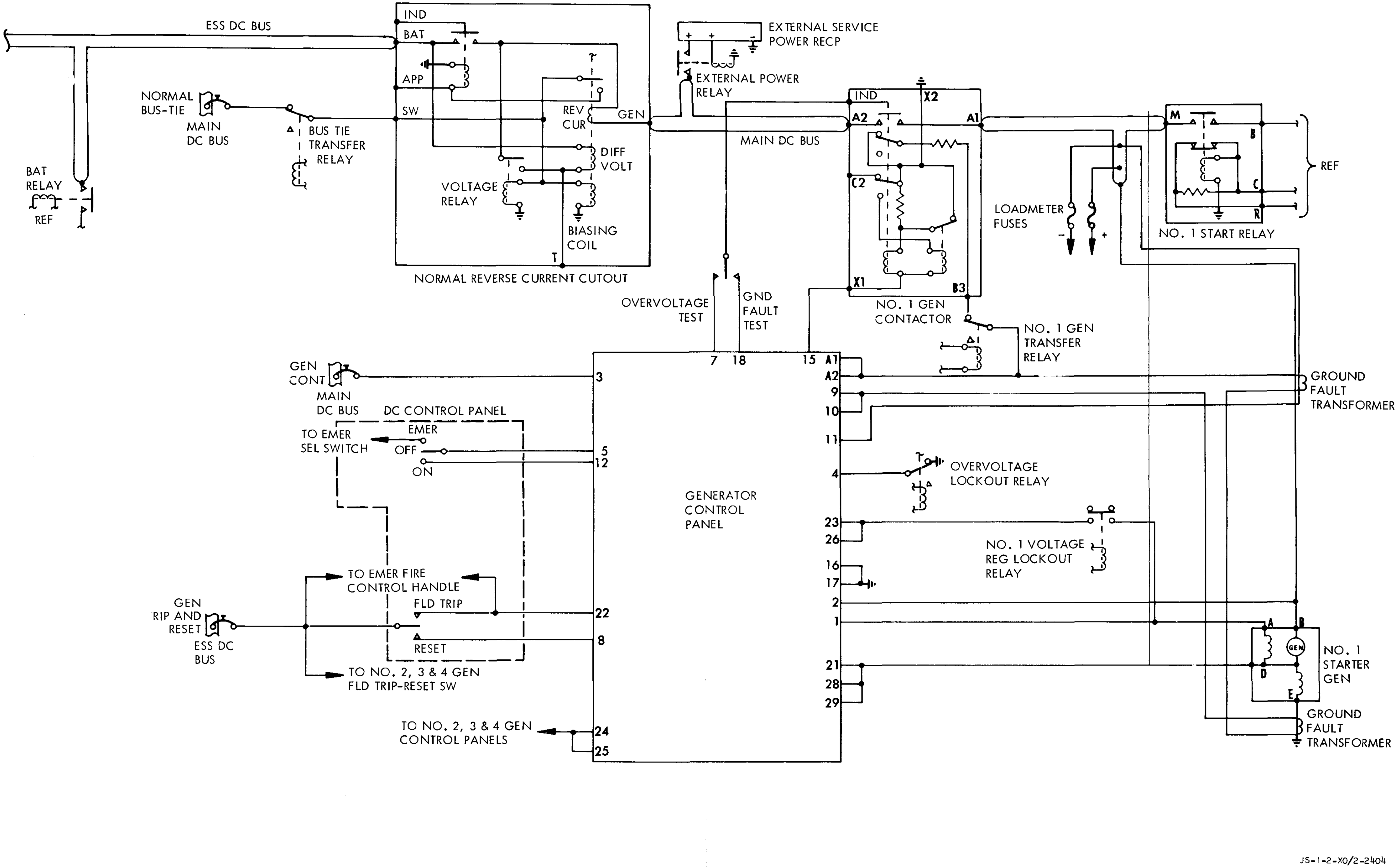
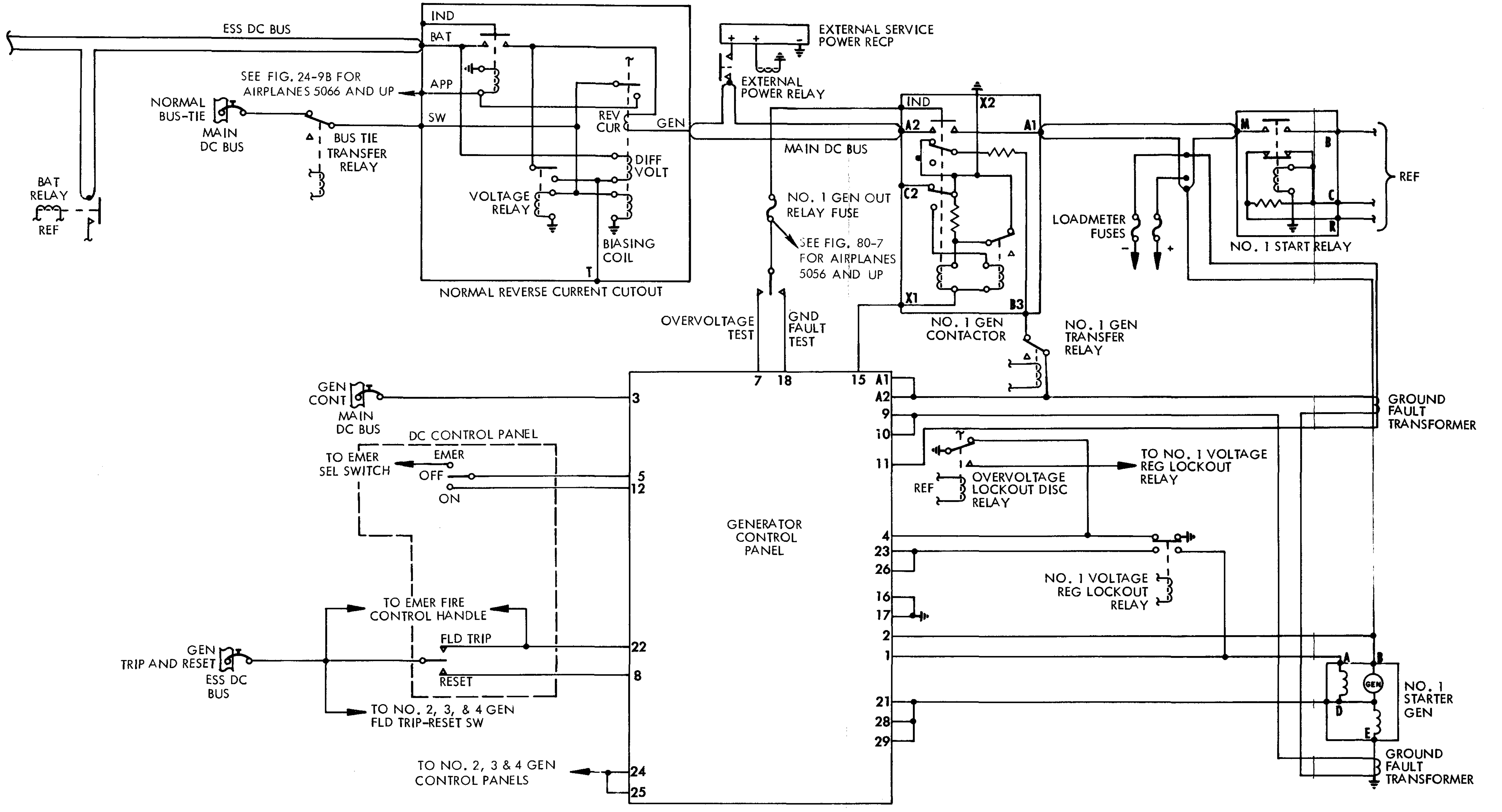


Figure 24-4. Normal DC Power Schematic Diagram (Airplanes 5001, 5002, 5004, and 5006)



JS-1-2-X0/2-2405

Figure 24-5. Normal DC Power Schematic Diagram (Airplanes 5003, 5005, 5007 and Up)

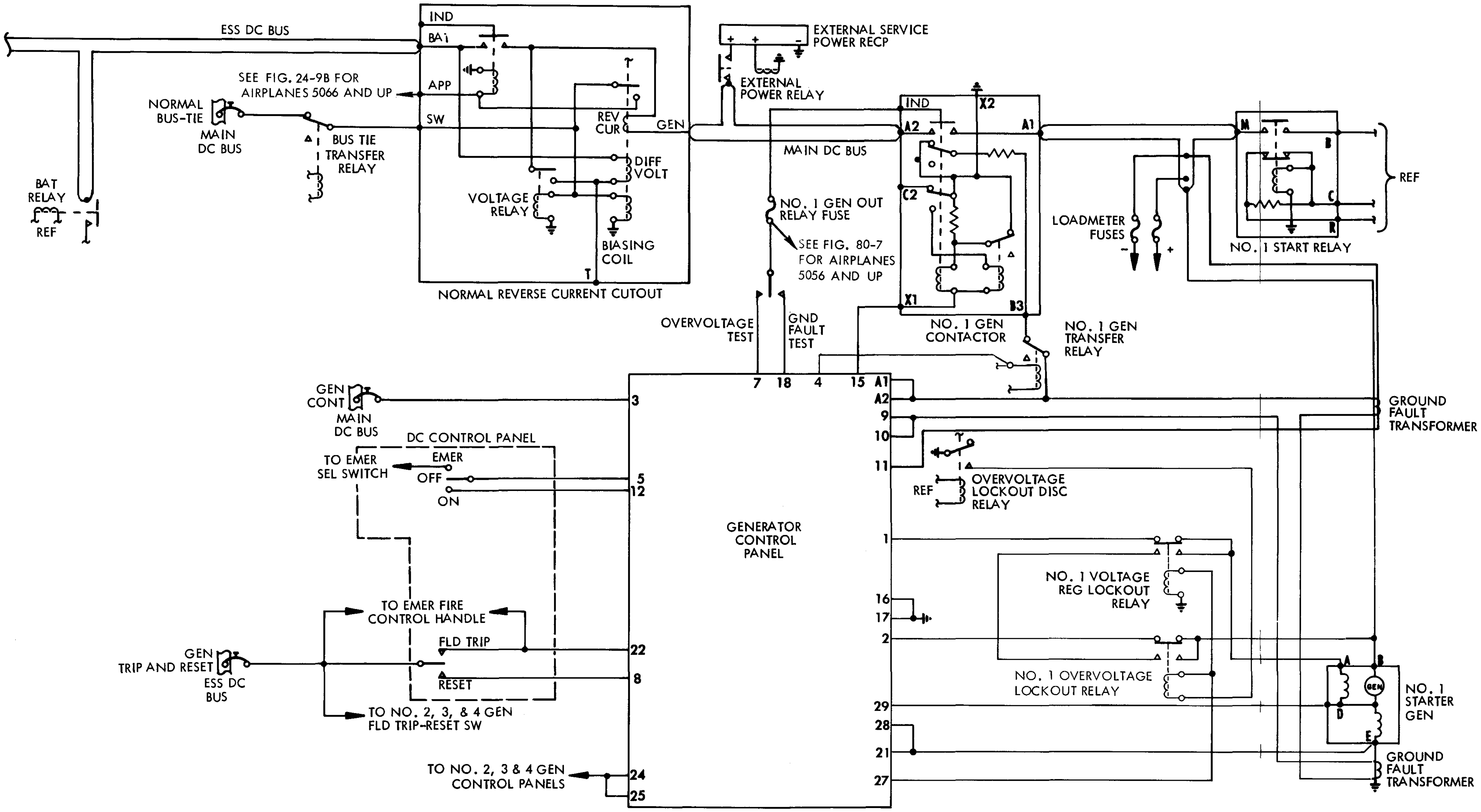


Figure 24-5A. Normal DC Power Schematic Diagram
(Airplanes Modified by Service Bulletin 329-294)

JS-1-2-X0/2-2405A

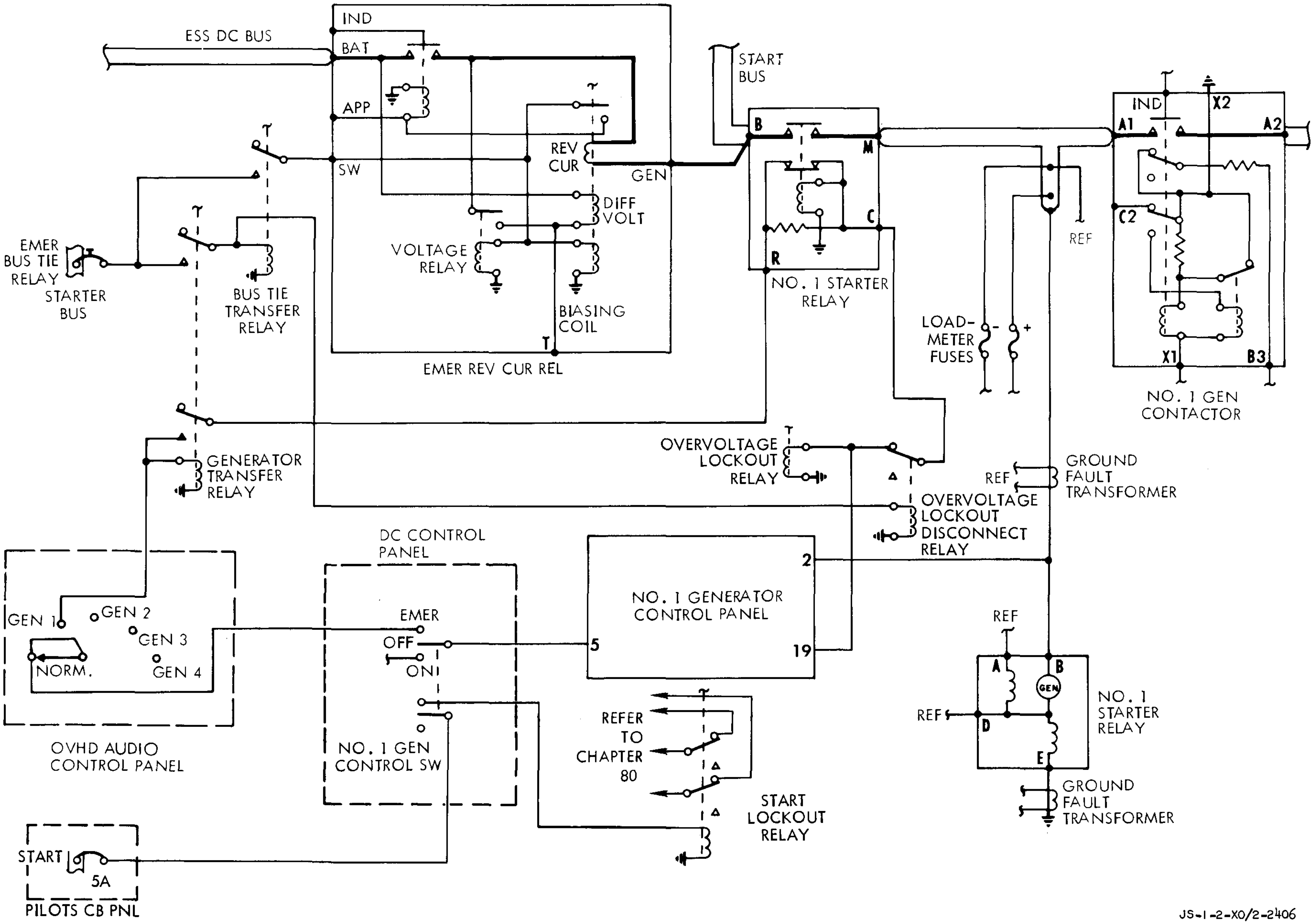


Figure 24-6. Emergency DC Power Schematic Diagram (Airplanes 5001, 5002, 5004 and 5006)

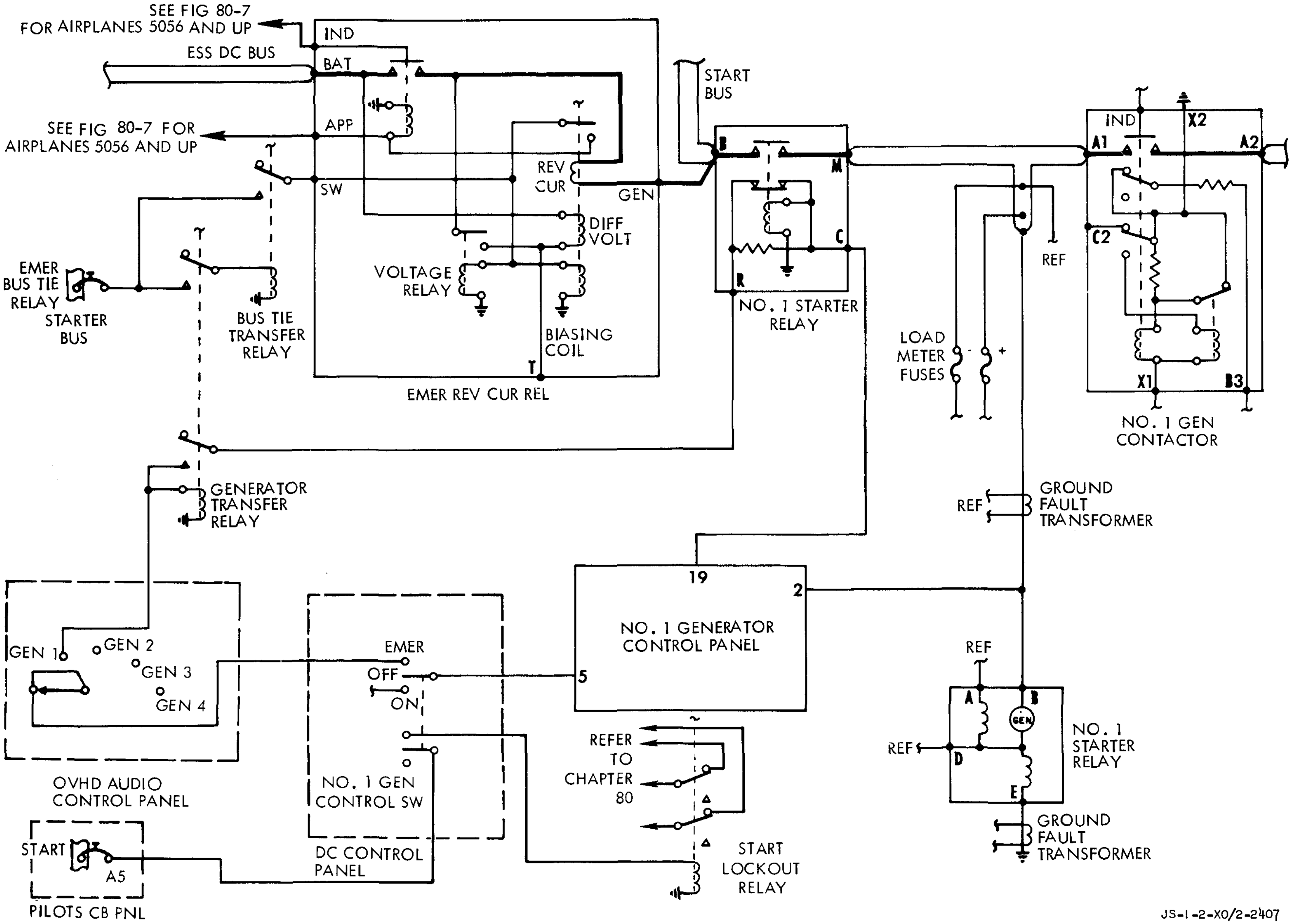


Figure 24-7. Emergency DC Power Schematic Diagram (Airplanes 5003, 5005, 5007 and Up)

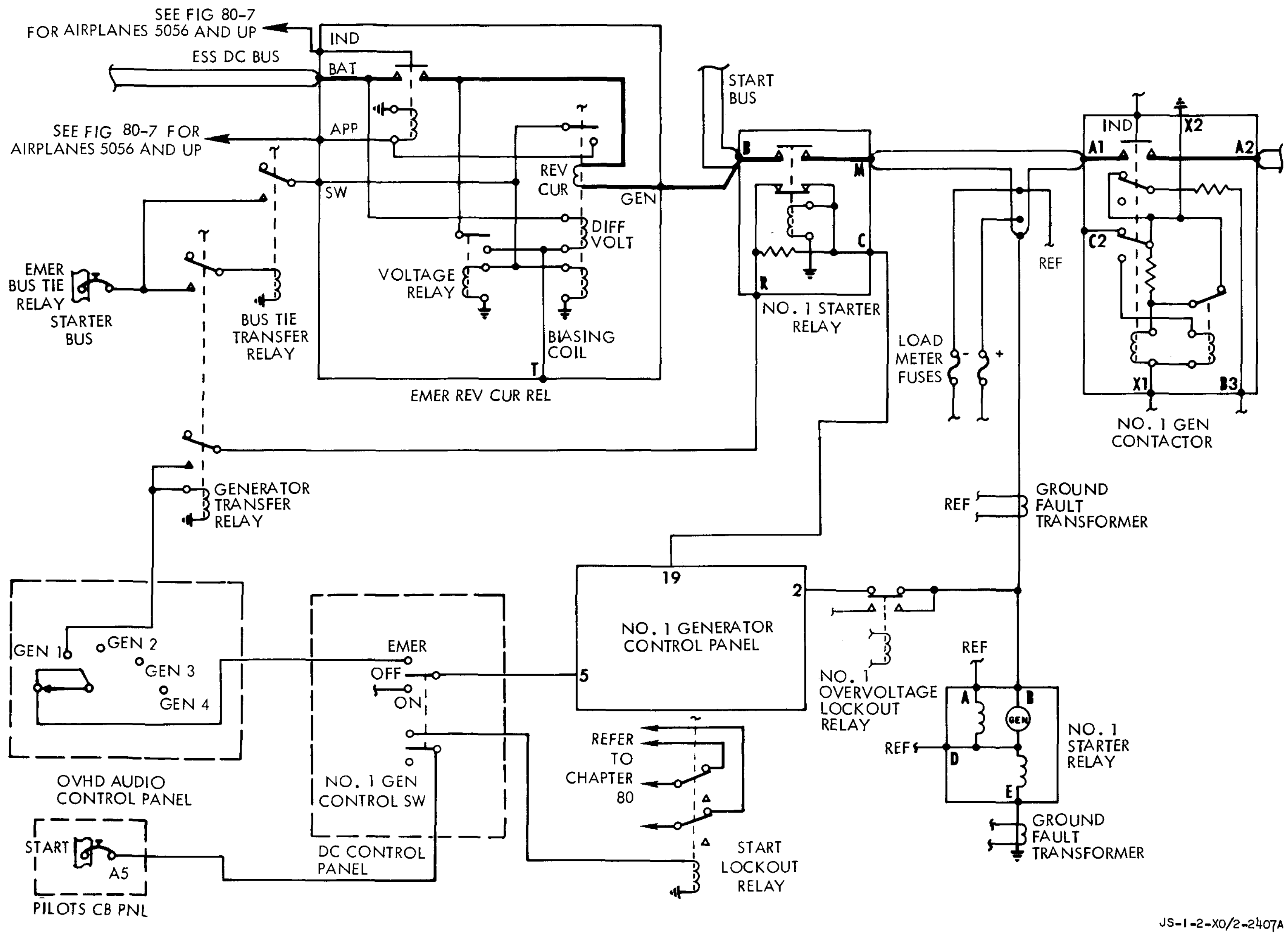
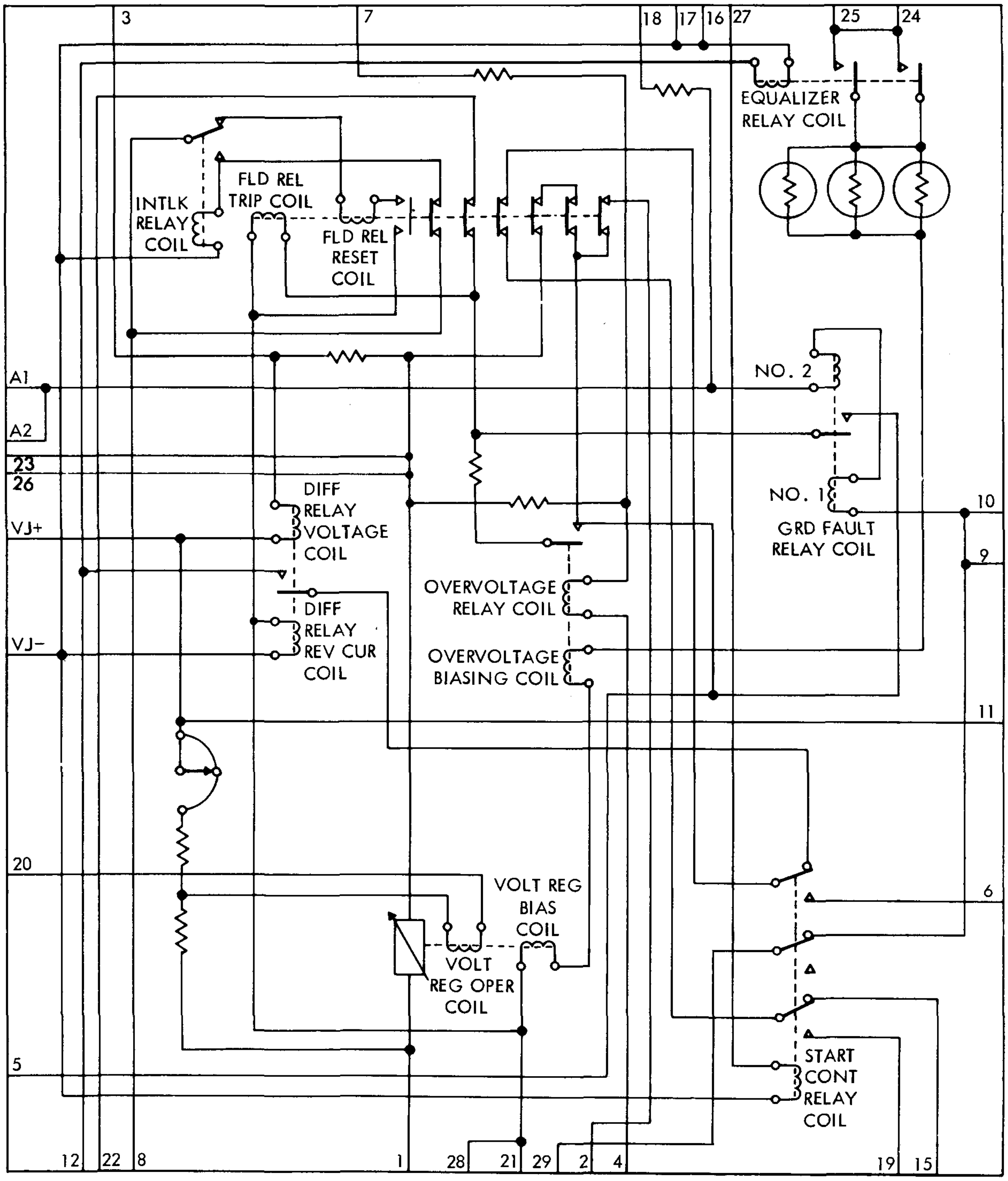


Figure 24-7A. Emergency DC Power Schematic Diagram (Airplanes Modified by Service Bulletin 329-294)



JS-1-2-X0/2-2408

Figure 24-8. Generator Control Panel Schematic Diagram (Airplanes not Modified by Service Bulletin 329-294)

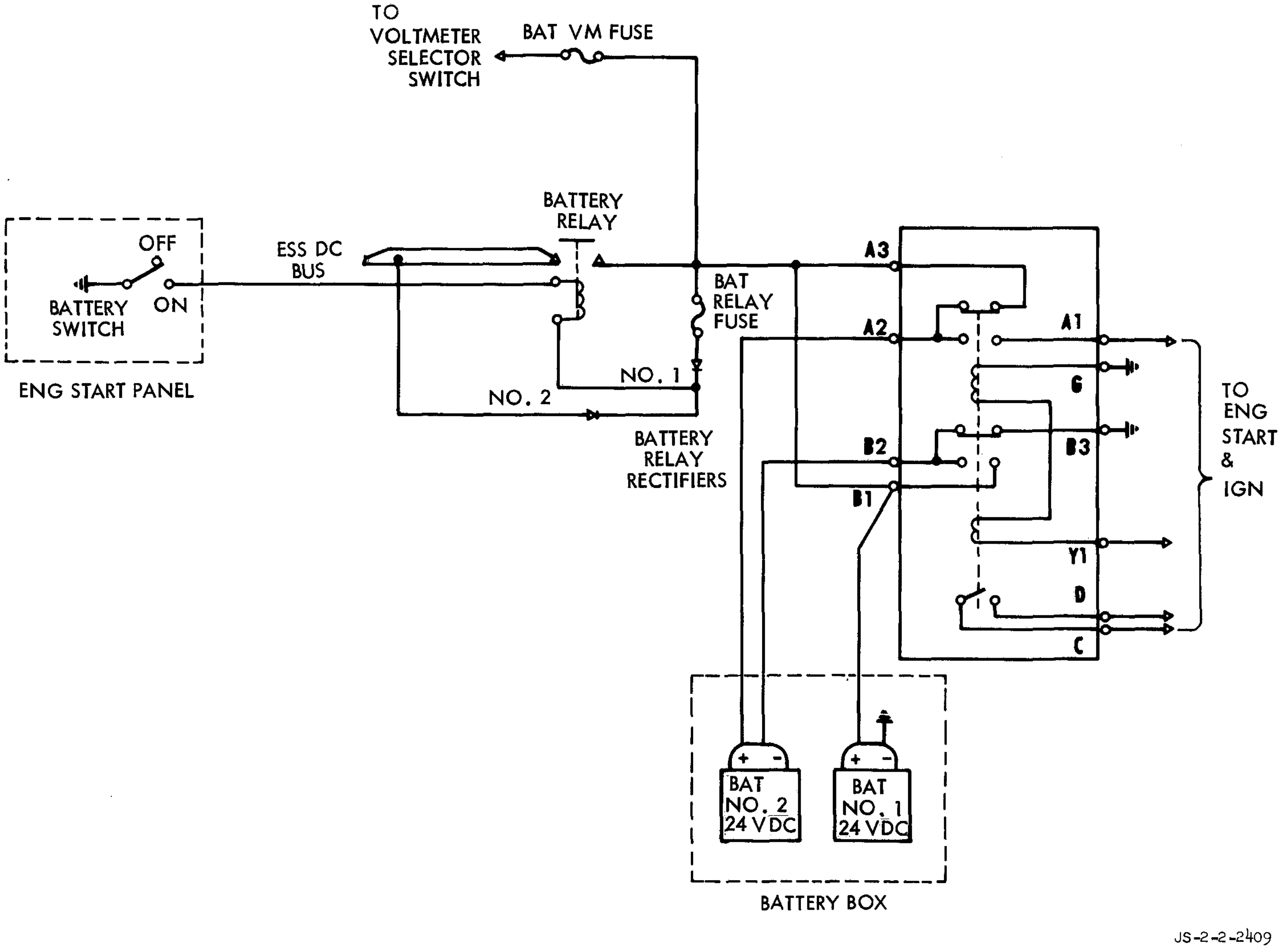
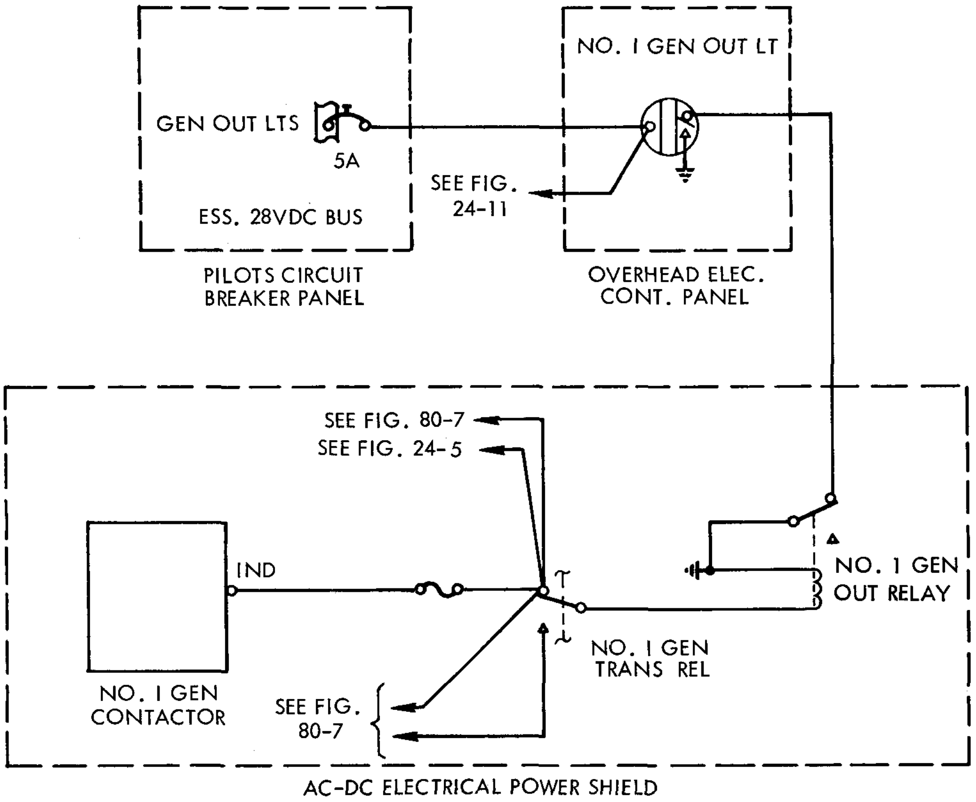


Figure 24-9. Battery System Schematic Diagram



NOTE

CIRCUIT SHOWN IS SAME
FOR GEN 2, 3, AND 4

Figure 24-9A. Generator Out Lights

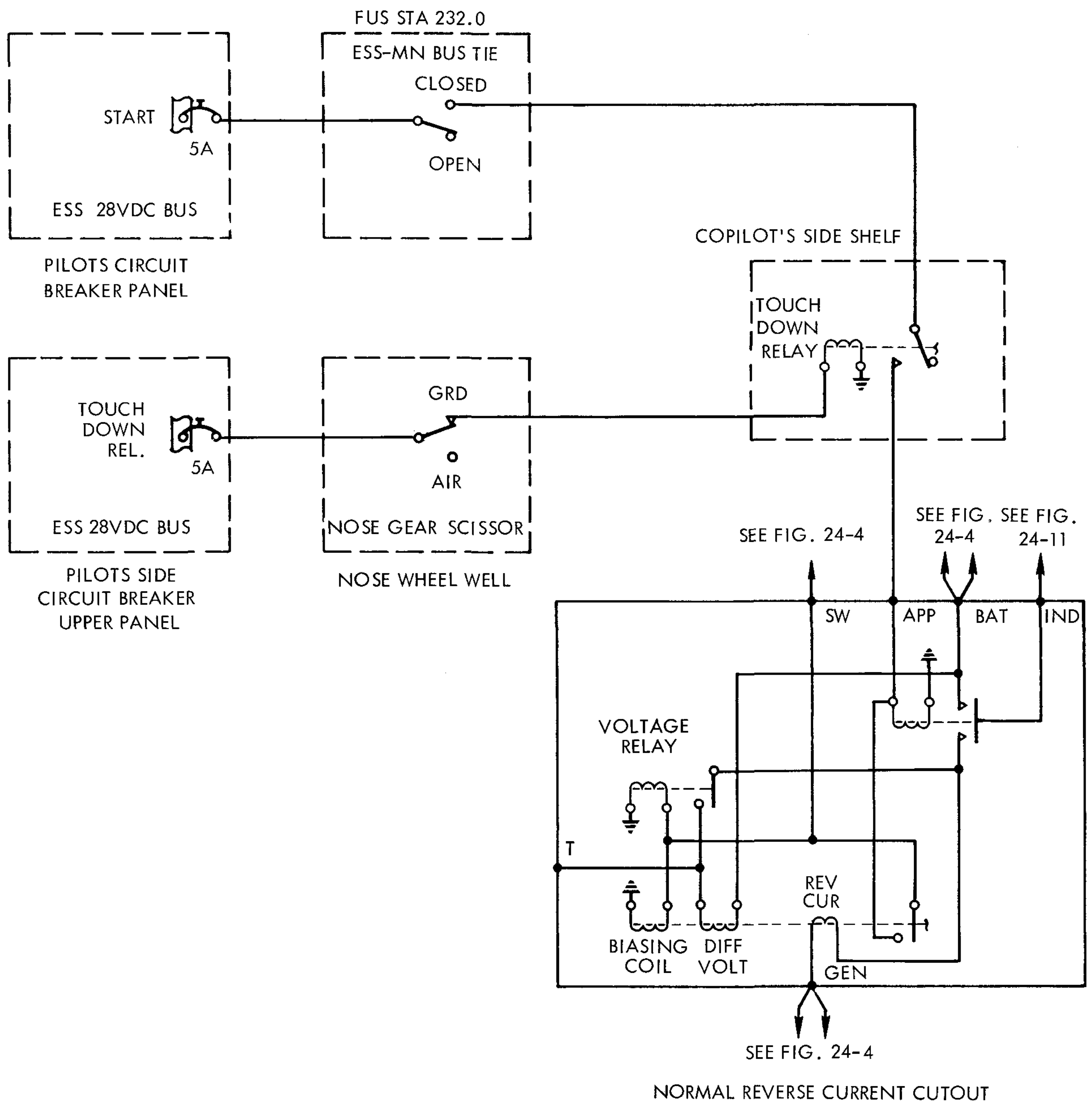
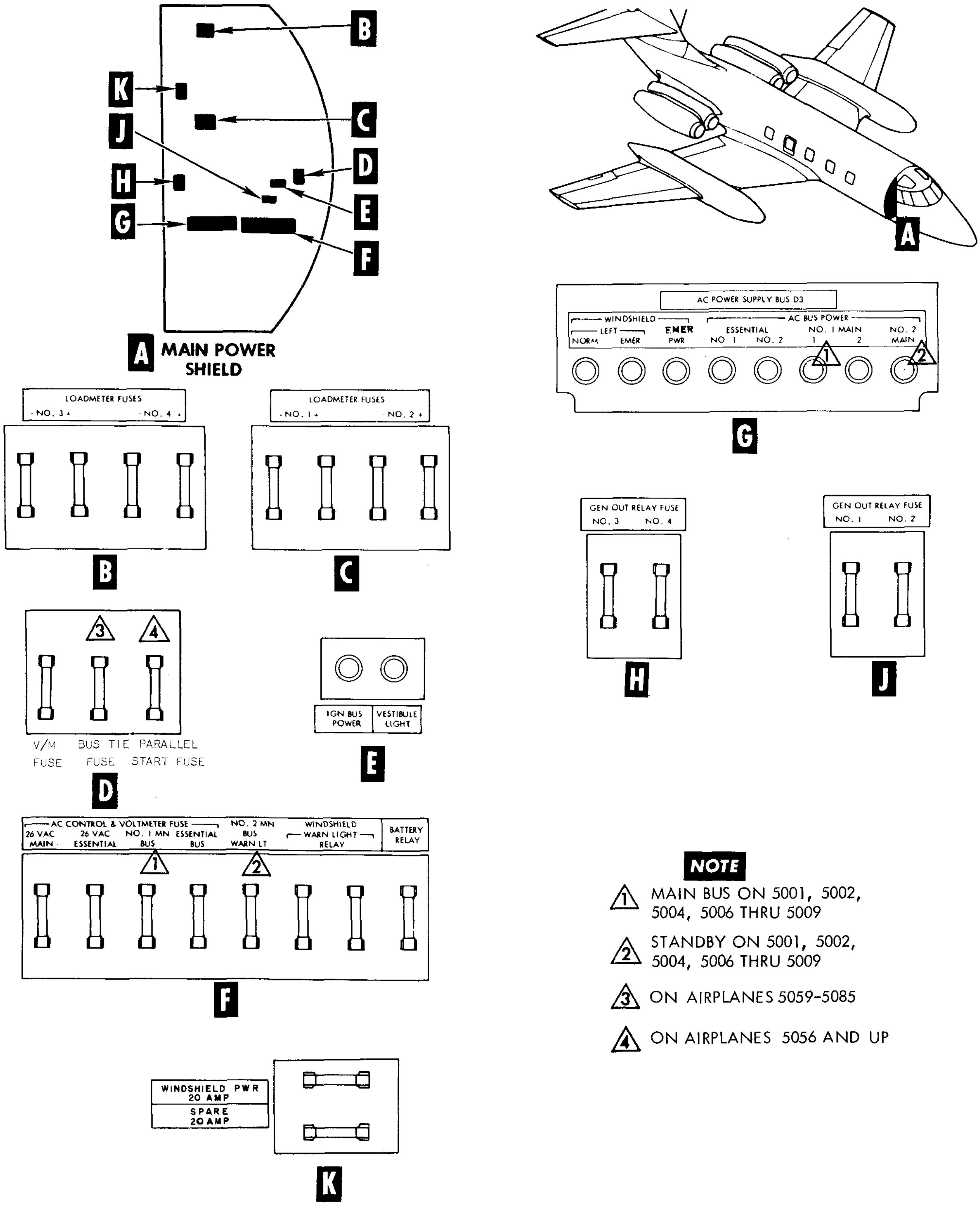


Figure 24-9B. Essential-Main Bus Tie (Airplanes 5066 and up)

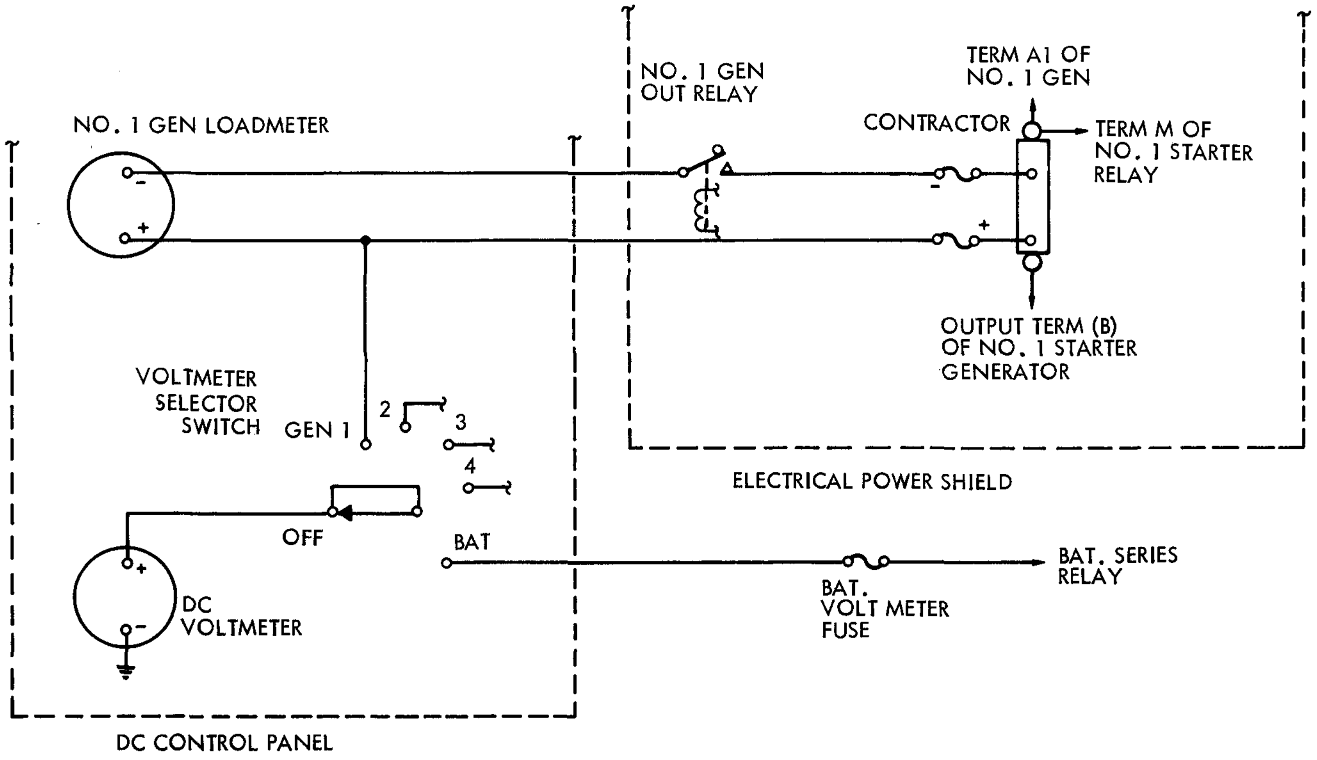


NOTE

- 1 MAIN BUS ON 5001, 5002, 5004, 5006 THRU 5009
- 2 STANDBY ON 5001, 5002, 5004, 5006 THRU 5009
- 3 ON AIRPLANES 5059-5085
- 4 ON AIRPLANES 5056 AND UP

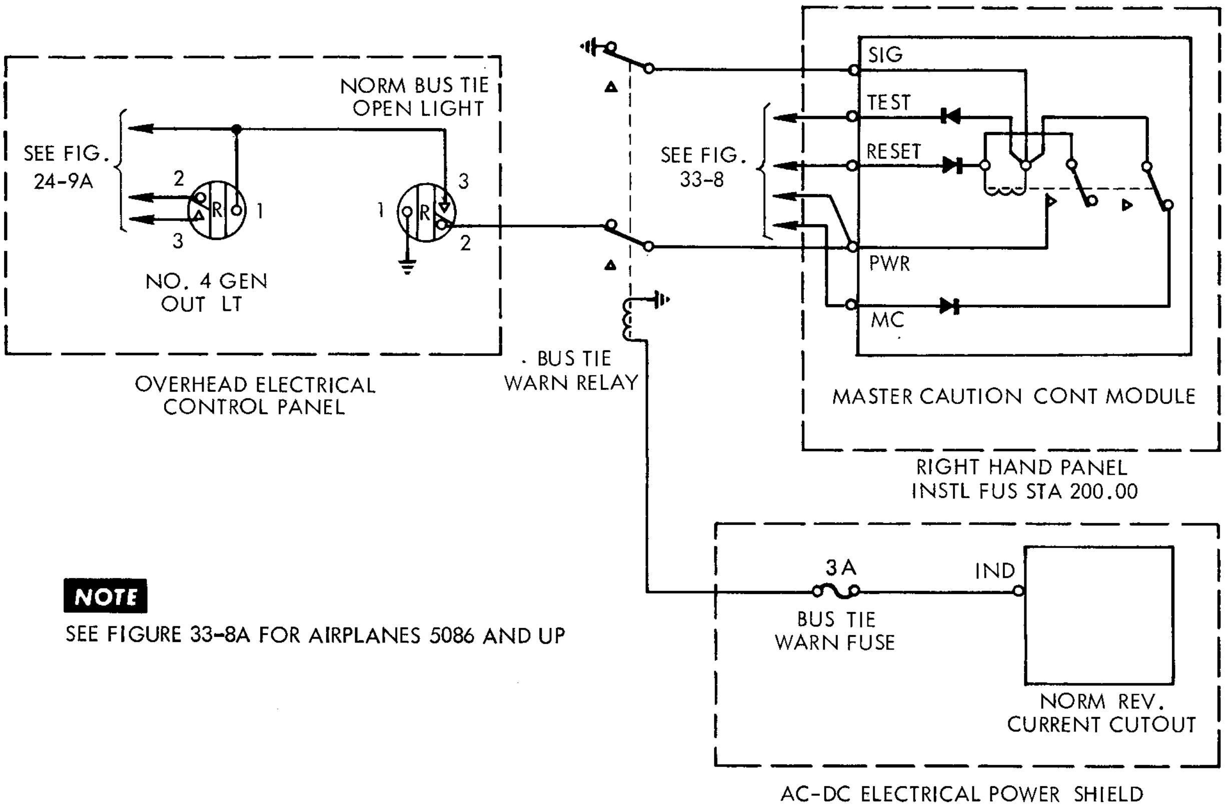
Figure 24-10. Circuit Breaker Panel and Fuses

Changed 1 December 1965



JS-1-2-X0/2-2411

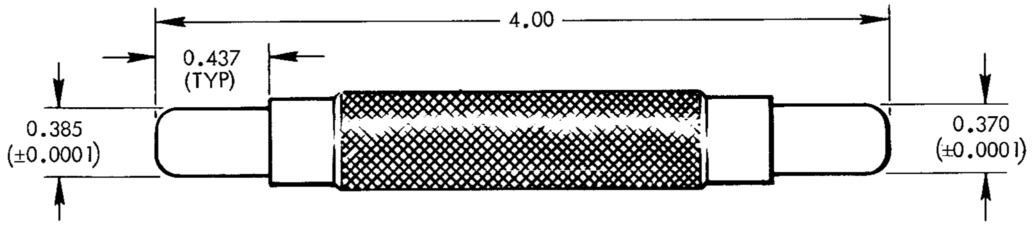
Figure 24-11. Voltmeter-Loadmeter Schematic Diagram



NOTE

SEE FIGURE 33-8A FOR AIRPLANES 5086 AND UP

Figure 24-11A. Bus Tie Warning Light (Airplanes 5059 thru 5085)



NOTE

INSPECTION GAUGE PART NO. 029
AVAILABLE FROM ELCON UPON REQUEST.

JS-1-2-X0/2-2411B

Figure 24-11B. Battery Connector Inspection Gauge

SECTION 24.1

AC POWER SYSTEM (AIRPLANES 5001, 5002, 5004 THRU 5009)

Contents Page

DESCRIPTION AND OPERATION 24-35
Inverters 24-35
AC Control Panel 24-35
Main Inverter Control Switch. 24-35
Windshield Inverter Control Switch 24-36
AC System Emergency Switch 24-36
Bus Selector Switch. 24-36
AC Voltmeter 24-36
Frequency Meter 24-36
Bus Out Lights 24-36
Warning Light Relays. 24-36
Instrument Transformers 24-36
Relay System 24-36
AC Power Distribution 24-37

OPERATIONAL CHECKOUT. 24-37

TROUBLESHOOTING. 24-38

REMOVAL AND REPLACEMENT 24-39

FIGURES
24-12. AC Power Distribution Diagram 24-43
24-13. Inverter Load Transfer. 24-44
24-14. AC Power Supply Components (Airplanes 5001, 5002, 5004, and 5006 thru 5009) 24-45
24-15. AC Power Supply Schematic Diagram (Airplanes 5001, 5002, 5004, and 5006 thru 5009) . 24-47

24.1.1 DESCRIPTION AND OPERATION. (See figure 24-15.) AC electrical power is supplied by a system composed of three inverters, two transformers, an AC control panel, and a network of relays. Using 28-volt DC input, the inverters provide 115-volt single-phase, 400-cycle AC output. The transformers step down a portion of the 115-volt AC to 26-volt AC. Normally, two inverters are operated, while one remains idle. If an operating inverter fails, the idle one can be used. If two inverters fail, the remaining one can handle the essential AC bus loads and heat the pilot's forward windshield. The idle inverter should be operated periodically to ensure that it remains operable.

24.1.1.1 INVERTERS. (See figure 24-14.) The three inverters are located in a soundproof enclosure under the aisle floor at fuselage station 295. All three inverters are similar rotary-type units. The inverter in the most aft position is a Leland MGE 23-340 waterproof unit, the other units are Leland MGE 23-34 inverters. Each inverter supplies 115-volt, single-phase, 400-cycle AC and is rated at 3,000 volt-amperes. Both the voltage and the frequency of the output can be set by rheostat adjustments, and carbon pile regulators keep the voltage and frequency within allowable limits. Normally, a main inverter supplies both the main and the essential AC buses; a windshield inverter provides for windshield heating loads. A standby inverter can be operated in place of either of the other two inverters.

24.1.1.2 AC CONTROL PANEL. (See figure 24-14.) All of the switches and indicators for the AC power supply system are on an AC control panel on the right forward portion of the overhead panel console. Two inverter control switches, an AC SYS switch, two BUS OUT lights, a voltmeter, a frequency meter, and a BUS SELECT switch for the frequency meter and the voltmeter, are on the panel.

24.1.1.3 MAIN INVERTER CONTROL SWITCH. The operation of the main inverter is controlled by a three-position ("ON," "OFF," "TRANSFER") MAIN INV

24.1.1.3 (Cont).

switch. Also, the standby inverter can be operated with this switch. With the MAIN INV switch in the "ON" position, the output of the main inverter is routed to the main and the essential AC buses; in the "TRANSFER" position, the main inverter stops operating and its loads are assumed by the standby inverter. If both the windshield and the standby inverters fail, the essential AC bus loads and the essential windshield loads are supplied by the main inverter if the MAIN INV switch is in the "ON" position and the WINDSHIELD ANTI-ICING power control switch is in the "EMER" position. If both the windshield and the main inverters fail, or if only essential DC power is available, these essential loads are supplied by the standby inverter if the MAIN INV switch is in the "TRANSFER" position and the WINDSHIELD ANTI-ICING power control switch is in the "EMER" position. Inverter load transfer and switch positions are shown in figure 24-13.

24.1.1.4 WINDSHIELD INVERTER CONTROL SWITCH. The operation of the windshield inverter is controlled by a three-position ("ON," "OFF," "TRANSFER") WINDSHIELD INVERTER switch. Also, the standby inverter can be operated with this switch. With the WINDSHIELD INVERTER switch in the "ON" position, the output of the windshield inverter is used to heat the pilots' compartment windshields; in the "TRANSFER" position, and provided the MAIN INV switch is not in the "TRANSFER" position, the windshield inverter stops operating and its loads are assumed by the standby inverter (see figure 24-13).

24.1.1.5 AC SYSTEM EMERGENCY SWITCH. A two-position ("NORMAL," "EMER") AC SYS switch provides for routing the output of one inverter to the essential AC bus and the pilot's forward windshield in case two inverters fail. When only the windshield inverter is operable, it assumes these essential loads if the AC SYS switch is in the "EMER" position and the WINDSHIELD INVERTER switch is in the "ON" position. When only the standby inverter is operable, it assumes these essential loads if the AC SYS switch is in the "EMER" position, the WINDSHIELD INVERTER switch is in the "TRANSFER" position, and the main switch is not in "TRANSFER" position (see figure 24-13).

24.1.1.6 BUS SELECTOR SWITCH. To select the bus whose voltage and frequency is to be measured, a BUS SELECT switch is provided. Either the 115-volt or the 26-volt, main or essential AC bus can be selected for a voltmeter reading; either the main or essential 115-volt, AC bus can be selected for a frequency reading. Fuses in the lines between the 115-volt buses and the BUS SELECT switch protect the meters.

24.1.1.7 AC VOLTMETER. Voltage at any of the four AC buses can be measured on an AC voltmeter on the AC control panel. The desired bus is selected with the BUS SELECT switch.

24.1.1.8 FREQUENCY METER. With either the main or the essential 115-volt, AC buses selected with the BUS SELECT switch, a frequency reading is obtained on a frequency meter on the AC control panel, provided voltage is available on the bus selected. Frequency for either of the 26-volt buses is the same as for the corresponding 115-volt bus.

24.1.1.9 BUS OUT LIGHTS. An AC BUS OUT light goes on when power is not available on the 115-volt, essential AC bus, provided the fuse is intact. A WINDSHIELD BUS OUT light goes on, under normal conditions, if voltage is not available at the windshield power transformer; it goes on, under emergency conditions, if voltage is not available at the windshield emergency power transformer.

24.1.1.10 WARNING LIGHT RELAYS. An essential bus warning light relay in the electrical power shield is energized when voltage is available on the 115-volt, essential AC bus. When this relay is de-energized, it provides a ground for the AC BUS OUT light and for the annunciator panel INVERTER OUT light, both of which then go on. A windshield inverter warning light relay, also in the electrical power shield, is energized through the windshield emergency power transfer relay by current from either the windshield power transformer or the windshield emergency power transformer. When de-energized, this relay grounds the WINDSHIELD BUS OUT light and the annunciator panel INVERTER OUT light, both of which then go on.

24.1.1.11 INSTRUMENT TRANSFORMERS. Two instrument transformers are mounted in the electrical power shield. The primary side of one transformer is connected to the 115-volt, essential AC bus, and the primary side of the second transformer to the 115-volt, main AC bus. The secondary side of both transformers supplies 26-volt, single-phase, 400-cycle, AC power, available on both the main and the essential 26-volt, AC buses.

24.1.1.12 RELAY SYSTEM. Eleven relays in the electrical power shield are arranged to effect the various inverter loadings previously described. When energized by an inverter control switch, DC inverter power relays connect 28-volt, DC input to an inverter. A main/standby inverter transfer relay coordinates the transfer of either the main inverter or the windshield inverter loads to the standby inverter. Each of three inverter output relays positions to route 115-volt, AC inverter output to the selected loads. Energized by either the AC SYS switch or the WINDSHIELD ANTI-ICING POWER CONTROL switch, a windshield emergency power transfer relay provides for the heating of only the essential windshield. An essential/main bus transfer relay, when energized, connects the main, 115-volt, AC bus to either the main or the standby inverter. This relay, when de-energized, disconnects the main AC bus and connects the windshield emergency transformer to the essential AC bus. Power to operate this relay is derived from the IND terminal of the normal reverse current cutout and the normally closed contacts of the wind-

24.1.1.12 (Cont).

shield emergency power transfer relay. Two windshield/main transfer relays, both energized by the AC SYS switch, operate when energized to transfer the output of one inverter to the essential AC buses and the essential windshield.

24.1.1.13 AC POWER DISTRIBUTION. (See figure 24-12.) Normally, essential AC power is routed through the de-energized contacts of the windshield main transfer relay No. 2, and main AC power is routed through the energized contacts of the essential/main bus transfer relay. In addition to distributing 115-volt, single-phase, 400-cycle AC at the pilots' compartment circuit breaker panels, both the main and the essential AC buses supply the primary side of instrument transformers. Windshield heating power is routed through the de-energized contacts of the windshield/main transfer relay No. 1. Regardless of which inverter is operating, 115-volt essential AC bus power is routed from the windshield main transfer relay No. 2, through two essential AC power circuit breakers in the electrical power shield, to the circuit breakers on the pilot's and copilot's circuit breaker panels. This power is then available for distribution to the various aircraft systems. A portion of the 115-volt essential bus AC passes through a fuse in the electrical power shield to the essential bus instrument transformer. The 26-volt essential AC bus instrument transformer output is available for distribution at the copilot's side and the copilot's wainscot circuit breaker panels. Main AC bus power at 115-volts is provided at the copilot's side and the copilot's wainscot circuit breaker panels through two main AC power circuit breakers in the electrical power shield. The primary side of the main bus instrument transformer is furnished 115-volt, main AC bus power through a fuse in the electrical power shield. The 26-volt output of the main bus instrument transformer is available at the copilot's side and the copilot's wainscot circuit breaker panels.

24.1.2 OPERATIONAL CHECKOUT. The AC power supply system can be checked by using the following procedure:

a. Make sure both the MAIN INV and the WINDSHIELD INVERTER switches are in the "OFF" position, the AC system NORM-EMER switch is in the "NORM" position, and the windshield ANTI-ICING switch is in the "NORM" position.

b. Make sure the following circuit breakers are open:

INV OUT LTS
MAIN INV CONT
WINDSHIELD INV CONT
WARNING INTERIOR LIGHTS
STANDBY INV CONT
WINDSHIELD INV TRANSFER

c. Connect external power to the airplane.

d. Close the INV OUT LTS circuit breaker.
RESULT: Both the AC BUS OUT light and the WINDSHIELD BUS OUT light should go on.

e. Close the WARNING INTERIOR LIGHTS circuit breaker.
RESULT: The INVERTER OUT strip on the annunciator panel and the two master CAUTION lights on the main instrument panel should go on.

f. Close the MAIN INV CONT circuit breaker, and place the MAIN INV switch in the "ON" position.
RESULT: The AC BUS OUT light should go out.

g. Place the BUS SELECT switch in the "115V MAIN" position.
RESULT: The AC voltmeter should indicate 115 (±6) volts; the airplane frequency meter should indicate 400 (±10) cycles per second. Check results with calibrated meters.

Note

System voltage and frequency must be adjusted whenever they deviate from the limits specified above. To make the voltage adjustment, connect a precision voltmeter to the main AC circuit breaker in the power shield and make the adjustment at the connector end of the control unit on top of the inverter. Set voltage at 115 volts as read on the precision meter. The frequency may be adjusted by removing the cover from the control unit and making the adjustment with the frequency control rheostat to give 400 cps on the precision frequency meter. The inverter should be warmed-up for 15 minutes prior to making these adjustments. The aircraft DC bus voltage must be within its limits and the inverter may be operated at no load. Upon completion of the voltage and frequency adjustments, the aircraft meters should be adjusted to read the same as the precision meters.

h. Place the BUS SELECT switch in the "26V MAIN" position.
RESULT: The pointer of the AC voltmeter should lie approximately halfway between 0 and 50 volts.

i. Place the BUS SELECT switch in the "115V ESS" position.
RESULT: Same as RESULT in step (g).

j. Using airplane loads, apply a load of 1.5 to 2.5 KVA to the main and essential AC buses.
RESULT: The AC voltmeter should indicate 115 (±6) volts; the airplane frequency meter should indicate 400 (±10) cycles per second.

k. Remove the load from the main and essential AC buses.

l. Place the MAIN INV switch in the "TRANSFER" position.
RESULT: The AC BUS OUT light should go on.

24.1.2 (Cont).

m. Close the STANDBY INV CONT circuit breaker.
RESULT: The AC BUS OUT light should go out, the AC voltmeter should indicate 115 (±6) volts, and the airplane frequency meter should indicate 400 (±10) cycles per second.

n. Repeat steps (j) and (k).

o. Close the WINDSHIELD INV CONT circuit breaker, and place the WINDSHIELD INVERTER switch in the "ON" position.
RESULT: The WINDSHIELD BUS OUT light should go out.

p. Place the AC SYS EMER switch in the "EMER" position.
RESULT: The WINDSHIELD BUS OUT light should remain out.

Note

The copilot's instruments will stop even though BUS OUT LT does not go on.

q. Make sure the BUS SELECT switch is in the "115V ESS" position.
RESULT: The AC voltmeter should indicate 115 (±6) volts; the airplane frequency meter should indicate 400 (±10) cycles per second.

r. Using airplane loads, apply a load to the essential AC bus and to the essential windshield.

Note

Windshield temperatures must be below 90°F or above -40°F to use the WINDSHIELD as a load.

RESULT: The AC voltmeter should indicate 115 (±6) volts; the airplane frequency meter should indicate 400 (±10) cycles per second.

s. Place the WINDSHIELD INVERTER switch in the "TRANSFER" position, and remove the load from the essential AC bus and the essential windshield.
RESULT: Both the AC BUS OUT light and the WINDSHIELD BUS OUT light should go on.

t. Place the MAIN INV switch in the "OFF" position.
RESULT: Both the AC BUS OUT light and the WINDSHIELD BUS OUT light should go out, the AC voltmeter should indicate 115 (±6) volts, and the airplane frequency meter should indicate 400 (±10) cycles per second.

u. Place the AC SYS EMER switch in the "NORMAL" position.
RESULT: The AC BUS OUT light should go on, and the WINDSHIELD BUS OUT light should remain out. No voltage should be indicated on the AC voltmeter.

v. Place the WINDSHIELD ANTI-ICING NORM-EMER switch in the "EMER" position.
RESULT: The AC BUS OUT light should remain on, the WINDSHIELD BUS OUT light should remain out, and the AC voltmeter should indicate no voltage.

w. Place the WINDSHIELD INVERTER switch in the "ON" position.
RESULT: The AC BUS OUT light should remain on and the WINDSHIELD BUS OUT light should remain out, the AC voltmeter should indicate no voltage.

x. Place the WINDSHIELD INVERTER switch in the "OFF" position, and the MAIN INV switch in the "ON" position.
RESULT: Both the AC BUS OUT light and the WINDSHIELD BUS OUT light should go out, the AC voltmeter should indicate 115 (±6) volts, and the airplane frequency meter should indicate 400 (±10) cycles per second. No voltage should remain on the main bus.

y. Place the MAIN INV switch in the "OFF" position, and the WINDSHIELD ANTI-ICING NORM-EMER switch in the "NORM" position.
RESULT: Both the AC BUS OUT light and the WINDSHIELD BUS OUT light should go on, and no voltage should be indicated on the AC voltmeter.

z. Disconnect external power from the airplane.

24.1.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO VOLTAGE - FREQUENCY NORMAL; MAIN		
Faulty AC voltmeter	Standby and windshield inverters give same indications.	Replace.
NO FREQUENCY - VOLTS NORMAL; MAIN		
Faulty frequency meter.	Standby and windshield inverters give same indications.	Replace.

24.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO VOLTAGE - NO FREQUENCY; MAIN		
Fuse in meter circuit	AC loads should operate normally.	Replace.
Faulty inverter.	Transfer to Standby give normal operation.	Replace.
Faulty inverter wiring.	Substitute Standby in main position give same results.	Repair.
VOLTAGE AND/OR FREQUENCY FLUCTUATION; MAIN		
FAULTY inverter.	Transfer to Standby gives normal operation.	Replace.
Faulty inverter wiring.	Substitute standby in main position gives same results.	Repair.
VOLTAGE AND/OR FREQUENCY OUT OF LIMITS; MAIN		
Faulty AC meters.	Standby and windshield inverter give same indications.	Replace.
Improperly adjusted regulator.	Transfer to standby gives normal operation.	Adjust per procedure in 24.1.2.
Faulty inverter.	Transfer to standby gives normal operation.	Replace.
Faulty wiring.	Substitute standby in main position gives same results.	Repair.
BUS - OUT LIGHT ON; INVERTER SWITCH ON - MAIN		
Fuse in meter circuit.	Meters indicate no output; AC loads operable.	Replace
Defective relay.	Meters and Loads normal	Replace
Faulty inverter	Transfer to standby gives normal operation	Replace
Faulty inverter wiring	Substitution of standby in main position gives same results.	Repair

24.1.4 REMOVAL AND REPLACEMENT.

24.1.4.1 INVERTERS. To gain access to the inverter compartment for the removal of the airplane inverters, remove the center floor panel between fuselage stations 270.5 and 310.0.

Note

Particular care must be taken during removal and installation of inverters to preserve the buildup of washers, spacers, and bolts as shown in figure 24-23. Trim the bushing to a length of 1.25 inches before installing. Tighten the bolt until washers cannot be turned by hand, and then back off approximately one turn.

INSPECTION NOTE

Inspect inverter mounting bolts for looseness or bottoming out against the bushing. The effectiveness of the mounting in preventing noise will be severely reduced if the bushing is pressed against the structure. If wear or aging of the rubber has caused looseness in the mount, the bushing may be shortened or washers may be added as required.

24.1.4.2 REMOVAL AND REPLACEMENT OF ARMATURE SHAFT BEARINGS ON MGE-23-34 AND MGE-23-340 INVERTERS. INVERTERS.

24.1.4.2.1 DISASSEMBLY. The removal of the bearings should be performed in accordance with the following instructions and in conjunction with figure 24-22.

a. Remove fan cover (1), two screws (2), and two washers (3).

b. Remove four screws (5), lift up on brush springs, and remove DC brushes (4) from their brush holders.

c. Remove the AC end cover (6), four screws (7), and four washers (8).

24.1.4.2.1 (Cont).

d. Remove four brush holder caps (9) and remove the AC brush assemblies (10 and 11).

e. Remove the lock nut (12), lock washers (13), fan (14), flat key (15) and a fan spacer (16) from the DC end of the armature shaft.

f. Remove the four screws (18) and lock washer (19). Remove the external bearing plate (17).

g. Remove four thru-bolt screws (22) and nuts (23) from opposite ends of the stator (67). Free stator leads from the DC shield assembly (36) and remove shield assembly from the stator. Tap the shield several times with a nonmetallic mallet to facilitate removal.

h. Remove the two screws (40), bearing plate (39), shim washer (41), loading spring (42) and shim washers (43) from the AC end shield assembly (64).

i. Remove the cover plate (44) from the AC end shield (64). Remove two screws (46) to free the stator leads from the AC end shield.

j. Remove the AC end shield assembly (64) from the stator (67). Tap the end shield several times with a fiber mallet to facilitate removal.

k. Remove the armature (66) from the DC end of the stator (67). Press ball bearings (37) and (65) from the shaft of armature (66). Remove an internal bearing plate (38) from the armature shaft.

CAUTION

Handle the armature carefully. Provide suitable protection for commutator, slip rings, and bearing journal surfaces.

24.1.4.2.2 INSPECTION.

a. AC shield assembly (64). The bearing housing must be smooth and free of scoring marks or fretting corrosion. The min/max I.D. of the bearing housing should be 1.3780/1.3788. Replace if necessary.

b. DC shield assembly (36). The bearing housing must be smooth and free of scoring marks or fretting corrosion. The min/max I.D. of the bearing housing should be 1.3780/1.3788. Replace if necessary.

c. Armature shaft (66). The bearing sections of the shaft must be smooth and free of scoring marks or fretting corrosion. The min/max O.D. of the bearing surface should be 0.5903/0.5906. Replace if necessary.

d. Before pressing on new ball bearing, make sure the shaft shoulder is square, free from burrs and undamaged. The shaft shoulder shall be free from any roughness caused by the spinning of the bearing or the bearing thrust. Make sure that the undercut on the shaft shoulder is free from dirt or damage to the extent that it will prevent the bearing from setting against the shaft shoulder. The shaft chamber shall not be damaged and shall be free from burrs. The bearing surfaces of the shaft should be clean.

24.1.4.2.3 REASSEMBLY.

Note

Make sure the internal bearing plate (38) is in position at the DC end of the armature before assembling ball bearings.

a. Press new ball bearings, Leland No. A-15884 (37) and (65) on both ends of the shaft of armature assembly (66). An arbor press or pressure sealing type press may be used. Apply pressure to the inner race by means of a fixture which insures a square seat on the bearing ring. Release the pressure immediately after the bearing is in place in order to avoid bending the shaft.

Note

The aligning fixture must be made in such a way that the force is applied squarely on the face of the bearing and exactly in the same perpendicular axis as the shaft axis without wobbling. This is required to avoid pressing the bearing on in such a manner that the bearing I.D. is actually shaving the shaft slightly due to a side force of the press or the aligning fixture. The fixture must be inspected frequently as no burrs or improper seating in the bearing races are permissible. The two faces of the fixture for pressing the bearing must be parallel and of smooth finish. Make sure that the tool will fit only over the steel section of the inner ring. It must not touch any part of the shield and/or contact seals. Precautions must be taken to avoid moving of the tool in the radial direction of the bearing.

b. Work the AC end shield assembly (64) over the shaft of the armature and onto the stator (67). Tap the shield lightly with a fiber mallet to seat.

c. Install the armature assembly (66) in the stator (67) from the AC end. Work the DC and shield assembly (36) in position over the bearing (37) and onto the stator (67). Tap the end shield with

24.1.4.2.3 (Cont).

a fiber mallet to fully seat it. Position the external bearing plate (17) over armature shaft and the DC shield assembly (36). Install four screws (18) with lock washers (19) through the bearing plate (17) and shield (36) and into the internal bearing plate (38). Tighten and lock wire the screws (18).

d. Install thru-bolt screws (22) and nuts (23) at AC end of unit. Torque thru-bolts evenly 40 to 50 inch-pounds at room temperature.

e. Connect stator leads to the AC end shield assembly (64) using two screws (46). Replace cover plate (44) to AC end shield assembly (64).

f. Place loading spring (42) on the AC end of the armature shaft, and meassure the "A" distance as indicated in figure 24-22, detail B. Add 0.010 to 0.015 inch to the "A" reading to obtain the required shim thickness. Remove the loading spring (42) and add shim washers (43); replace loading spring. Position shim washers (41) and bearing plate (39), and fasten in place with two screws (40).

CAUTION

Failure to properly shim the loading spring results in incorrect bearing forces, and ultimately causes premature bearing failure.

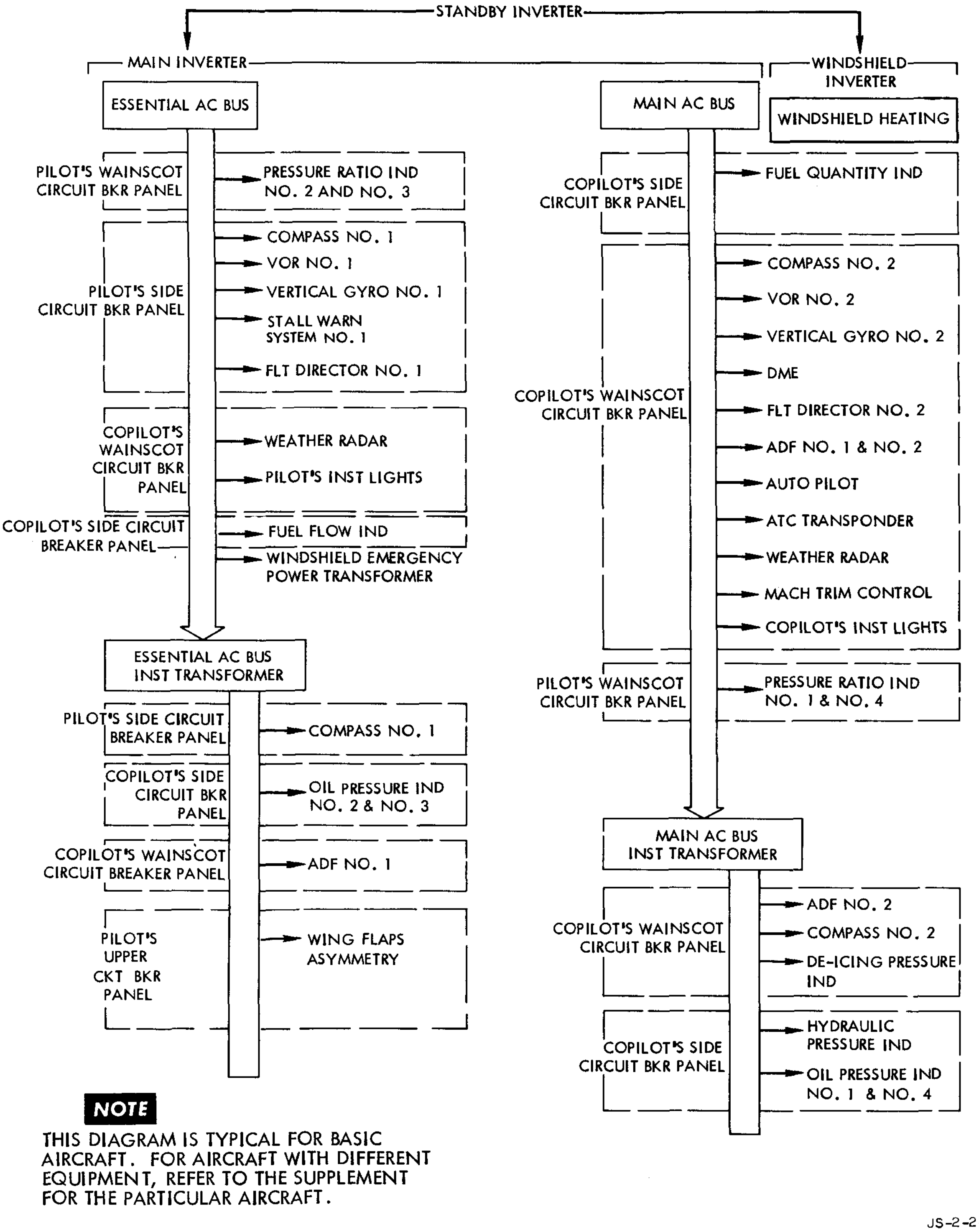
g. Check that the armature rotates freely when turned by hand.

h. Install DC brushes by lifting brush tension springs and installing four new DC brush assemblies (4) in the brush holder assemblies. The face of the brushes must fit the contour of the commutator. Attach each set of brush leads to its brush holder with a screw (5).

i. Install straight AC brushes (10) and angle brush assembly (11) in the AC end shield. Assemble a brush holder cap (9) at each brush holder. To the DC end of the armature shaft, assemble in order, a fan spacer (16), flat key (15), fan (14), new lock washer (13) and a lock nut (12). Bend the tabs of lock washer (13) to lock the nut (12).

j. Position the AC end cover (6) on the AC shield assembly and attach with four screws (7) and washers (8).

k. Assemble the fan cover (1) to the DC end shield with two screws (2) and washers (3).


STANDBY INVERTER
MAIN INVERTER
WINDSHIELD INVERTER
ESSENTIAL AC BUS
MAIN AC BUS
WINDSHIELD HEATING
PILOT'S WAINSCOT CIRCUIT BKR PANEL
PRESSURE RATIO IND NO. 2 AND NO. 3
PILOT'S SIDE CIRCUIT BKR PANEL
COMPASS NO. 1
VOR NO. 1
VERTICAL GYRO NO. 1
STALL WARN SYSTEM NO. 1
FLT DIRECTOR NO. 1
COPILOT'S WAINSCOT CIRCUIT BKR PANEL
WEATHER RADAR
PILOT'S INST LIGHTS
COPILOT'S SIDE CIRCUIT BREAKER PANEL
FUEL FLOW IND
WINDSHIELD EMERGENCY POWER TRANSFORMER
ESSENTIAL AC BUS INST TRANSFORMER
PILOT'S SIDE CIRCUIT BREAKER PANEL
COMPASS NO. 1
COPILOT'S SIDE CIRCUIT BKR PANEL
OIL PRESSURE IND NO. 2 & NO. 3
COPILOT'S WAINSCOT CIRCUIT BREAKER PANEL
ADF NO. 1
PILOT'S UPPER CKT BKR PANEL
WING FLAPS ASYMMETRY
COPILOT'S SIDE CIRCUIT BKR PANEL
FUEL QUANTITY IND
COPILOT'S WAINSCOT CIRCUIT BKR PANEL
COMPASS NO. 2
VOR NO. 2
VERTICAL GYRO NO. 2
DME
FLT DIRECTOR NO. 2
ADF NO. 1 & NO. 2
AUTO PILOT
ATC TRANSPONDER
WEATHER RADAR
MACH TRIM CONTROL
COPILOT'S INST LIGHTS
PILOT'S WAINSCOT CIRCUIT BKR PANEL
PRESSURE RATIO IND NO. 1 & NO. 4
MAIN AC BUS INST TRANSFORMER
COPILOT'S WAINSCOT CIRCUIT BKR PANEL
ADF NO. 2
COMPASS NO. 2
DE-ICING PRESSURE IND
COPILOT'S SIDE CIRCUIT BKR PANEL
HYDRAULIC PRESSURE IND
OIL PRESSURE IND NO. 1 & NO. 4

NOTE

THIS DIAGRAM IS TYPICAL FOR BASIC AIRCRAFT. FOR AIRCRAFT WITH DIFFERENT EQUIPMENT, REFER TO THE SUPPLEMENT FOR THE PARTICULAR AIRCRAFT.

JS-2-2-2420

Figure 24-12. AC Power Distribution Diagram

Switch Positions	MAIN INV SWITCH	ON	ON	TRANSFER	ON	ON, OFF, TRANSFER	ON, OFF	TRANSFER
	WINDSHIELD INV SWITCH	ON	TRANSFER	ON	ON, OFF, TRANSFER	ON	TRANSFER	ON, OFF, TRANSFER
	AC SYS SWITCH	NORMAL	NORMAL	NORMAL	*NORMAL	EMER	EMER	*NORMAL
Inverter Loads	ESS AC LOADS	M	M	S	M	W	S	S
	MAIN AC LOADS	M	M	S				
	WINDSHIELD LOADS	W	S	W				
	ESS WS LOADS	W	S	W	M	W	S	S

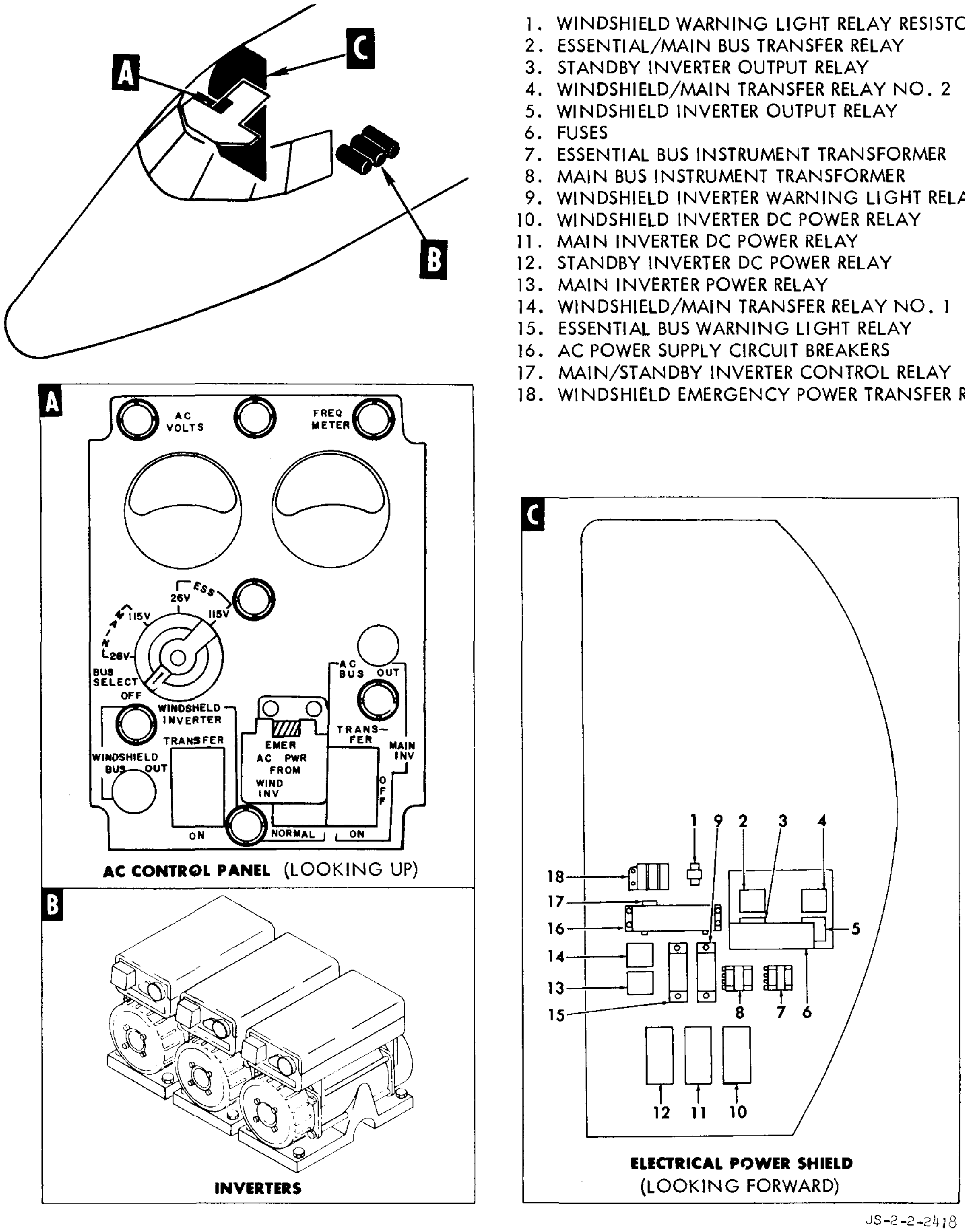
NOTE: In the Inverter Load columns, M, W, and S stand for the main, the windshield, and the standby inverters, respectively. The particular AC loads are supplied by the inverter specified when the control switches are in the positions given to the left.

*WINDSHIELD ANTI-ICING POWER CONTROL switch in "EMER" position

JS-2-2-2419

Figure 24-13. Inverter Load Tansfer

1. WINDSHIELD WARNING LIGHT RELAY RESISTOR
2. ESSENTIAL/MAIN BUS TRANSFER RELAY
3. STANDBY INVERTER OUTPUT RELAY
4. WINDSHIELD/MAIN TRANSFER RELAY NO. 2
5. WINDSHIELD INVERTER OUTPUT RELAY
6. FUSES
7. ESSENTIAL BUS INSTRUMENT TRANSFORMER
8. MAIN BUS INSTRUMENT TRANSFORMER
9. WINDSHIELD INVERTER WARNING LIGHT RELAY
10. WINDSHIELD INVERTER DC POWER RELAY
11. MAIN INVERTER DC POWER RELAY
12. STANDBY INVERTER DC POWER RELAY
13. MAIN INVERTER POWER RELAY
14. WINDSHIELD/MAIN TRANSFER RELAY NO. 1
15. ESSENTIAL BUS WARNING LIGHT RELAY
16. AC POWER SUPPLY CIRCUIT BREAKERS
17. MAIN/STANDBY INVERTER CONTROL RELAY
18. WINDSHIELD EMERGENCY POWER TRANSFER RELAY



AC CONTROL PANEL (LOOKING UP)

INVERTERS

ELECTRICAL POWER SHIELD (LOOKING FORWARD)

Figure 24-14. AC Power Supply Components (Airplanes 5001, 5002, 5004, and 5006 thru 5009)

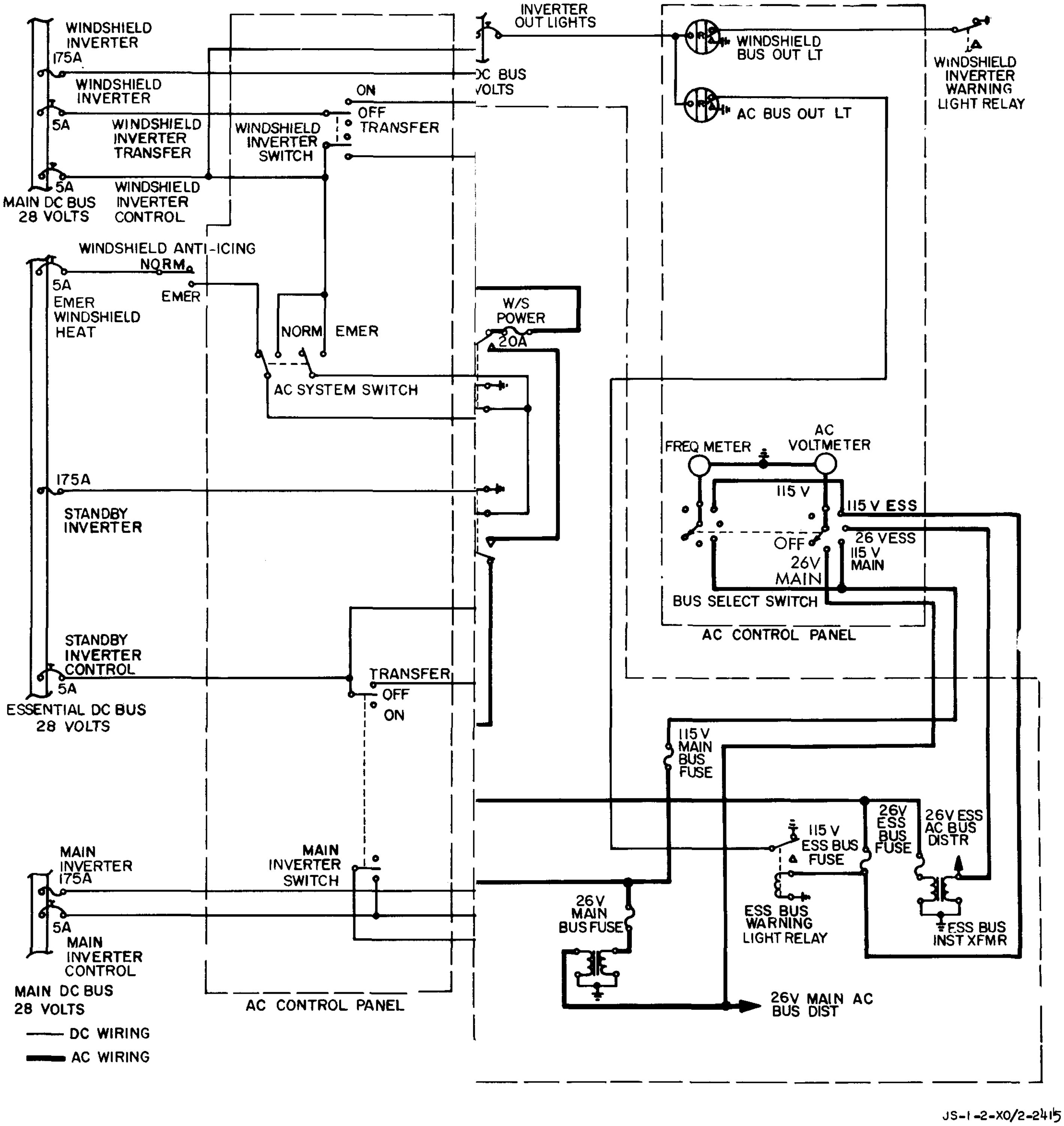


Figure 24-15. AC Power Supply Schemat

SECTION 24.2
AC POWER SYSTEM (AIRPLANES 5003, 5005, 5010 AND UP)

Contents Page

DESCRIPTION AND OPERATION 24-49
- Inverters 24-49
- AC Control Panel 24-49
- Main Inverter Control Switch 24-50
- AC System Emergency Switch 24-50
- Bus Selector Switch 24-50
- AC Voltmeter 24-50
- Frequency Meter 24-50
- Bus Out Lights 24-50
- Warning Light Relays 24-50
- Instrument Transformers 24-51
- Relay System 24-51
- AC Power Distribution 24-51

OPERATIONAL CHECKOUT 24-51

TROUBLESHOOTING 24-52

REMOVAL AND REPLACEMENT 24-53

FIGURES

- 24-16. Inverter Load Transfer 24-57
- 24-17. AC Power Distribution Diagram 24-58
- 24-18. AC Power Supply Components (Airplanes 5003, 5005, 5010, and up) 24-59
- 24-19. AC Power Supply Schematic Diagram (Airplanes 5141, and up) 24-60A
- 24-19A. AC Power Supply Schematic Diagram (Airplanes 5003, 5046 thru 5140) 24-60C
- 24-20. AC Power Supply Schematic Diagram (Airplanes 5005, 5011 thru 5016, 5018 thru 5021, 5023, 5025, and 5029) 24-61
- 24-21. AC Power Supply Schematic Diagram (Airplanes 5033, and 5035 thru 5040) 24-63
- 24-22. Motor-Generator Assembly (Inverter) 24-65
- 24-23. Inverter Installation 24-67

24.2.1 DESCRIPTION AND OPERATION. (See figures 24-19 thru 24-21.) AC electrical power is supplied by a system composed of three inverters, two transformers, an AC control panel, and a network of relays. Using 28-volt DC input, the inverters provide 115-volt, single-phase, 400-cycle, AC output. The transformers step down a portion of the 115-volt AC to 26-volt AC. Normally, all three inverters are operated. If either the No. 1 main inverter or the windshield inverter fails, the loads can be assumed by the No. 2 main inverter. The No. 2 main AC bus will then be de-energized. If any two inverters fail, the remaining inverter can handle the essential AC bus loads and heat the pilot's forward windshield.

24.2.1.1 INVERTERS. (See figure 24-18.) The three inverters are located in a soundproof enclosure under the aisle floor at fuselage station 295. All three inverters are similar rotary-type units. On aircraft 5001 thru 5040, the inverter in the most aft position is a Leland MGE23-340 waterproof unit, the other units are Leland MGE23-34 inverters. MGE23-340 units may be interchanged in the forward and center positions but MGE23-34 cannot be installed in the aft position. On aircraft 5046 and up some airplanes have three MGE23-34 inverters and some airplanes have three MGE23-340 units. Refer to Electrical and Electronics Wiring Diagram Book (Wiring Diagrams JR111 and JR716) for the installation in a particular aircraft. On these aircraft the MGE23-34 and MGE23-340 units may be interchanged in any of the three positions. Each inverter supplies 115-volt, single-phase, 400-cycle AC, and is rated at 3,000 volt-amperes. Both the voltage and the frequency of the output can be set by rheostat adjustments, and two carbon pile regulators keep voltage and frequency within allowable limits. Normally, the No. 1 main inverter supplies both the No. 1 main and the essential AC buses; the windshield inverter provides for windshield heating loads. The No. 2 main AC bus is supplied by the No. 2 main inverter.

24.2.1.2 AC CONTROL PANEL. (See figure 24-18.) All of the switches and indicators for the AC power supply system are on an AC control panel on the right forward portion of the overhead panel console. Three inverter control switches, three BUS OUT lights, an AC SYS switch, a voltmeter, a frequency meter, and a BUS SELECT switch are all on the AC control panel.

24.2.1.3 NO. 1 MAIN INVERTER CONTROL SWITCH. The operation of the No. 1 main inverter is controlled by a three-position ("ON," "OFF," "TRANSFER") NO. 1 MAIN INV switch. Also, the routing of the output of the No. 2 main inverter can be controlled with this switch. With the No. 1 MAIN INV switch in the "ON" position, the output of the No. 1 main inverter is routed to the No. 1 main and the essential AC buses. In the "TRANSFER" position, the No. 1 main inverter stops operating. The No. 1 main and essential bus loads are assumed by the No. 2 main inverter provided it is turned on. In addition, the No. 2 main bus loads are dropped. If both the No. 2 main and the windshield inverters fail, the essential AC bus loads and pilot's forward windshield are supplied by the No. 1 main inverter, provided the No. 1 MAIN INV switch is in the "ON" position and the WINDSHIELD ANTI-ICING POWER CONTROL switch is in the "EMER" position.

24.2.1.4 WINDSHIELD INVERTER CONTROL SWITCH. The operation of the windshield inverter is controlled by a three-position ("ON," "OFF," "TRANSFER") WINDSHIELD INV switch. Also, the routing of the output of the No. 2 main inverter can be controlled with this switch, provided the NO. 1 MAIN INV switch is not in the "TRANSFER" position. With the WINDSHIELD INV switch in the "ON" position, the output of the windshield inverter is used to heat the pilots' compartment windshields. In the "TRANSFER" position, the windshield inverter stops operating, and the loads are assumed by the No. 2 main inverter and the main No. 2 bus lines or dropped provided the main No. 2 inverter is turned on and the NO. 1 MAIN INV switch is not the "TRANSFER" position. If both the No. 1 main and the No. 2 main inverters fail, the essential AC bus loads and the essential windshield loads are supplied by the windshield inverter, provided the WINDSHIELD INV switch is in the "ON" position and the AC SYS switch is in the "EMER" position. (See figure 24-18.)

24.2.1.5 NO. 2 MAIN INVERTER CONTROL SWITCH. A two-position ("ON," "OFF") NO. 2 MAIN INV switch controls the operation of the No. 2 main inverter, but routing of the output of this inverter can be controlled by either the NO. 1 MAIN INV switch or the WINDSHIELD INV switch. With the No. 2 MAIN INV switch in the "ON" position, the No. 2 main inverter is supplied essential DC bus power, and the output is normally routed to the No. 2 main AC bus.

24.2.1.6 AC SYSTEM EMERGENCY SWITCH. A two-position ("NORMAL," "EMER") AC SYS switch provides for routing the output of one inverter to the essential AC bus and the pilot's forward windshield in case two inverters fail. If only the windshield inverter is operable, the windshield inverter assumes the essential loads if the AC SYS switch is in the "EMER" position, and the WINDSHIELD INV switch is in the "ON" position. If only the No. 2 main inverter is operable, the No. 2 main inverter assumes the essential loads if the AC SYS switch is in the "EMER" position and the WINDSHIELD INV switch is in the "TRANSFER" position, and provided the NO. 1 MAIN INV switch is not in the "TRANSFER" position. (See figure 24-18.)

24.2.1.7 BUS SELECTOR SWITCH. To select the bus, for measurement of bus voltage and frequency, a BUS SELECT switch is provided. Either the 115-volt or the 26-volt, No. 1 main or the essential AC bus, or the No. 2 main AC bus can be selected for a voltmeter reading; either the No. 1 main, the No. 2 main, or the essential 115-volt AC bus can be selected for a frequency reading. Both meters are protected by fuses.

24.2.1.8 AC VOLTMETER. Voltage of the AC buses can be measured on an AC voltmeter which is mounted on the AC control panel. The bus to be measured is selected with the BUS SELECT switch.

24.2.1.9 FREQUENCY METER. With either the No. 1 main, the No. 2 main, or the essential 115-volt, AC bus selected with the BUS SELECT switch, a frequency reading is obtained on a frequency meter, which is mounted on the AC control panel, if voltage is available on the selected bus. Frequency for either of the 26-volt buses is the same as for the corresponding 115-volt bus, but is not monitored because the low voltage causes an incorrect frequency reading.

24.2.1.10 BUS OUT LIGHTS. The NO. 1 MAIN BUS OUT light goes on when power is not available on the 115-volt essential AC bus, if the fuse is intact. The NO. 2 MAIN BUS OUT light goes on when power is not available on the No. 2 main AC bus, if the fuse is intact. Under normal windshield heating operation, the WINDSHIELD BUS OUT light goes on if voltage is not available at the windshield power transformer; under emergency windshield heating operation, the WINDSHIELD BUS OUT light goes on if power is not available at the windshield emergency power transformer.

24.2.1.11 WARNING LIGHT RELAYS. The essential bus warning light relay in the electrical power shield is energized when voltage is available on the 115-volt essential AC bus. When this relay is de-energized, it provides a ground for the NO. 1 MAIN BUS OUT light and for the annunciator warning light controller. Then the NO. 1 MAIN BUS OUT light goes on, and also the INVERTER OUT module on the annunciator warning panel, together with two master caution lights go on. The No. 2 main bus warning light relay remains energized while power is on the No. 2 main AC bus. When de-energized, this relay furnishes a ground to both the NO. 2 MAIN BUS OUT light and to the annunciator warning light controller. Then the NO. 2 MAIN BUS OUT light goes on, and also the INVERTER OUT module on the annunciator warning panel, together with two master caution lights, go on. The No. 2 main bus warning light relay remains energized while power is on the No. 2 main AC bus. When de-energized, this relay furnishes a ground to both the NO. 2 MAIN BUS OUT light

24.2.1.11 (Cont).

and to the annunciator warning light controller. Then the NO. 2 MAIN BUS OUT light goes on, and the INVERTER OUT module on the annunciator warning panel, together with the master caution lights, go on. The windshield inverter warning light relay is energized through the windshield emergency power transfer relay by current from either the windshield power transformer or from the windshield emergency power transformer. When de-energized, this relay grounds the WINDSHIELD BUS OUT light and the warning light controller, giving appropriate indication of the power failure on the AC control panel and on the main instrument panel. The WINDSHIELD BUS OUT light, the INVERTER OUT module on the annunciator warning panel, and the two master caution lights all go on.

24.2.1.12 INSTRUMENT TRANSFORMERS. Two instrument transformers are mounted in the electrical power shield. The primary side of one transformer is connected to the 115-volt essential AC bus, and the primary side of the second transformer is connected to the 115-volt, No. 1 main AC bus. The secondary side of both transformers furnishes 26-volt, single-phase, 400-cycle AC power, which is available on both the main and the essential 26-volt, AC busses.

24.2.1.13 RELAY SYSTEM. Twelve relays in the electrical power shield are arranged to effect the various inverter loadings previously described. When energized by the corresponding inverter control switch, DC inverter input power relays connect 28-volt, DC power to the appropriate inverter. Each of three inverter output relays routes 115-volt, AC power to the selected loads. A No. 1 main/windshield bus transfer relay controls the routing of the output of the No. 2 main inverter. This relay is, in turn, controlled by a No. 1 main transfer control relay. Whether power is routed to the No. 1 main AC bus or the windshield emergency power transformer is determined by a No. 1 main bus disconnect relay. A windshield/No. 1 main transfer relay No. 1 directs power to either the windshield power transformer or to the windshield emergency power transformer, and the windshield/No. 1 main transfer relay No. 2 controls power routed to the essential, the No. 1 main, and the emergency windshield loads. Energized by either the AC SYS switch or the WINDSHIELD ANTI-ICING POWER CONTROL switch, a windshield emergency power transfer relay provides for heating of only the pilot's forward windshield.

24.2.1.14 AC POWER DISTRIBUTION. (See figure 24-17.) Normally, essential AC bus power is furnished through the de-energized contacts of the windshield/No. 1 main transfer relay No. 2, and the No. 1 main AC bus power through the energized contacts of the No. 1 main bus disconnect relay. In addition to distributing 115-volt, single-phase, 400-cycle, AC power at the pilots' compartment circuit breaker panels, both the No. 1 main and the essential AC buses supply the primary side of the instrument transformers. The No. 2 main AC bus power is routed directly to the bus from the inverter output relay. Windshield heating power is routed through the de-energized contacts of the windshield/No. 1 main transfer relay No. 1.

24.2.2 OPERATIONAL CHECKOUT. The following procedure can be used to check the operation of the AC power supply system:

a. Make sure the NO. 1 MAIN INV, the NO. 2 MAIN INV, and the WINDSHIELD INV switches are in the "OFF" position; the WINDSHIELD ANTI-ICING NORM-EMER switch is in the "NORM" position; and the AC SYS switch is in the "NORMAL" position.

b. Make sure the following circuit breakers are open: NO. 1 MAIN INVERTER CONT, NO. 2 MAIN INV CONT, WINDSHIELD INVERTER CONT, WINDSHIELD INVERTER TRANSFER, and EMER WINDSHIELD HEAT.

c. Make sure the following circuit breakers are closed: NORM BUS TIE RELAY, WARNING INTERIOR LIGHTS, and INV OUT LTS.

d. Connect external power to the airplane.
RESULT: The NO. 1 MAIN BUS OUT, the NO. 2 MAIN BUS OUT, and the WINDSHIELD BUS OUT lights should go on. Also, the INVERTER OUT annunciator module and the two master CAUTION lights should go on.

e. Close the NO. 1 MAIN INVERTER CONT circuit breaker, place the NO. 1 MAIN INV switch in the "ON" position, and successively place the BUS SELECT switch in the "115V NO. 1 MAIN," "115V ESS," "26V NO. 1 MAIN," and the "26V ESS" positions.
RESULT: The NO. 1 MAIN BUS OUT light should go off. With the BUS SELECT switch in the "115V NO. 1 MAIN" and the "115V ESS" positions, the AC voltmeter should indicate 115(±6) volts, and the frequency meter 400(±10) cycles per second. With the BUS SELECT switch in the "26V NO. 1 MAIN" and the "26V ESS" positions, the AC voltmeter should indicate between 0 and 50 volts.

f. Check the results of the previous step with a calibrated voltmeter and frequency meter.
RESULT: The readings obtained with the airplane AC voltmeter and frequency meter should be in close agreement with those obtained with the calibrated meters.

Note

System voltage and frequency must be adjusted whenever they deviate from the limits specified above. To make the voltage adjustment, connect a precision voltmeter to the main AC

24.2.2 (Cont).

circuit breaker in the power shield, and make the adjustment at the connector end of the control unit or top of the inverter. Set voltage at 115 volts as read on the precision meter. The frequency may be adjusted by removing the cover from the control unit and making the adjustment with the frequency control rheostat to give 400 CPS on the precision frequency meter. The inverter should be warmed up for 15 minutes prior to making these adjustments. The aircraft DC bus voltage must be within its limits, and the inverter may be operated at no load. Upon completion of the voltage and frequency adjustments, the aircraft meters must be adjusted to read the same as the precision meters.

g. Close the EMER WINDSHIELD HEAT circuit breaker, and place the WINDSHIELD ANTI-ICING NORM-EMER switch in the "EMER" position.
RESULT: The NO. 1 MAIN BUS OUT light should remain off, and the WINDSHIELD BUS OUT light should go off.

h. Close the NO. 2 MAIN INV CONT circuit breaker, place the NO. 2 MAIN INV switch in the "ON" position, and place the BUS SELECT switch in the "115V NO. 2 MAIN" position.
RESULT: The NO. 2 MAIN BUS OUT light should go off, the AC voltmeter should indicate 115(±6) volts, and the frequency meter should indicate 400(±10) cycles per second.

i. Place the NO. 1 MAIN INV switch in the "TRANSFER" position.
RESULT: The NO. 2 MAIN BUS OUT light should go on, but both the NO. 1 MAIN BUS OUT light and the WINDSHIELD BUS OUT light should remain off.

j. Place the WINDSHIELD ANTI-ICING NORM-EMER switch in the "NORM" position.
RESULT: The WINDSHIELD BUS OUT light should go on, but the NO. 1 MAIN BUS OUT light should remain off.

k. Close the WINDSHIELD INVERTER CONTROL circuit breaker, place the NO. 1 MAIN INV switch in the "OFF" position, and the WINDSHIELD INV switch in the "TRANSFER" position.
RESULT: The NO. 1 MAIN BUS OUT light should go on, the WINDSHIELD BUS OUT light should go off, and the NO. 2 MAIN BUS OUT light should remain on.

l. Close the WINDSHIELD INVERTER TRANSFER circuit breaker and place the AC SYS switch in the "EMER" position.
RESULT: The NO. 1 MAIN BUS OUT light should go off, and the NO. 2 MAIN BUS OUT light should remain on.

m. Place the NO. 2 MAIN INV switch in the "OFF" position, the WINDSHIELD INV switch in the "ON" position, and the BUS SELECT switch in the "115V NO. 1 MAIN" position.
RESULT: The AC voltmeter should indicate 115 (±6) volts, the frequency meter should indicate 400 (±10) cycles per second, and the warning lights should be as before.

n. Place the AC SYS switch in the "NORM" position.
RESULT: The NO. 1 MAIN BUS OUT light should go on.

o. Place the NO. 1 MAIN INV switch and the NO. 2 MAIN INV switch in the "ON" position.
RESULT: All warning lights should go off.

p. Apply a load of 2 (±0.5) KVA to each of the three inverters.

Note

Windshield temperature must be below 90°F to check windshield inverter loading.

RESULT: Under load, the output of each inverter should remain at 115(±6) volts and 400(±10) cycles per second.

q. Disconnect external power from the airplane, and restore all equipment to normal.

24.2.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO VOLTAGE - FREQUENCY NORMAL; NO. 1 MAIN		
Faulty AC voltmeter.	No. 2 and windshield inverters give same indications.	Replace.
NO FREQUENCY - VOLTAGE NORMAL; NO. 1 MAIN		
Faulty frequency meter.	No. 2 and windshield inverters give same indications.	

24.2.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO VOLTAGE - NO FREQUENCY; NO. 1 MAIN		
Fuse in meter circuit.	AC loads should operate normally.	Replace.
Faulty inverter.	Transfer to No. 2 main gives normal operation.	Replace.
Faulty inverter wiring.	Substitute No. 2 main in No. 1 position give same results.	Repair.
VOLTAGE AND/OR FREQUENCY FLUCTUATION NO. 1 MAIN		
Faulty inverter.	Transfer to No. 2 main gives normal operation.	Replace.
Faulty inverter wiring.	Substitution of No. 2 main in No. 1 position gives same results.	Repair.
VOLTAGE AND/OR FREQUENCY OUT OF LIMITS - NO. 1 MAIN		
Faulty AC meter.	No. 2 and windshield inverter give same indications.	Replace.
Improperly adjusted regulator.	Transfer to No. 2 main gives normal operation.	Adjust per para. 24.2.2.
Faulty inverter.	Transfer to No. 2 main gives normal operation.	Replace.
Faulty inverter wiring.	Substitution of No. 2 main in No. 1 position gives same results.	Repair.
BUS - OUT LIGHT ON; INVERTER SWITCH ON - NO. 1 MAIN		
Fault in meter circuit	Meters indicate no output; AC loads operable.	Replace.
Defective relay.	Meters and loads normal	Replace.
Faulty inverter	Transfer to No. 2 main gives normal operation.	Replace.
Faulty inverter wiring	Substitution of No. 2 main in No. 1 position gives same results.	Repair.

24.2.4 REMOVAL AND REPLACEMENT.

24.2.4.1 INVERTERS. To gain access to the inverter compartment for the removal of the airplane inverters, remove the center floor panel between fuselage stations 270.5 and 310.0.

Note

Particular care must be taken during removal and installation of inverters to preserve the buildup of washers, spacers, and bolts as shown in figure 24-23. Trim the bushing to a length of 1.25 inches before installing. Tighten the bolt until washers cannot be turned by hand, and then back off approximately one turn. turn.

INSPECTION NOTE

Inspect inverter mounting bolts for looseness or bottoming out against the bushing. The effectiveness of the mounting in preventing noise will be severely reduced if the bushing is pressed against the structure. If wear or aging of the rubber has caused looseness in the mount, the bushing may be shortened or washers may be added as required.

24.2.4.2 REMOVAL AND REPLACEMENT OF ARMATURE SHAFT BEARINGS ON MGE-23-34 AND MGE-23-340 INVERTERS. INVERTERS.

24.2.4.2.1 DISASSEMBLY. The removal of the bearings should be performed in accordance with the following instructions and in conjunction with figure 24-22.

a. Remove fan cover (1), two screws (2), and two washers (3).

b. Remove four screws (5), lift up on brush springs, and remove DC brushes (4) from their brush holders.

c. Remove the AC end cover (6), four screws (7), and four washers (8).

24.2.4.2.1 (Cont).

d. Remove four brush holder caps (9) and remove the AC brush assemblies (10 and 11).

e. Remove the lock nut (12), lock washers (13), fan (14), flat key (15) and a fan spacer (16) from the DC end of the armature shaft.

f. Remove the four screws (18) and lock washer (19). Remove the external bearing plate (17).

g. Remove four thru-bolt screws (22) and nuts (23) from opposite ends of the stator (67). Free stator leads from the DC shield assembly (36) and remove shield assembly from the stator. Tap the shield several times with a nonmetallic mallet to facilitate removal.

h. Remove the two screws (40), bearing plate (39), shim washer (41), loading spring (42) and shim washers (43) from the AC end shield assembly (64).

i. Remove the cover plate (44) from the AC end shield (64). Remove two screws (46) to free the stator leads from the AC end shield.

j. Remove the AC end shield assembly (64) from the stator (67). Tap the end shield several times with a fiber mallet to facilitate removal.

k. Remove the armature (66) from the DC end of the stator (67). Press ball bearings (37) and (65) from the shaft of armature (66). Remove an internal bearing plate (38) from the armature shaft.

CAUTION

Handle the armature carefully. Provide suitable protection for commutator, slip rings, and bearing journal surfaces.

24.2.4.2.2 INSPECTION.

a. AC shield assembly (64). The bearing housing must be smooth and free of scoring marks or fretting corrosion. The min/max I.D. of the bearing housing should be 1.3780/1.3788. Replace if necessary.

b. DC shield assembly (36). The bearing housing must be smooth and free of scoring marks or fretting corrosion. The min/max I.D. of the bearing housing should be 1.3780/1.3788. Replace if necessary.

c. Armature shaft (66). The bearing sections of the shaft must be smooth and free of scoring marks or fretting corrosion. The min/max O.D. of the bearing surface should be 0.5903/0.5906. Replace if necessary.

d. Before pressing on new ball bearing, make sure the shaft shoulder is square, free from burrs and undamaged. The shaft shoulder must be free from any roughness caused by the spinning of the bearing or the bearing thrust. Make sure that the undercut on the shaft shoulder is free from dirt or damage to the extent that it will prevent the bearing from setting against the shaft shoulder. The shaft chamber must not be damaged and must be free from burrs. The bearing surfaces of the shaft should be clean.

24.2.4.2.3. REASSEMBLY.

Note

Make sure the internal bearing plate (38) is in position at the DC end of the armature before assembling ball bearings.

a. Press new ball bearings, Leland No. A-15884 (37) and (65) on both ends of the shaft of armature assembly (66). An arbor press or pressure sealing type press may be used. Apply pressure to the inner race by means of a fixture which insures a square seat on the bearing ring. Release the pressure immediately after the bearing is in place in order to avoid bending the shaft.

Note

The aligning fixture must be made in such a way that the force is applied squarely on the face of the bearing and exactly in the same perpendicular axis as the shaft axis without wobbling. This is required to avoid pressing the bearing on in such a manner that the bearing I.D. is actually shaving the shaft slightly due to a side force of the press or the aligning fixture. The fixture must be inspected frequently, as no burrs or improper seating in the bearing races must be allowed. The two faces of the fixture for pressing the bearing must be parallel and of smooth finish. Make sure that the tool will fit only over the steel section of the inner ring. It must not touch any part of the shield and/or contact seals. Precautions must be taken to avoid moving of the tool in the radial direction of the bearing.

b. Work the AC end shield assembly (64) over the shaft of the armature and onto the stator (67). Tap the shield lightly with a fiber mallet to seat.

c. Install the armature assembly (66) in the stator (67) from the AC end. Work the DC end shield assembly (36) in position over the bearing (37) and onto the stator (67). Tap the end shield with a fiber mallet to fully seat it. Position the external bearing plate (17) over armature shaft and the DC shield assembly (36). Install four screws (18) with lock washers (19) through the bearing plate (17) and shield (36) and into the internal bearing plate (38). Tighten and lock wire the screws (18).

24.2.4.2.3 (Cont).

d. Install thru-bolt screws (22) and nuts (23) at AC end of unit. Torque thru-bolts evenly 40 to 50 IN. LB at room temperature.

e. Connect stator leads to the AC end shield assembly (64) using two screws (46). Replace cover plate (44) to AC end shield assembly (64).

f. Place loading spring (42) on the AC end of the armature shaft and measure the "A" distance as indicated in figure 24-22, detail B. Add 0.010 to 0.015 inch to the "A" reading to obtain the required shim thickness. Remove the loading spring (42) and add shim washers (43); replace loading spring. Position shim washers (41) and bearing plate (39), and fasten in place with two screws (40).

CAUTION

Failure to properly shim the loading spring results in incorrect bearing forces, and ultimately causes premature bearing failure.

g. Check that the armature rotates freely when turned by hand.

h. Install DC brushes by lifting brush tension springs and installing four new DC brush assemblies (4) in the brush holder assemblies. The face of the brushes must fit the contour of the commutator. Attach each set of brush leads to its brush holder with a screw (5).

i. Install straight AC brushes (10) and angle brush assembly (11) in the AC end shield. Assemble a brush holder cap (9) at each brush holder. To the DC end of the armature shaft, assemble in order, a fan spacer (16), flat key (15), fan (14), new lock washer (13) and a lock nut (12). Bend the tabs of lock washer (13) to lock the nut (12).

j. Position the AC end cover (6) on the AC shield assembly and attach with four screws (7) and washers (8).

k. Assemble the fan cover (1) to the DC end shield with two screws (2) and washers (3).

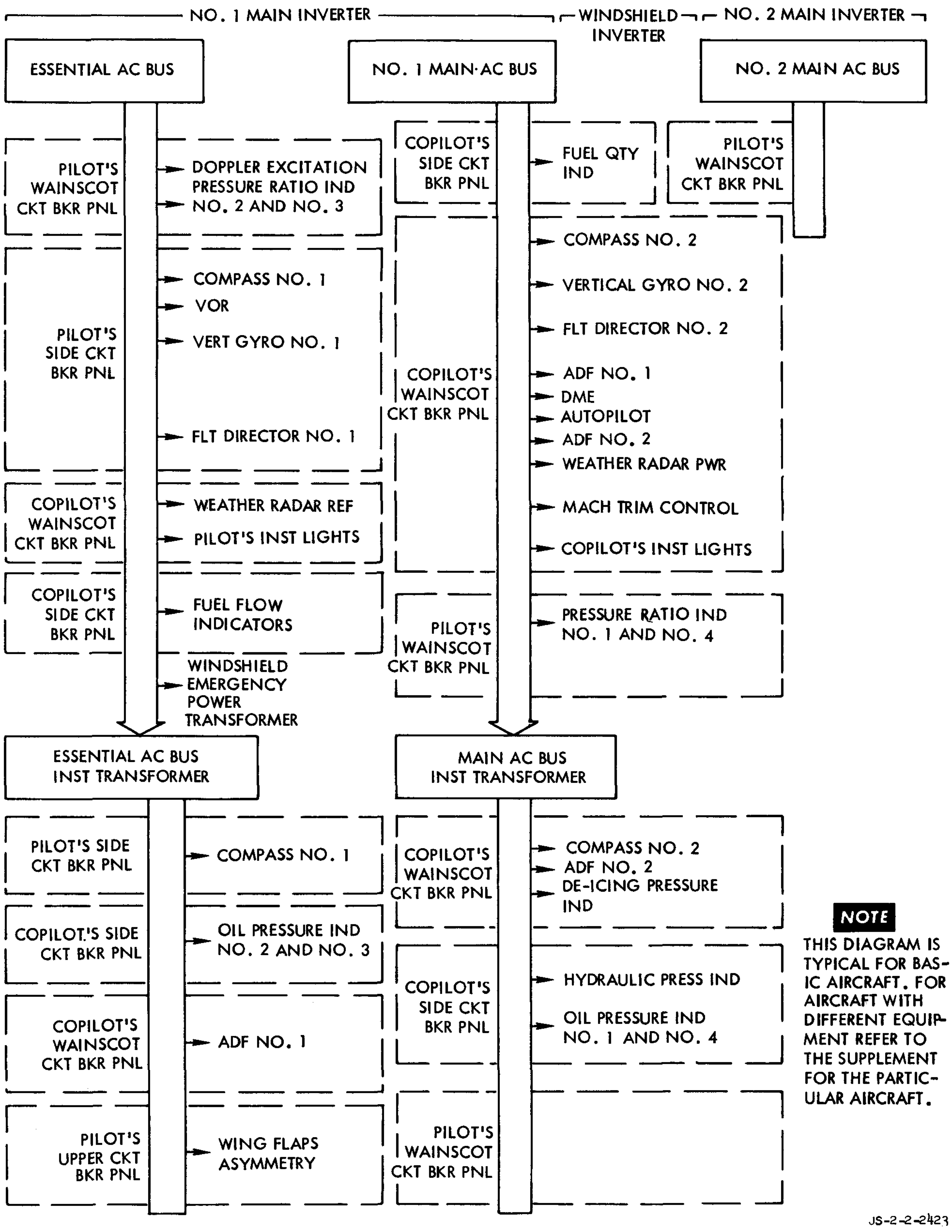
					(M-2 & W FAILED)	(M-1 & M-2 FAILED)	(M-1 & W FAILED)	(M-1 & W FAILED)
Switch Positions	NO. 1 MAIN INV SWITCH	ON	ON	TRANSFER	ON	TRANSFER	ON, OFF	TRANSFER
	NO. 2 MAIN INV SWITCH	ON	ON	ON	ON-OFF	ON-OFF	ON	ON
	WINDSHIELD INV SWITCH	ON	TRANSFER	ON	ON, OFF, TRANSFER	ON	TRANSFER	ON, OFF, TRANSFER
	AC SYS SWITCH	NORMAL	NORMAL	NORMAL	*NORMAL	EMER	EMER	*NORMAL
Inverter Loads	ESS AC LOADS	M-1	M-1	M-2	M-1	W	M-2	M-2
	NO. 1 MAIN AC LOADS	M-1	M-1	M-2				
	NO. 2 MAIN AC LOADS	M-2						
	WINDSHIELD LOADS	W	M-2	W				
	ESS WS LOADS	W	M-2	W	M-1	W	M-2	M-2

NOTE: In the Inverter Load columns, M-1, M-2, and W stand for the No. 1 main, the No. 2 main, and the windshield inverters, respectively. The particular AC loads are supplied by the inverter specified when the control switches are in the positions given to the left.

*WINDSHIELD ANTI-ICING POWER CONTROL switch in "EMER" position

JS-2-2-2422

Figure 24-16. Inverter Load Transfer

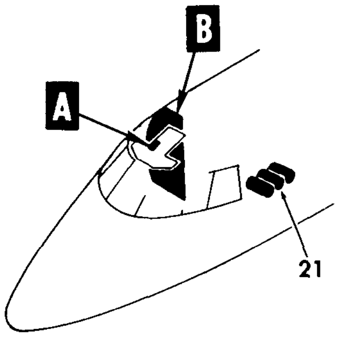

NO. 1 MAIN INVERTER
WINDSHIELD INVERTER
NO. 2 MAIN INVERTER
ESSENTIAL AC BUS
NO. 1 MAIN AC BUS
NO. 2 MAIN AC BUS
PILOT'S WAINSCOT CKT BKR PNL
DOPPLER EXCITATION
PRESSURE RATIO IND NO. 2 AND NO. 3
COPILOT'S SIDE CKT BKR PNL
FUEL QTY IND
PILOT'S WAINSCOT CKT BKR PNL
PILOT'S SIDE CKT BKR PNL
COMPASS NO. 1
VOR
VERT GYRO NO. 1
FLT DIRECTOR NO. 1
COPILOT'S WAINSCOT CKT BKR PNL
COMPASS NO. 2
VERTICAL GYRO NO. 2
FLT DIRECTOR NO. 2
ADF NO. 1
DME
AUTOPILOT
ADF NO. 2
WEATHER RADAR PWR
MACH TRIM CONTROL
COPILOT'S INST LIGHTS
COPILOT'S WAINSCOT CKT BKR PNL
WEATHER RADAR REF
PILOT'S INST LIGHTS
COPILOT'S SIDE CKT BKR PNL
FUEL FLOW INDICATORS
PILOT'S WAINSCOT CKT BKR PNL
PRESSURE RATIO IND NO. 1 AND NO. 4
WINDSHIELD EMERGENCY POWER TRANSFORMER
ESSENTIAL AC BUS INST TRANSFORMER
MAIN AC BUS INST TRANSFORMER
PILOT'S SIDE CKT BKR PNL
COMPASS NO. 1
COPILOT'S WAINSCOT CKT BKR PNL
COMPASS NO. 2
ADF NO. 2
DE-ICING PRESSURE IND
COPILOT'S SIDE CKT BKR PNL
OIL PRESSURE IND NO. 2 AND NO. 3
COPILOT'S SIDE CKT BKR PNL
HYDRAULIC PRESS IND
OIL PRESSURE IND NO. 1 AND NO. 4
COPILOT'S WAINSCOT CKT BKR PNL
ADF NO. 1
PILOT'S UPPER CKT BKR PNL
WING FLAPS ASYMMETRY
PILOT'S WAINSCOT CKT BKR PNL
JS-2-2-2423

NOTE

THIS DIAGRAM IS TYPICAL FOR BASIC AIRCRAFT. FOR AIRCRAFT WITH DIFFERENT EQUIPMENT REFER TO THE SUPPLEMENT FOR THE PARTICULAR AIRCRAFT.

Figure 24-17. AC Power Distribution Diagram

1. NO. 1 MAIN/WINDSHIELD BUS TRANSFER RELAY
2. NO. 1 MAIN BUS DISCONNECT RELAY
3. WINDSHIELD/NO. 1 MAIN TRANSFER RELAY NO. 2
4. NO. 2 MAIN INVERTER OUTPUT RELAY
5. WINDSHIELD INVERTER OUTPUT RELAY
6. WINDSHIELD/NO. 1 MAIN TRANSFER RELAY NO. 1
7. NO. 1 MAIN INVERTER OUTPUT RELAY
8. FUSES
9. ESSENTIAL BUS INSTRUMENT TRANSFORMER
10. MAIN BUS INSTRUMENT TRANSFORMER
11. WINDSHIELD INVERTER DC POWER RELAY
12. NO. 1 MAIN INVERTER DC INPUT RELAY
13. NO. 2 MAIN INVERTER DC INPUT RELAY
14. WINDSHIELD INVERTER WARNING LIGHT RELAY
15. ESSENTIAL BUS WARNING LIGHT RELAY
16. NO. 2 MAIN BUS WARNING LIGHT RELAY
17. WINDSHIELD WARNING LIGHT RELAY RESISTOR
18. AC POWER SUPPLY CIRCUIT BREAKERS
19. NO. 1 MAIN TRANSFER CONTROL RELAY
20. WINDSHIELD EMERGENCY POWER TRANSFER RELAY
21. INVERTERS



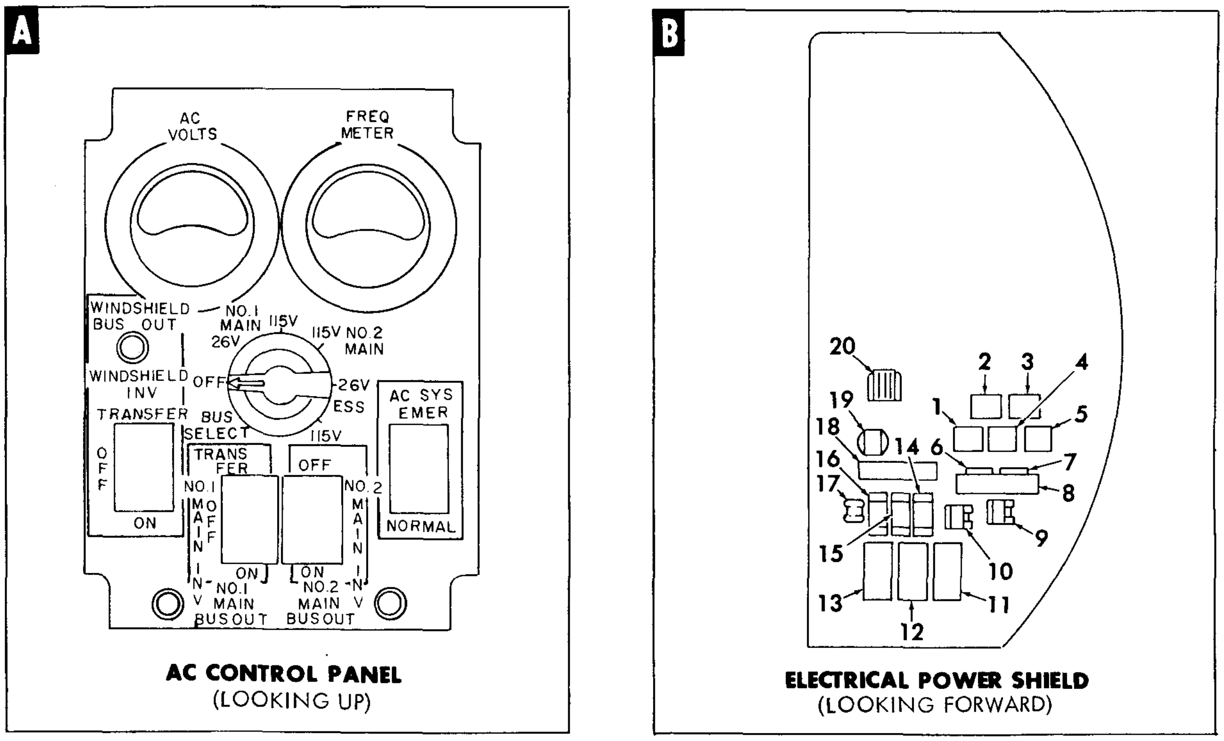


Figure 24-18. AC Power Supply Components (Airplanes 5003, 5005, 5010 and Up)

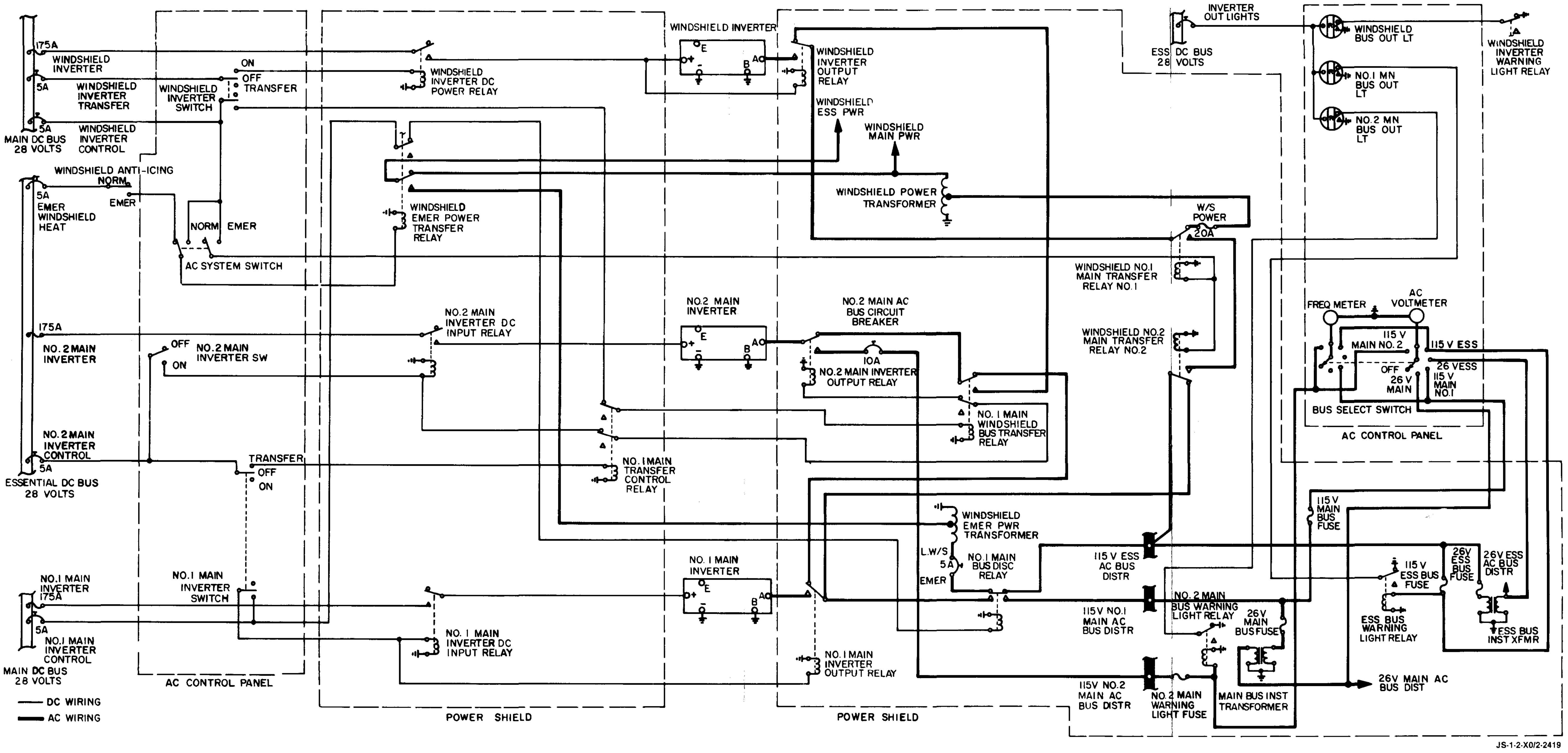

175A
WINDSHIELD INVERTER
5A
WINDSHIELD INVERTER TRANSFER
5A
WINDSHIELD INVERTER CONTROL
MAIN DC BUS 28 VOLTS
ON
OFF
TRANSFER
WINDSHIELD INVERTER SWITCH
WINDSHIELD INVERTER DC POWER RELAY
WINDSHIELD INVERTER
E
A
B
WINDSHIELD INVERTER OUTPUT RELAY
WINDSHIELD ESS PWR
WINDSHIELD MAIN PWR
WINDSHIELD POWER TRANSFORMER
W/S POWER
20A
INVERTER OUT LIGHTS
ESS DC BUS 28 VOLTS
WINDSHIELD BUS OUT LT
NO.1 MN BUS OUT LT
NO.2 MN BUS OUT LT
WINDSHIELD INVERTER WARNING LIGHT RELAY
WINDSHIELD ANTI-ICING
NORM
5A
EMER WINDSHIELD HEAT
EMER
NORM
EMER
AC SYSTEM SWITCH
T
WINDSHIELD EMER POWER TRANSFER RELAY
WINDSHIELD NO.1 MAIN TRANSFER RELAY NO.1
WINDSHIELD NO.2 MAIN TRANSFER RELAY NO.2
175A
NO.2 MAIN INVERTER
OFF
ON
NO.2 MAIN INVERTER SW
NO.2 MAIN INVERTER DC INPUT RELAY
NO.2 MAIN INVERTER
NO.2 MAIN AC BUS CIRCUIT BREAKER
10A
NO.2 MAIN INVERTER OUTPUT RELAY
NO. 1 MAIN WINDSHIELD BUS TRANSFER RELAY
FREQ METER
AC VOLTMETER
115 V MAIN NO.2
115 V ESS
OFF
26 VESS
26 V MAIN
115 V MAIN NO.1
BUS SELECT SWITCH
AC CONTROL PANEL
NO.2 MAIN INVERTER CONTROL
5A
ESSENTIAL DC BUS 28 VOLTS
TRANSFER
OFF
ON
NO. 1 MAIN TRANSFER CONTROL RELAY
WINDSHIELD EMER PWR TRANSFORMER
L.W/S
5A
EMER
NO.1 MAIN BUS DISC RELAY
115V MAIN BUS FUSE
115 V ESS AC BUS DISTR
115 V ESS BUS FUSE
26V ESS BUS FUSE
26V ESS AC BUS DISTR
ESS BUS WARNING LIGHT RELAY
ESS BUS INST XFMR
NO.1 MAIN INVERTER
175A
5A
NO.1 MAIN INVERTER CONTROL
MAIN DC BUS 28 VOLTS
NO.1 MAIN INVERTER SWITCH
NO. 1 MAIN INVERTER DC INPUT RELAY
NO. 1 MAIN INVERTER
NO.1 MAIN INVERTER OUTPUT RELAY
115V NO.1 MAIN AC BUS DISTR
NO.2 MAIN BUS WARNING LIGHT RELAY
26V MAIN BUS FUSE
115V NO.2 MAIN AC BUS DISTR
NO.2 MAIN WARNING LIGHT FUSE
MAIN BUS INST TRANSFORMER
26V MAIN AC BUS DIST
AC CONTROL PANEL
POWER SHIELD
POWER SHIELD
DC WIRING
AC WIRING
JS-1-2-X0/2-2419

Figure 24-19. AC Power Supply Schematic Diagram (Airplanes 5141 and Up)

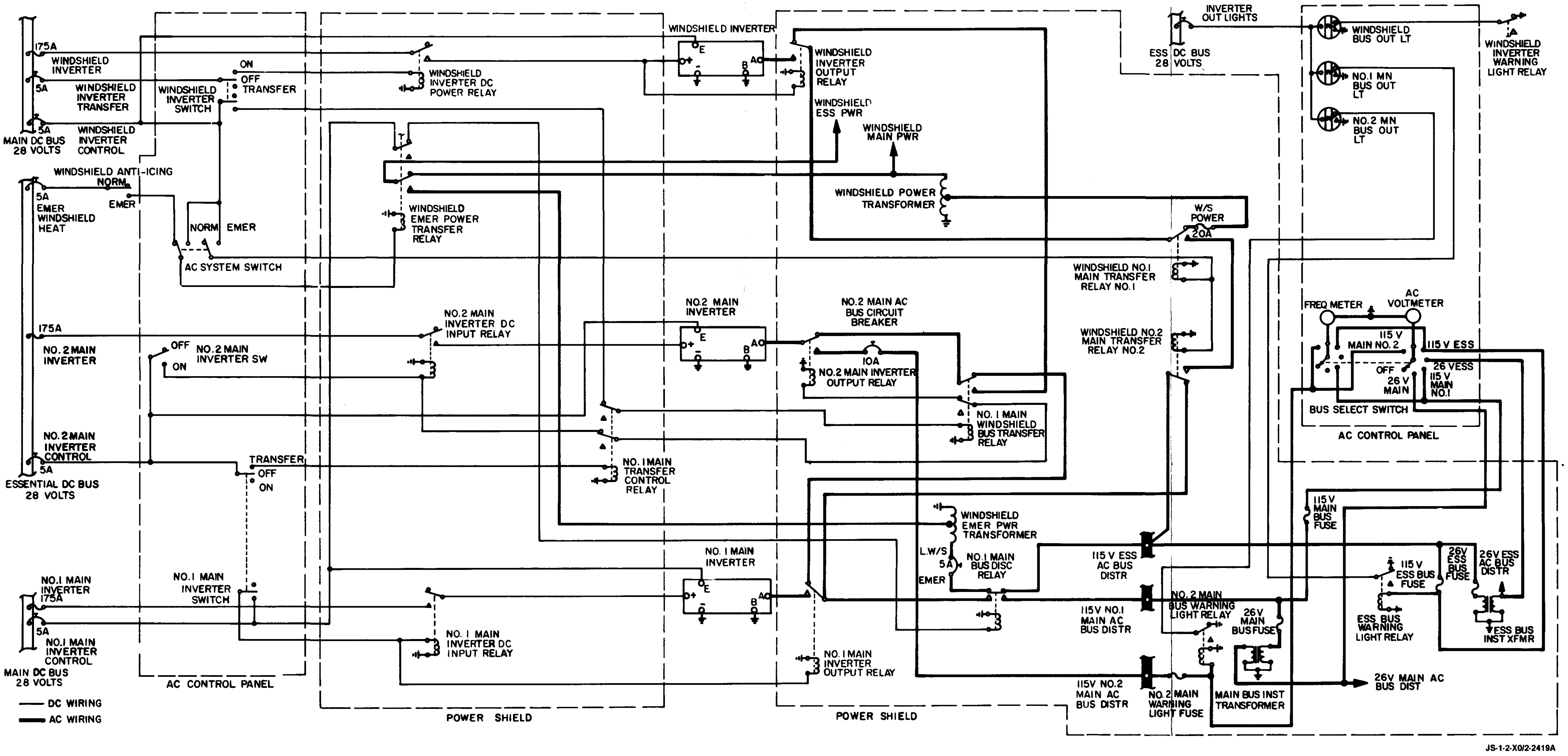

175A
WINDSHIELD INVERTER
5A
WINDSHIELD INVERTER TRANSFER
5A
WINDSHIELD INVERTER CONTROL
MAIN DC BUS 28 VOLTS
WINDSHIELD INVERTER SWITCH
ON
OFF
TRANSFER
WINDSHIELD INVERTER DC POWER RELAY
WINDSHIELD INVERTER
E
B
AC
WINDSHIELD INVERTER OUTPUT RELAY
WINDSHIELD ESS PWR
WINDSHIELD MAIN PWR
WINDSHIELD POWER TRANSFORMER
WINDSHIELD ANTI-ICING
NORM
EMER
5A
EMER WINDSHIELD HEAT
AC SYSTEM SWITCH
WINDSHIELD EMER POWER TRANSFER RELAY
175A
NO. 2 MAIN INVERTER
OFF
ON
NO. 2 MAIN INVERTER SW
NO. 2 MAIN INVERTER DC INPUT RELAY
NO. 2 MAIN INVERTER
NO. 2 MAIN AC BUS CIRCUIT BREAKER
10A
NO. 2 MAIN INVERTER OUTPUT RELAY
NO. 1 MAIN WINDSHIELD BUS TRANSFER RELAY
NO. 2 MAIN INVERTER CONTROL
5A
ESSENTIAL DC BUS 28 VOLTS
TRANSFER
OFF
ON
NO. 1 MAIN TRANSFER CONTROL RELAY
WINDSHIELD EMER PWR TRANSFORMER
L.W/S 5A
EMER
NO. 1 MAIN BUS DISC RELAY
NO. 1 MAIN INVERTER
NO. 1 MAIN INVERTER 175A
5A
NO. 1 MAIN INVERTER CONTROL
MAIN DC BUS 28 VOLTS
NO. 1 MAIN INVERTER SWITCH
NO. 1 MAIN INVERTER DC INPUT RELAY
NO. 1 MAIN INVERTER OUTPUT RELAY
AC CONTROL PANEL
POWER SHIELD
POWER SHIELD
DC WIRING
AC WIRING
INVERTER OUT LIGHTS
ESS DC BUS 28 VOLTS
WINDSHIELD BUS OUT LT
NO. 1 MN BUS OUT LT
NO. 2 MN BUS OUT LT
WINDSHIELD INVERTER WARNING LIGHT RELAY
W/S POWER
20A
WINDSHIELD NO. 1 MAIN TRANSFER RELAY NO. 1
WINDSHIELD NO. 2 MAIN TRANSFER RELAY NO. 2
FREQ METER
AC VOLTMETER
115 V MAIN NO. 2
115 V ESS
OFF
26 VESS
26 V MAIN
115 V MAIN NO. 1
BUS SELECT SWITCH
AC CONTROL PANEL
115 V MAIN BUS FUSE
115 V ESS AC BUS DISTR
115V NO. 1 MAIN AC BUS DISTR
115V NO. 2 MAIN AC BUS DISTR
NO. 2 MAIN BUS WARNING LIGHT RELAY
26V MAIN BUSFUSE
NO. 2 MAIN WARNING LIGHT FUSE
MAIN BUS INST TRANSFORMER
115 V ESS BUS FUSE
26V ESS BUS FUSE
26V ESS AC BUS DISTR
ESS BUS WARNING LIGHT RELAY
ESS BUS INST XFMR
26V MAIN AC BUS DIST
JS-1-2-X0/2-2419A

Figure 24-19A. AC Power Supply Schematic Diagram (Airplanes 5003, 5046 thru 5140)

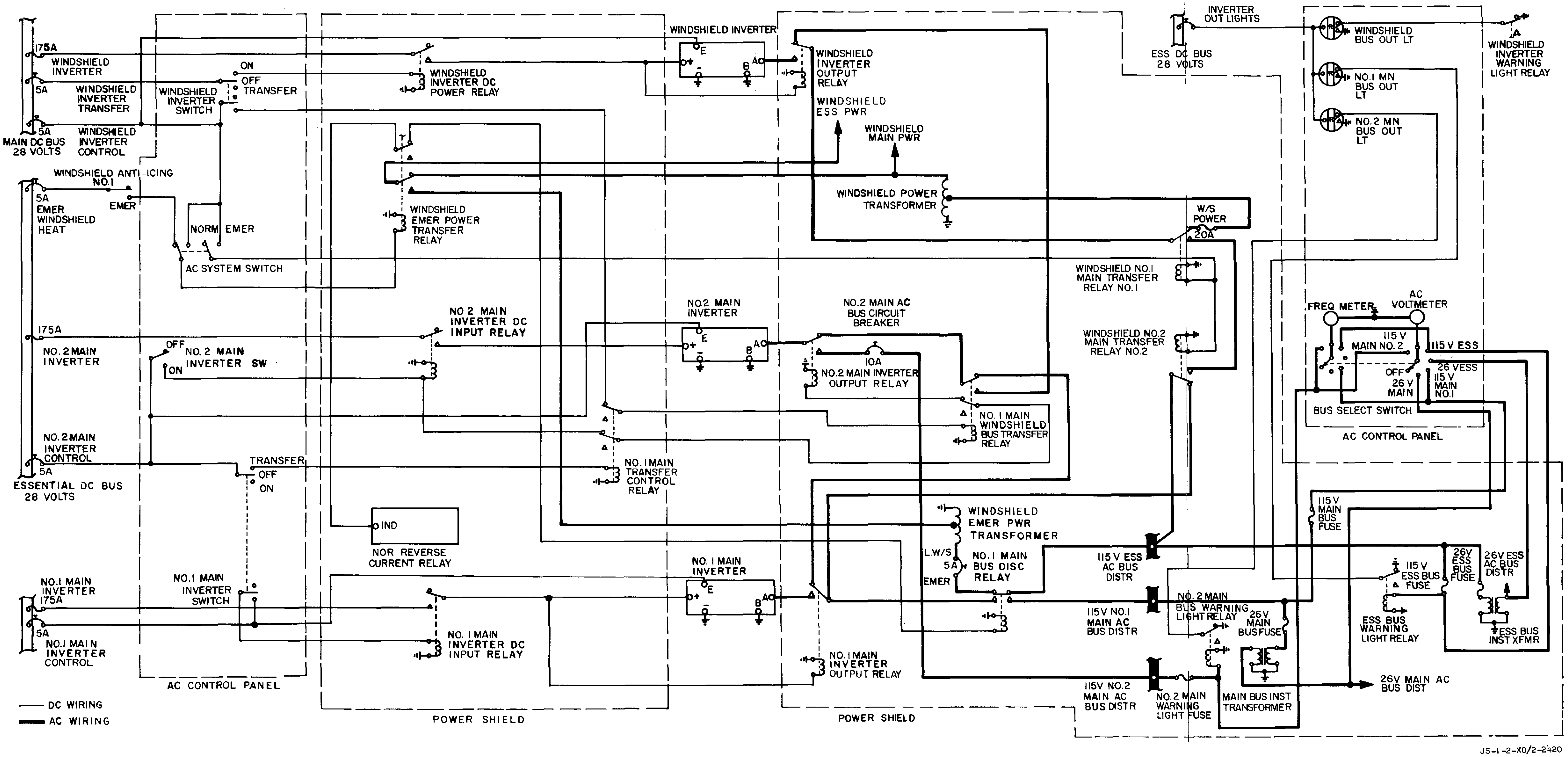

175A
WINDSHIELD INVERTER
5A
WINDSHIELD INVERTER TRANSFER
5A
WINDSHIELD INVERTER CONTROL
MAIN DC BUS 28 VOLTS
WINDSHIELD INVERTER SWITCH
ON
OFF
TRANSFER
WINDSHIELD INVERTER
E
+
-
B
AC
WINDSHIELD INVERTER DC POWER RELAY
WINDSHIELD INVERTER OUTPUT RELAY
WINDSHIELD ESS PWR
WINDSHIELD MAIN PWR
WINDSHIELD POWER TRANSFORMER
INVERTER OUT LIGHTS
ESS DC BUS 28 VOLTS
WINDSHIELD BUS OUT LT
NO.1 MN BUS OUT LT
NO.2 MN BUS OUT LT
WINDSHIELD INVERTER WARNING LIGHT RELAY
WINDSHIELD ANTI-ICING NO.1
5A
EMER WINDSHIELD HEAT
EMER
NORM
EMER
AC SYSTEM SWITCH
WINDSHIELD EMER POWER TRANSFER RELAY
W/S POWER 20A
WINDSHIELD NO.1 MAIN TRANSFER RELAY NO.1
WINDSHIELD NO.2 MAIN TRANSFER RELAY NO.2
175A
NO.2 MAIN INVERTER
OFF
ON
NO. 2 MAIN INVERTER SW
NO 2 MAIN INVERTER DC INPUT RELAY
NO.2 MAIN INVERTER
NO.2 MAIN AC BUS CIRCUIT BREAKER
10A
NO.2 MAIN INVERTER OUTPUT RELAY
NO. 1 MAIN WINDSHIELD BUS TRANSFER RELAY
FREQ METER
AC VOLTMETER
115 V MAIN NO. 2
115 V ESS
26 VESS
OFF
26 V MAIN
115 V MAIN NO.1
BUS SELECT SWITCH
AC CONTROL PANEL
NO.2 MAIN INVERTER CONTROL
5A
ESSENTIAL DC BUS 28 VOLTS
TRANSFER
OFF
ON
NO.1 MAIN TRANSFER CONTROL RELAY
IND
NOR REVERSE CURRENT RELAY
WINDSHIELD EMER PWR TRANSFORMER
L.W/S
5A
EMER
NO.1 MAIN BUS DISC RELAY
115 V ESS AC BUS DISTR
115V MAIN BUS FUSE
115 V ESS BUS FUSE
26V ESS BUS FUSE
26V ESS AC BUS DISTR
ESS BUS WARNING LIGHT RELAY
ESS BUS INST XFMR
NO.1 MAIN INVERTER 175A
5A
NO.1 MAIN INVERTER CONTROL
NO.1 MAIN INVERTER SWITCH
NO.1 MAIN INVERTER DC INPUT RELAY
NO. 1 MAIN INVERTER
NO.1 MAIN INVERTER OUTPUT RELAY
115V NO.1 MAIN AC BUS DISTR
NO. 2 MAIN BUS WARNING LIGHT RELAY
26V MAIN BUS FUSE
115V NO.2 MAIN AC BUS DISTR
NO.2 MAIN WARNING LIGHT FUSE
MAIN BUS INST TRANSFORMER
26V MAIN AC BUS DIST
AC CONTROL PANEL
POWER SHIELD
POWER SHIELD
DC WIRING
AC WIRING
JS-1-2-X0/2-2420

Figure 24-20. AC Power Supply Schematic Diagram (Airplanes 5005, 5011 thru 5016, 5018 thru 5021, 5023, 5025, and 5029)

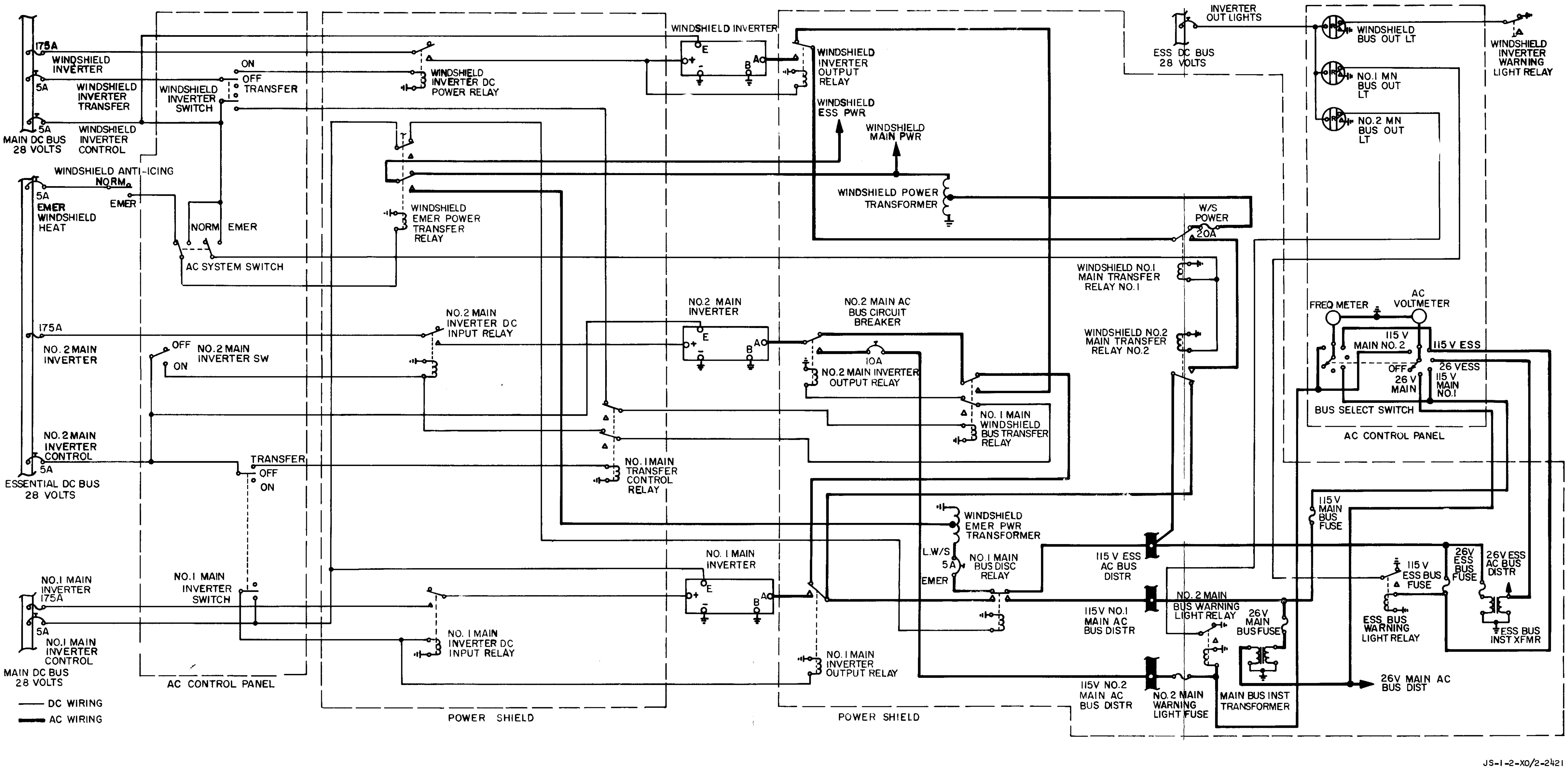

175A
WINDSHIELD INVERTER
5A
WINDSHIELD INVERTER TRANSFER
5A
WINDSHIELD INVERTER CONTROL
MAIN DC BUS 28 VOLTS
WINDSHIELD INVERTER SWITCH
ON
OFF
TRANSFER
WINDSHIELD INVERTER DC POWER RELAY
WINDSHIELD INVERTER
E
+
-
B
AC
WINDSHIELD INVERTER OUTPUT RELAY
WINDSHIELD ESS PWR
WINDSHIELD MAIN PWR
WINDSHIELD POWER TRANSFORMER
INVERTER OUT LIGHTS
ESS DC BUS 28 VOLTS
WINDSHIELD BUS OUT LT
NO.1 MN BUS OUT LT
NO.2 MN BUS OUT LT
WINDSHIELD INVERTER WARNING LIGHT RELAY
WINDSHIELD ANTI-ICING
NORM
EMER
5A
EMER WINDSHIELD HEAT
NORM
EMER
AC SYSTEM SWITCH
WINDSHIELD EMER POWER TRANSFER RELAY
W/S POWER
20A
WINDSHIELD NO.1 MAIN TRANSFER RELAY NO.1
WINDSHIELD NO.2 MAIN TRANSFER RELAY NO.2
FREQ METER
AC VOLTMETER
115 V MAIN NO. 2
115 V ESS
OFF
26 V MAIN
26 VESS
115 V MAIN NO.1
BUS SELECT SWITCH
AC CONTROL PANEL
175A
NO. 2 MAIN INVERTER
OFF
ON
NO.2 MAIN INVERTER SW
NO.2 MAIN INVERTER DC INPUT RELAY
NO.2 MAIN INVERTER
NO.2 MAIN AC BUS CIRCUIT BREAKER
10A
NO.2 MAIN INVERTER OUTPUT RELAY
NO. 1 MAIN WINDSHIELD BUS TRANSFER RELAY
NO.2 MAIN INVERTER CONTROL
5A
ESSENTIAL DC BUS 28 VOLTS
TRANSFER
OFF
ON
NO. 1 MAIN TRANSFER CONTROL RELAY
WINDSHIELD EMER PWR TRANSFORMER
L.W/S 5A
EMER
NO.1 MAIN BUS DISC RELAY
115 V ESS AC BUS DISTR
115V MAIN BUS FUSE
115 V ESS BUS FUSE
26V ESS BUS FUSE
26V ESS AC BUS DISTR
ESS BUS WARNING LIGHT RELAY
ESS BUS INST XFMR
NO.1 MAIN INVERTER
175A
5A
NO.1 MAIN INVERTER CONTROL
MAIN DC BUS 28 VOLTS
NO.1 MAIN INVERTER SWITCH
AC CONTROL PANEL
NO. 1 MAIN INVERTER DC INPUT RELAY
NO. 1 MAIN INVERTER
NO. 1 MAIN INVERTER OUTPUT RELAY
115V NO.1 MAIN AC BUS DISTR
NO. 2 MAIN BUS WARNING LIGHT RELAY
26V MAIN BUS FUSE
115V NO.2 MAIN AC BUS DISTR
NO.2 MAIN WARNING LIGHT FUSE
MAIN BUS INST TRANSFORMER
26V MAIN AC BUS DIST
POWER SHIELD
POWER SHIELD
DC WIRING
AC WIRING
JS-1-2-X0/2-2421

Figure 24-21. AC Power Supply Schematic Diagram (Airplanes 5033, 5035, thru 5040)

1. FAN COVER
2. SCREW
3. WASHER
4. DC BRUSH
5. SCREW
6. AC END COVER
7. SCREW
8. WASHER
9. BRUSH HOLDER CAP
10. STRAIGHT BRUSH ASSEMBLY
11. ANGLE BRUSH ASSEMBLY
12. LOCK NUT
13. LOCK WASHER
14. ELECTRICAL ROTATING EQUIPMENT FAN
15. FLAT KEY
16. FAN SPACER
17. EXTERNAL BEARING PLATE
18. SCREW
19. WASHER
20. SCREW
21. DC END BELL UNIT
22. THRU-BOLT SCREW
23. NUT
24. SCREW
25. LEAD ASSEMBLY
26. CAPACITOR ASSEMBLY
27. SCREW
28. CAPACITOR CLIP
29. SCREW
30. WASHER
31. BUSHING
32. SPRING
33. BRUSH SPRING PIN
34. SET SCREW
35. BRUSH HOLDER
36. DC SHIELD ASSEMBLY
37. BALL BEARING
38. INTERNAL BEARING PLATE
39. BEARING PLATE
40. SCREW
41. SHIM WASHER
42. LOADING SPRING
43. SHIM WASHER
44. AC END COVER PLATE
45. SCREW
46. SCREW
47. WASHER
48. WASHER
49. AC END SHIELD
50. RESISTOR ASSEMBLY
51. SCREW
52. WASHER
53. CAPACITOR ASSEMBLY
54. SCREW
55. WASHER
56. CAPACITOR CLAMP
57. AIR DUCT
58. SCREW
59. WASHER
60. LEAD ASSEMBLY
61. SCREW
62. WASHER
63. WASHER
64. AC END SHIELD ASSEMBLY
65. BALL BEARING
66. ARMATURE ASSEMBLY
67. STATOR

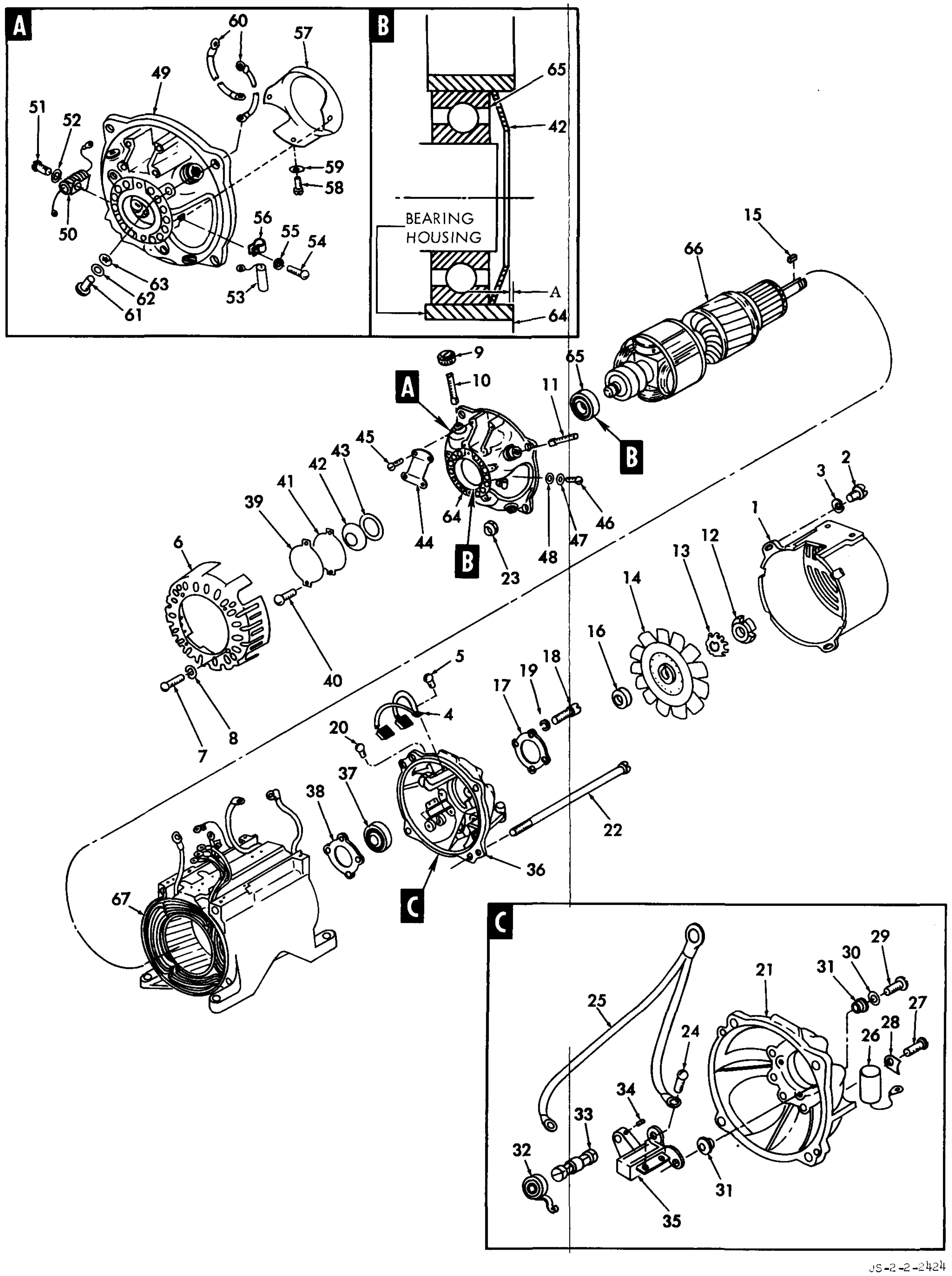


Figure 24-22. Motor-Generator Assembly (Inverter)

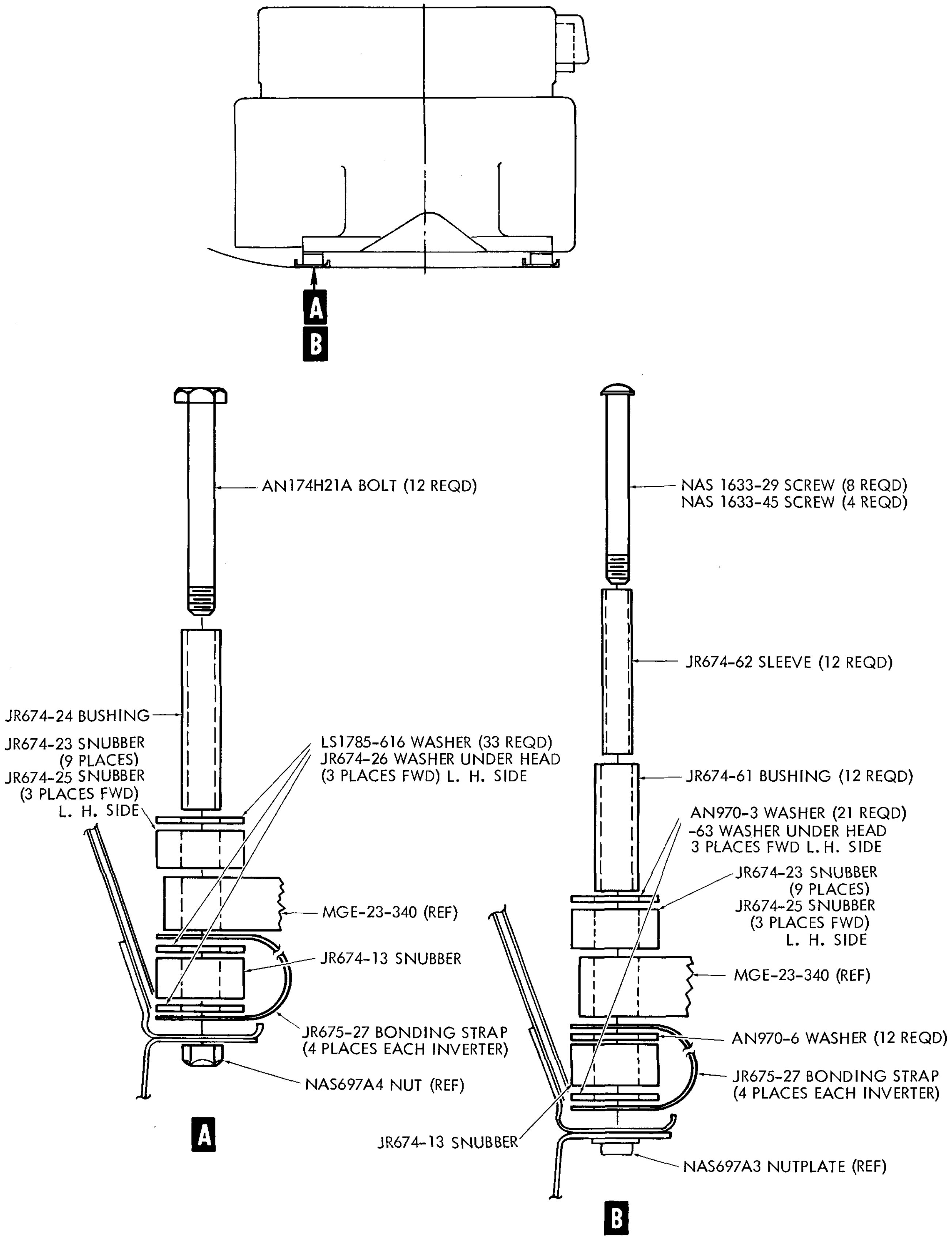


Figure 24-23. Inverter Installation

SECTION 24.3
RECOMMENDED WIRING PRACTICES

Contents Page

RECOMMENDED WIRING PRACTICES 24-69
Terminal Lugs 24-69
Splices 24-69
Connectors 24-69
Insulation of Terminations 24-70
Termination of Shields 24-70
Bundle and Routing Requirements 24-71
Proximity to Flammable Fluid Lines 24-71
Clamping 24-71
Wire Connections 24-71

FIGURES
24-25. Floating Shield Termination 24-70

24.3.1 RECOMMENDED WIRING PRACTICES. The following recommendations are made to indicate the wiring standards used to a large extent, but not exhaustively, for the production JetStar aircraft. Where special situations require more extensive requirements, these will be found in the file of type design data at the manufacturer.

24.3.2 TERMINAL LUGS. Crimped solderless, preinsulated type terminal lugs are generally used throughout the aircraft. Wire terminal lugs should not be bent more than 45 degrees. Bent lugs shall not be straightened or rebent. Double wire termination is sometimes approved. This provides two wires terminated in a single lug. GS071B or GSC128T sleeve must first be properly crimped to the two wires before inserting in the lug. This table indicates the wire and terminal lug combinations to be used:

DUAL WIRE TERMINATIONS

WIRE SIZE	LUG SIZE
22ga & 22ga	14-16
22ga & 20ga	14-16
22ga & 18ga	14-16
22ga & 16ga	14-16
20ga & 20ga	14-16
20ga & 18ga	14-16
20ga & 16ga	10-12
18ga & 18ga	10-12
18ga & 16ga	10-12
16ga & 16ga	10-12

The amount of exposed conductor after stripping shall be held to a minimum. Only sufficient conductor to insert full length of crimp barrel is required. No restaking or reswaging of terminal lugs is acceptable.

24.3.3 SPLICES. Splices shall not be made by twisting and soldering. No splice shall be located under wire harness clamps. Quality crimp splice connectors with proper crimping is recommended. Splices in unpressurized areas shall be waterproofed by (1) use of a waterproof splice, (2) application of tank sealant at both ends of a preinsulated regular splice, or (3) application of SCL thermofit tubing over splice.

24.3.4 CONNECTORS. Connectors shall be installed in accordance with the connector manufacturer's recommendations. Wires soldered to connector contacts shall not have their insulation charred. Connector contacts shall be in correct alignment after assembly. When fire zone wire is terminated at "K" type connectors and the wire insulation's outside diameter is greater than the opening in insulator (insert), it shall be terminated by removing the outer nylon jacket, crimping pin to wire, and overcoating trimmed section of wire insulation with insulating varnish. Unused pins in rubber grommets of Amphenol 69 connectors shall be plugged with MS25251 (-16, -12, -8) end seals. MS3420 bushing shall be sized so that there will be no crushing of the wire insulation.

The Amphenol 69 series connectors are no longer available and can be replaced with MS3450W( )-( )( ) ( ) receptacle & MS3456W( )-( )( )( ) plug. A backshell clamp MS3416-( ) EC must also be used for moisture sealing. The following examples demonstrate

24.3.4 (Cont)

callout comparison. Note that the "100" suffix is not needed and that the dash is omitted whenever short contact shell sizes 8S through 16S are required.

Examples:

69-0R22-14PW100	receptacle is replaced with
MS3450W22-14PW	receptacle and
MS3416-22EC	backshell clamp
69-6R14S-2SX100	plug is replaced with
MS3456W14S2SX	plug and
MS3416-14EC	backshell clamp

24.3.5 INSULATION OF TERMINATIONS. Insulating sleeving and tape is used to protect electrical wiring against short circuit and corrosion as follows:

(1) All lugs not preinsulated.

(2) Termination at connectors not otherwise insulated.

(3) Connectors with rubber grommet fitted pins need no other insulation added.

(4) Completed connections with less than 0.12 inch air gap shall be insulated with heat shrinkable sleeving.

(5) Connections to resistors over 5 watts require insulation approved for 400°F.

(6) Potted connections require no additional insulation.

(7) Shield pigtails and/or splices.

(8) Continuous bare wire jumpers for interconnection shall have insulating sleeves between connections.

(9) Ferrules shall be insulated with heat shrinkable tubing except inside fuel tanks.

Vinyl tubing shall not be used in areas exceeding 200°F, polyolefin and Kynar tubing shall not be used in areas exceeding 300°F and TFE tubing will be used where temperature exceeds 300°F. SCL thermofit shall be used only for waterproofing applications.

Use the following heat levels for shrinking tubing:

All except TFE	275 to 360°F
TFE tubing	625-725°F

24.3.6 TERMINATION OF SHIELDS. Cable shields shall be terminated with appropriate ferrule or solder sleeve and insulated with heat shrink sleeving. Ferrule size shall be the smallest that will accept the inner conductors. In pylon and nacelle area and within 4 inches of heat exchanger, use sleeving suitable for 400°F.

Shielding termination shall be accomplished so that all loose strands are secure with a minimum of cutting and breaking. When shields are to be grounded, wire jumpers shall be attached to the shield by a ferrule or solder sleeve and used as pigtails. Where common connections are required between two or more shields, an additional jumper may be added. When termination is at a connector or cable clamp adapter, the pigtail shall break away from cable under the adapter when possible, or, at least, as close as possible to the adapter. Do not add insulating sleeves to ferrules that are clamped to fuel tank probes. No shield termination shall be placed under the harness clamp.

Floating shield termination shall be made by folding the shield back over the cable as shown in figure 24-25. Shrinkable sleeving shall extend a minimum of 1/4-inch over each end of the folded portion of the shield.

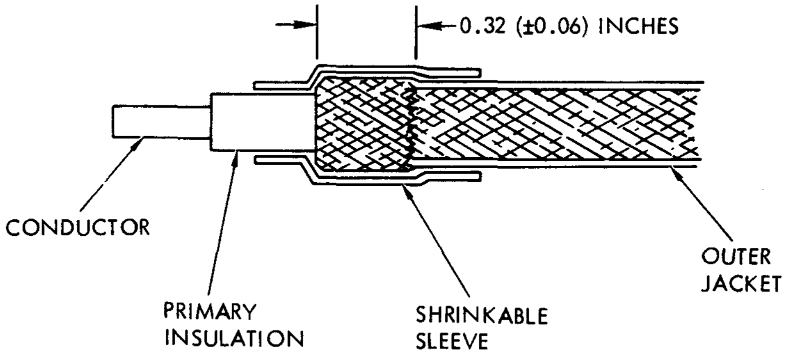


Figure 24-25. Floating Shield Termination

24.3.7 BUNDLE AND ROUTING REQUIREMENTS. Wire and cable bundles shall be tied using flat braid or plastic strips. All wire and cables in pylon and nacelle shall be tied with teflon pre-coated glass braid. Braid shall not be used for attachment to structure. Keep spot ties to a minimum but use as required to produce a neat and orderly installation.

Routing shall provide (1) accessibility for maintenance, (2) minimum possibility of damage, (3) minimum radio noise and (4) minimum interference with compass and other magnetic equipment.

Wires and cables imprinted in red may be bundled but shall be routed separately from other wiring. AC signal wires may be routed with DC wires. Wires and cable imprinted in green may be bundled but shall be routed separately from other wiring.

Wiring shall be routed away from all heated equipment ducts and lines that exceed 200°F to prevent insulation deterioration. A drip loop should be provided for all exposed wiring to prevent liquids from running down the harness into connectors, terminal blocks, connector panels, or other devices.

When identical connectors are used in adjacent locations, bundles shall be routed and supported so that improper connections cannot be made. Except for drip loop requirements, only enough slack shall be left in bundles to (1) permit ease of maintenance, (2) permit replacement of terminals, (3) prevent mechanical strain, and (4) permit free movement of shock and vibration mounts. Harnesses shall be routed and supported to prevent chafing, that is contact with other harnesses and sharp or abrasive surfaces. Harnesses that can be pushed into a sharp or abrasive surface without slipping in adjacent clamps, and spring back a minimum 1/4-inch, are acceptable when the sharp or abrasive surface is wrapped or covered with a smooth, non-abrasive material. Rerouting is required without the 1/4-inch spring back. Harnesses in conduits, troughs, and ducts shall be supported within 10 inches of entry and exit points to preclude chafing. When relative movement occurs (such as across hinge lines) the harness shall be supported to prevent strain at termination points and to provide a harness twisting action without damaging wire bending. Wires and cable shall not be fastened inside conduit, insulated tubing, or permanently closed troughs or ducts. Rubber chafing strips will be cemented inside wire feed through holes above 1.75 inches in diameter. MS35489 rubber grommets will be used in smaller holes.

Minimum bend radii should be held to:

Wire	3 times wire diameter
Bundle	5 times bundle outside diameter
Flex-type, Coaxial Cable	6 times outside diameter
Semi-rigid, Coaxial Cable	10 times outside diameter

24.3.8 PROXIMITY TO FLAMMABLE FLUID LINES. As an objective, wiring should be held at least 6 inches from flammable fluid lines and oxygen lines. Never should this separation be allowed to fall below 1/2-inch. To prevent the possibility of wiring coming into contact with lines containing oxygen or flammable fluids, even when severed, the distance between wire clamps and ties shall be no greater than the distances between the wire and the noted lines.

24.3.9 CLAMPING. The following clamps are generally used:

MS21919WDG	200°F Maximum - No fuel or oil exposure.
LS7357WSST	In fuel tanks and fuel valve compartments.
MS21919WH	Above 200°F. No fuel or synthetic oil exposure.
LS7357SST or WSST	In fuel and synthetic oil areas.

Non-wedge clamps may be used in size 1/4-inch and below.

For adding wires to existing bundles: Open all clamps, remove spot ties, route new wire without interweaving, carefully close all clamps without damaging wire or pinching the insulation, replace spot ties, and do not thread new wire under existing clamps or ties.

24.3.10 WIRE CONNECTIONS. Lockwashers shall be used on terminations size No. 4 and smaller and under screw heads for sizes 6, 8, and 10. Flat washers are required between nuts and terminals and for 0.25 inch and larger screws or studs. Flat washers will not be used where they cause interference with insulation, fail to lie flat against terminal lug, reduce creepage path or air gap to less than 0.12 inch, or with extremely short studs. Anodized or dyed washers should not be used. Brass washers should be used with copper terminals. Good contact is required on all grounds with no more than four terminal lugs placed on any ground screw. When stacked on one stud/screw, the largest terminal lug will be at the bottom and all others stacked in order of decreasing size. Torque values for terminal installation will be as follows:

24.3.10 (Cont).

COPPER TERMINAL INSTALLATION TORQUE VALUES

STUD SIZE	TORQUE (IN.-LB.)	
	PLAIN NUTS WITH LOCK WASHERS	SELF-LOCKING NUTS
6 - 32	12 - 17	16 - 20
10 - 32	15 - 20	25 - 30
1/4" - 28	55 - 65	80 - 90
5/16" - 24	70 - 80	120 - 130
3/8" - 24	110 - 120	180 - 190
1/2" - 20	135 - 150	245 - 260
NOTE: These values do not apply when equipment nameplate indicates a torque value.		

Safety wiring in accordance with MS33540 is required for all connectors outside the cockpit area. Connectors common to the same unit may be safetied together, provided that rotation of one will not result in loosening another.

CHAPTER 25
EQUIPMENT AND FURNISHINGS

TABLE OF CONTENTS

Section		Page
25.0	PILOT'S AND COPILOT'S SEATS	25-3
25.1	SAFETY BELTS AND INERTIA REEL	25-5
25.2	HAND FIRE EXTINGUISHERS	25-7

SECTION 25.0
PILOT'S AND COPILOT'S SEATS

Contents Page

DESCRIPTION AND OPERATION 25-3

REMOVAL AND INSTALLATION 25-3

25.0.1 DESCRIPTION AND OPERATION. The pilot's and copilot's seats are adjustable for reclining, fore-aft and vertical movement. Both armrests of the seats are also adjustable. The fore-aft adjustment lever is centered under the lower forward edge, below the seat cushions. The reclining and vertical adjustment levers are located on the inboard side of the seats. The vertical adjustment mechanism is spring loaded and should not be operated unless the seat is occupied. Adjustment limits of the seats are 10 inches fore-aft and 5 inches vertical.

25.0.2 REMOVAL AND INSTALLATION.

a. Remove the stops from the forward and aft ends of the seat tracks.

b. Roll the seat forward, and lift the forward wheels of the seat out through the openings previously occupied by the stops.

c. Roll the seat aft, and lift the rear wheels of the seat out through the openings previously occupied by the stops.

d. Replace the stops in the seat tracks.

e. To install pilot's seat and copilot's seat, perform steps (a) thru (d) in reverse order.

SECTION 25.1
SAFETY BELTS AND INERTIA REEL

Contents

Contents	Page
PILOT'S AND COPILOT'S SAFETY BELTS	25-5
INERTIA REELS	25-5

25.1.1 PILOT'S AND COPILOT'S SAFETY BELTS. The safety belts consist of a four-strap harness, having shoulder and lap straps leading to a rotary-type buckle. The shoulder-harness straps pass through a guide at the top aft edge of the seat back and attach to the inertia reel located on the lower back side of the seat. The lap straps are attached to lugs on each side of the seats by means of quick disconnect fittings. The rotary buckle is semi-permanently attached to the outboard lap strap. To fasten the harness, snap the inboard-lap strap and the shoulder harness strap into the rotary buckle, then pull up the shoulder and lap straps until snug. To release the straps rotate the front plate on the buckle. To release the shoulder strap only, press the tab at the top of the buckle.

25.1.2 INERTIA REELS. The inertia reel, to which the shoulder straps are attached, is controlled by a locking lever on the outboard side of the seats. When the control lever is in the unlocked position, the shoulder straps will retract freely and can be extended unless locked by a "G" force against the straps. When the inertia reel control is in the locked position, the straps cannot be extended and all of the slack will be taken up.

SECTION 25.2
HAND FIRE EXTINGUISHERS

Contents Page

DESCRIPTION AND OPERATION 25-7

25.2.1 DESCRIPTION AND OPERATION. Two hand fire extinguishers are provided for emergency use. A CO_2 type extinguisher is mounted in the vestibule adjacent to the cockpit. The charged weight of the extinguisher is 8.2 pounds. In the passenger area there is a liquid type fire extinguisher with a charged weight of 6.88 ± 0.25 pounds. The mixture for refilling this extinguisher should be in accordance with the manufacturer's recommendations.

CHAPTER 26
FIRE PROTECTION

TABLE OF CONTENTS

Contents — Page

DESCRIPTION AND OPERATION 26-1
General 26-1
Overheat Detection System 26-2
Fire Detection System 26-2
Overheat and Fire Detection Systems Test Switch 26-2
Fire Isolation System 26-2
Fire Extinguishing System 26-2
Directional Control Valves 26-3

OPERATIONAL CHECKOUT 26-3
Fire Extinguishing System Checkout Procedure 26-3
Overheat and Fire Detection Systems Checkout Procedure 26-4

TROUBLESHOOTING 26-4
Overheat and Fire Detection Systems 26-4
Fire Extinguishing System 26-5

PRESERVATION 26-6
Preservation of the Fire Extinguishing System 26-6
Depreservation of the Fire Extinguishing System 26-6

GROUND ENGINE FIRE EXTINGUISHING PROCEDURE 26-6

TORQUE VALUES FOR FIRE EXTINGUISHING LINE FITTINGS 26-6

FIGURES

26-1. Fire Protection System Components 26-7
26-2. Fire and Overheat Detector System Schematic Diagram 26-8
26.3. Fire Extinguishing System Schematic Diagram 26-9

26.1 DESCRIPTION AND OPERATION.

26.1.1 GENERAL. (See figure 26-1.) Each engine compartment is divided into two zones by the transverse vertical firewall between the compressor section (zone II) and the turbine and exhaust section (zone I). The overheat detection system indicates an overheat condition in zone I of an engine by a flashing light in the fire emergency handle for that engine. The fire detection system indicates a fire in zone II of an engine by a steady light of the fire emergency handle. Only zone II is protected by the fire extinguishing system, since combustible fluids are isolated from zone I by the vertical firewall.

26.1.2 OVERHEAT DETECTION SYSTEM. (See figure 26-2.) The overheat detection system for each engine consists of a sensing element, flasher unit, control unit for each zone, two lights in the fire emergency handle, and one part of a four-pole test switch. The sensing element is a wire embedded in a special ceramic core and enclosed by an inconel tube. The electrical resistance of the ceramic core decreases as the temperature increases, permitting a small current flow between the wire and the inconel tube. The tube forms an electrical ground for the circuit. If the temperature, and hence the current, increases sufficiently to trip the control unit, the lights in the fire emergency handle flash.

26.1.3 FIRE DETECTION SYSTEM. Operation of the fire detection system is similar to that of the overheat detection system, except the lights in the fire emergency handle remain steady for a fire signal.

26.1.4 OVERHEAT AND FIRE DETECTION SYSTEMS TEST SWITCH. A FIRE DET WARN LT TEST switch on the main instrument panel tests the detection systems in zone I and zone II of all four engines. When the switch is placed in the "FWD ZONE" position, it tests the fire detection systems of all four engines. Placing the switch in the "AFT ZONE" position tests the overheat detection systems of all four engines. Satisfactory test results are indicated by steady illumination of the lights in the fire emergency handle for the fire detection system check, and flashing of the lights for the overheat warning system check.

26.1.5 FIRE ISOLATION SYSTEM. The fire isolation system confines a fire to the area in which it starts until it can be extinguished. If a fire starts in any engine, operation of the corresponding fire emergency handle brings about all the switching necessary to completely isolate the fire.

CAUTION

Do not use the fire emergency handles to shut down an engine except in an emergency. Using the emergency handles allows the engine fuel pump to operate in a dry state and can result in pump damage. If the engine is shut down in this manner for any reason, inspect all engine fuel pump and engine fuel control filters for evidence of fuel pump damage.

26.1.5.1 ENGINE SHUTDOWN-FIRE EMERGENCY. The switches actuated by the fire emergency handles complete the circuits required to shut down the engines. Pulling the fire emergency handle for any engine closes the fuel system shutoff valve, hydraulic pump outlet and inlet valve (No. 2 and No. 3 engines only), and the bleed air shutoff valve. The fire emergency handle also breaks the automatic ignition path through the stall warning system (No. 1 and No. 4 engines only), breaks all ignition circuits to the igniters (No. 2 and No. 3 engines only), opens the generator field circuit, positions fire extinguishing agent directional control valves to direct the fire extinguishing fluid to the correct engine, and arms the extinguishing agent discharge circuit.

CAUTION

After two fire emergency control handles have been pulled, the second handle cannot be pushed in to reestablish an agent flow path to the first nacelle. Flow to the desired nacelle can be established only by pushing-in, if pulled, and pulling again. Agent will always be directed to the nacelle for which the last handle was pulled.

CAUTION

If two or more fire emergency control handles are pulled and electrical power to the essential bus is interrupted for any reason, all handles should be pushed in and then repositioned as desired. Failure to do this may cause a stepping relay in the fire extinguisher valve control assembly to "hunt" continuously and eventually burn out.

26.1.5.2 ENGINE OPERATION AFTER FIRE EMERGENCY SHUTDOWN. The following procedure is recommended to restore the engine to operating status if the engine was shut down with the fire emergency handle:

a. Return the fire emergency handle to its normal position.

b. Install serviceable fire extinguisher bottles and discharge squibs.

c. Visually inspect the position of the firewall shutoff valves to make sure they reopened when the fire emergency handle was returned to its normal position.

d. Perform a normal engine inspection and an operational checkout of the fire extinguishing, overheat, and fire detection systems before engine operation.

26.1.6 FIRE EXTINGUISHING SYSTEM.

a. The fire extinguishing system provides fire protection for each of the four engines. (See figure 26-3.) Bromo-trifluoromethane (BT) is the recommended fire extinguishing agent; however, other

26.1.6 (Cont).

agents may be used when BT is not available provided the system is drained and purged before servicing with the different agent. The agent is contained in two bottles. The bottles are discharged one at a time, and the agent from either bottle can be directed to any engine. Each bottle is supercharged with nitrogen and is discharged by an electrically exploded charge. The bottle outlet lines are connected by a check valve so the agent released from any bottle enters a common outlet line leading to a directional control valve. Three solenoid-operated directional control valves in the system are positioned to direct agent flow. The outlets for the agent are perforated tubes. The controls for the system are on the main instrument panel.

26.1.6 (Cont).

b. To discharge fire extinguishing agent into any engine bay, pull the corresponding fire emergency handle, and move the FIRE EXT AGENT switch to "NORMAL". To discharge the second container, move the switch to "RESERVE".

26.1.7 DIRECTIONAL CONTROL VALVES. The directional control valves are solenoid-operated flapper valves. Each valve has one inlet and two outlets. The outlet ports are marked "NC" and "NO" to indicate which port is normally closed and which is normally open when the valve is de-energized. When the valve is energized, the flapper blocks the normally open port. A spring returns the flapper to the normal position.

26.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Air Compressor	75 PSI (Min.) 500 PSI (Max.)	To supply air into the fire extinguishing agent line to determine correct positioning of directional valves.
Hose, length of		To connect air compressor to fire extinguishing lines.
Generator, external power	28 volts, DC	To supply power to the airplane during the checkout.
Voltmeter	To measure 28 volts, DC, with an accuracy of ±1 volt.	To measure voltage at test points during checkout.

26.2.1 FIRE EXTINGUISHING SYSTEM CHECKOUT PROCEDURE.

a. Ground the airplane structure and connect the external power source to the airplane external power receptacle.

b. Disconnect the electrical connector from each fire extinguisher agent discharge squib.

WARNING

Tape all terminals and secure all wires clear of squib terminals before applying current to the circuit to avoid any possibility of firing the bottle.

26.2.1 (Cont).

c. Energize the DC buses and close the following circuit breakers on the pilot's circuit breaker panel.

(1) Hydraulic Shutoff, Eng. No. 3
(2) Hydraulic Shutoff, Eng. No. 2
(3) Fire Extinguisher Control Power
(4) Fuel Shutoff Valve, Eng. No. 1
(5) Fuel Shutoff Valve, Eng. No. 2
(6) Fuel Shutoff Valve, Eng. No. 3
(7) Fuel Shutoff Valve, Eng. No. 4
(8) Bleed Air Shutoff Valve, Eng. No. 1
(9) Bleed Air Shutoff Valve, Eng. No. 2
(10) Bleed Air Shutoff Valve, Eng. No. 3
(11) Bleed Air Shutoff Valve, Eng. No. 4

26.2.1 (Cont).

d. Pull the No. 3 engine fire emergency handle.
RESULT: The Engine No. 3 hydraulic pump inlet valve, hydraulic shutoff valve, bleed air shutoff valve, and fuel shutoff valve should close.

e. Position the FIRE EXT AGT switch in the "NORMAL" position and check the voltage at the normal and reserve fire extinguisher bottle electrical connections.
RESULT: There should be 28-volt DC at the normal bottle connections and zero volts at the reserve bottle connections.

f. Position the FIRE EXT AGT switch in the "RESERVE" position and check the voltage at both fire extinguisher agent bottle electrical connections.
RESULT: There should be 28-volt DC at the electrical connector of the reserve bottle and zero volts at the electrical connector of the normal bottle.

g. Return the FIRE EXT AGT switch to "OFF," and check the voltage at both fire extinguisher agent bottle electrical connections.
RESULT: There should be zero volts at both connectors.

h. Return the No. 3 engine fire emergency handle to normal.
RESULT: The No. 3 engine hydraulic pump inlet valve, hydraulic shutoff valve, bleed air shutoff valve, and fuel shutoff valve should open.

i. Check the voltage at the electrical connections of the normal and reserve fire extinguisher agent bottles while the FIRE EXT AGT switch is in the "NORMAL" and "RESERVE" position.
RESULT: The voltage at the connectors of both bottles should be zero volts while the switch is in either position.

j. Pull the No. 3 engine fire emergency handle, and apply air at 100 PSI to the fire extinguisher agent line at the fire extinguisher agent bottle connection.

Note

The air pressure should be applied suddenly and air allowed to flow for 2 minutes to seat the directional control valves.

RESULT: The air should be flowing from the agent discharge tube in the No. 3 engine bay. A small flow may be detected at the other three engines.

k. Repeat steps (d) through (j) for the other three engines. Engines No. 1 and 4 do not have a hydraulic pump or hydraulic shutoff valves.

l. Turn off the air supply, disconnect all test equipment, and reconnect airplane wiring to return all systems to normal.

26.2.2 OVERHEAT AND FIRE DETECTION SYSTEMS CHECKOUT PROCEDURE.

a. Operation of the FIRE DET WARN LT TEST switch makes a complete check of the overheat and fire detection system. Placing the switch in the "AFT ZONE" position checks the continuity of the overheat sensing elements, the amplifiers, and the lights in the fire emergency handle. The lights in the handle should flash.

b. Placing the switch in the "FWD ZONE" position checks the continuity of the fire detection sensing elements, the amplifiers, and the lights in the fire emergency handle. The lights in the handle should remain on and steady.

26.3 TROUBLESHOOTING.

26.3.1 OVERHEAT AND FIRE DETECTION SYSTEMS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
OVERHEAT OR FIRE WARNING LIGHTS IN A FIRE EMERGENCY HANDLE DO NOT ILLUMINATE WHEN TEST SWITCH IS OPERATED.		
Essential DC bus not energized.	Check for 28-volt DC at the bus.	If correct voltage is not avaliable, troubleshoot the DC power supply system.
The detector sensing element is broken.	Check continuity of the sensing element.	Replace the sensing element.

26.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
OVERHEAT OR FIRE WARNING LIGHTS IN A FIRE EMERGENCY HANDLE DO NOT ILLUMINATE WHEN TEST SWITCH IS OPERATED (Cont).		
FIRE DETECTOR TEST circuit breaker is open.	Check the circuit breaker.	Close the circuit breaker.
Control unit is defective.	Check for 28-volt DC between pin F and ground and between pins H and B of the control unit.	Replace the unit if voltage is present at F, but not between pins H and B.
Bulbs in fire emergency handle burned out.	Substitute new bulbs.	Replace bulbs.
Test switch is defective.	Check switch for continuity.	Replace defective switch.
OVERHEAT LIGHT BURNS CONTINUOUSLY WHEN TEST SWITCH IS OPERATED.		
Flasher unit is defective.	Substitute a new flasher.	Replace defective flasher.

26.3.2 FIRE EXTINGUISHING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FIRE EXTINGUISHING AGENT DOES NOT DISCHARGE.		
Essential DC bus not energized.	Check bus for 28-volt DC.	If voltage is not present, troubleshoot DC power supply system.
FIRE EXTINGUISHER circuit breaker open.	Inspect circuit breaker.	Close circuit breaker.
Agent discharge squib is defective.	Disconnect electrical wires at squib. Use a voltmeter to check for voltage on aircraft wires when switch is closed. CAUTION Do not connect meter to squib terminals.	If voltage exists, replace squib. If voltage does not exist, check circuit to determine cause of trouble, and rectify as necessary.
Fire emergency handle switch defective.	Check switch for continuity.	Replace defective switch.
AGENT DISCHARGES INTO WRONG ENGINE BAY.		
Directional control valves are binding.	Attempt to operate valves by hand.	Replace valves that do not open or close freely.
Stepping switch defective.	Check operation of stepping switch by observing operation of directional valves when fire handles are pulled.	Replace switch if it is not in correct position according to the pulled fire emergency handle.

26.3.2 FIRE EXTINGUISHING SYSTEM. (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
AGENT DISCHARGES INTO WRONG ENGINE BAY. (Cont).		
Plumbing misconnected	Trace plumbing to ensure correct connection from valves through pylon to engine bay.	Correct plumbing connections

26.4 PRESERVATION.

26.4.1 PRESERVATION OF THE FIRE EXTINGUISHER SYSTEM DURING PROLONGED INACTIVITY. If an airplane is to be inactive for a prolonged period, observe the following precautions to protect the fire extinguishing system from deterioration and damage.

a. Remove the squibs from the airplane for storage.

b. Maintain properly serviced, portable fire extinguishers in the airplane.

Note

Protect the liquid type fire extinguisher from freezing temperatures.

c. Inspect the portable fire extinguishers at 30-day intervals for proper servicing.

26.4.2 DEPRESERVATION OF THE FIRE EXTINGUISHING SYSTEM AFTER PROLONGED INACTIVITY. Before returning the airplane to service after prolonged inactivity, reinstall the squibs in the airplane, and make sure the entire system is properly serviced.

26.5 GROUND ENGINE FIRE EXTINGUISHING PROCEDURE.

Use the following procedure if there is evidence of a fire within the engine or extremely high exhaust gas temperature indication:

a. Place the throttle for the affected engine in the "OFF" position.

b. As soon as the tachometer indicates deceleration to 40 percent RPM or less, position the ENG SEL switch to the affected engine.

c. Place the POWER SEL switch in either "GPU" or "BAT," depending on the source of power available.

d. Place the AIR START switch in the "OFF" position.

e. Place the START switch in the "START" position, and maintain starter operation for 20 seconds.

f. Repeat starter operation as necessary.

CAUTION

Do not spray ground fire extinguisher agent into the engine air intake or exhaust tailpipe except in case of a fire which obviously could endanger the airplane.

26.6 TORQUE VALUES FOR FIRE EXTINGUISHING LINE FITTINGS.

Torque values for the NAS596 rigid connectors used in the fire extinguishing system are shown in figure 28-4A.

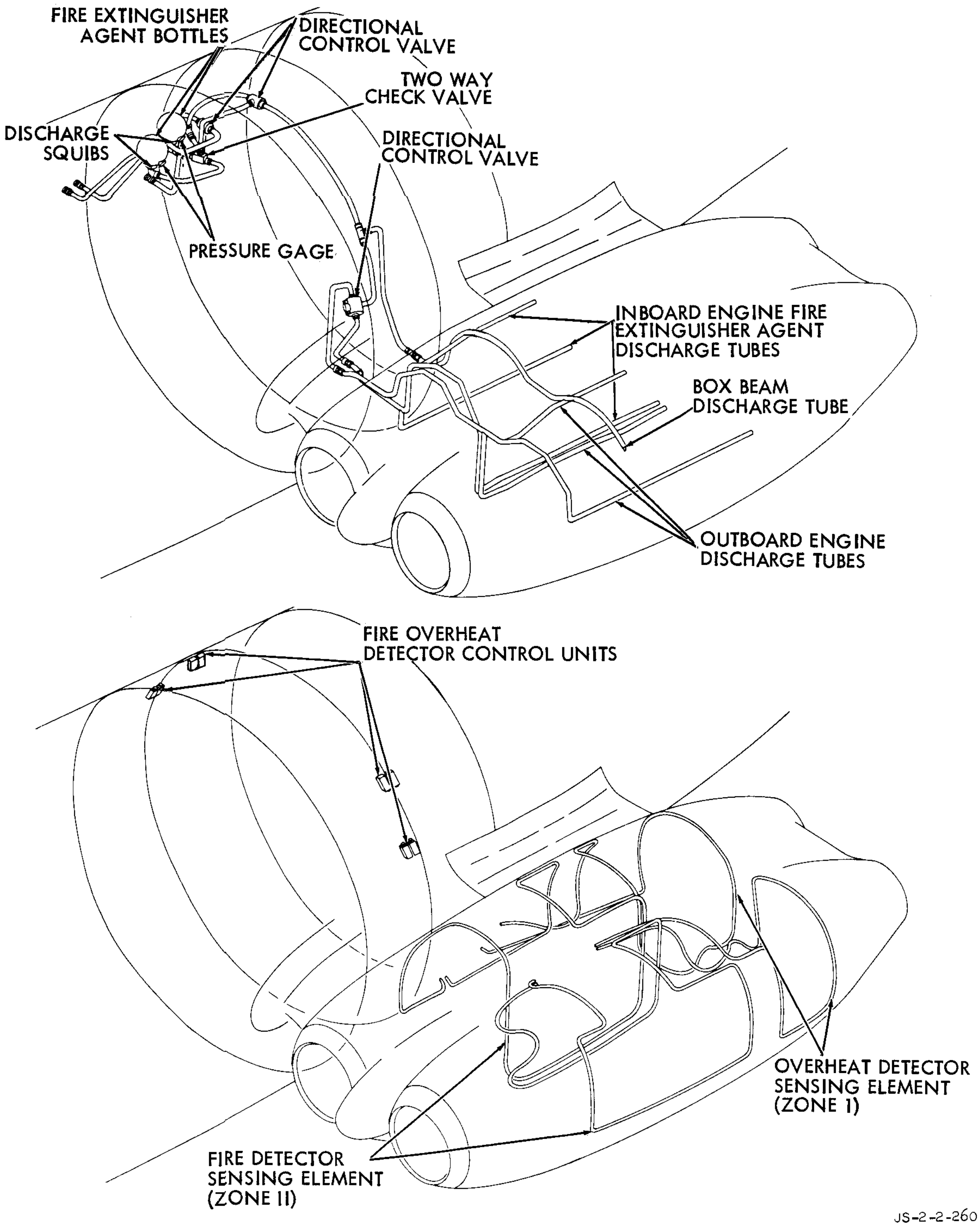
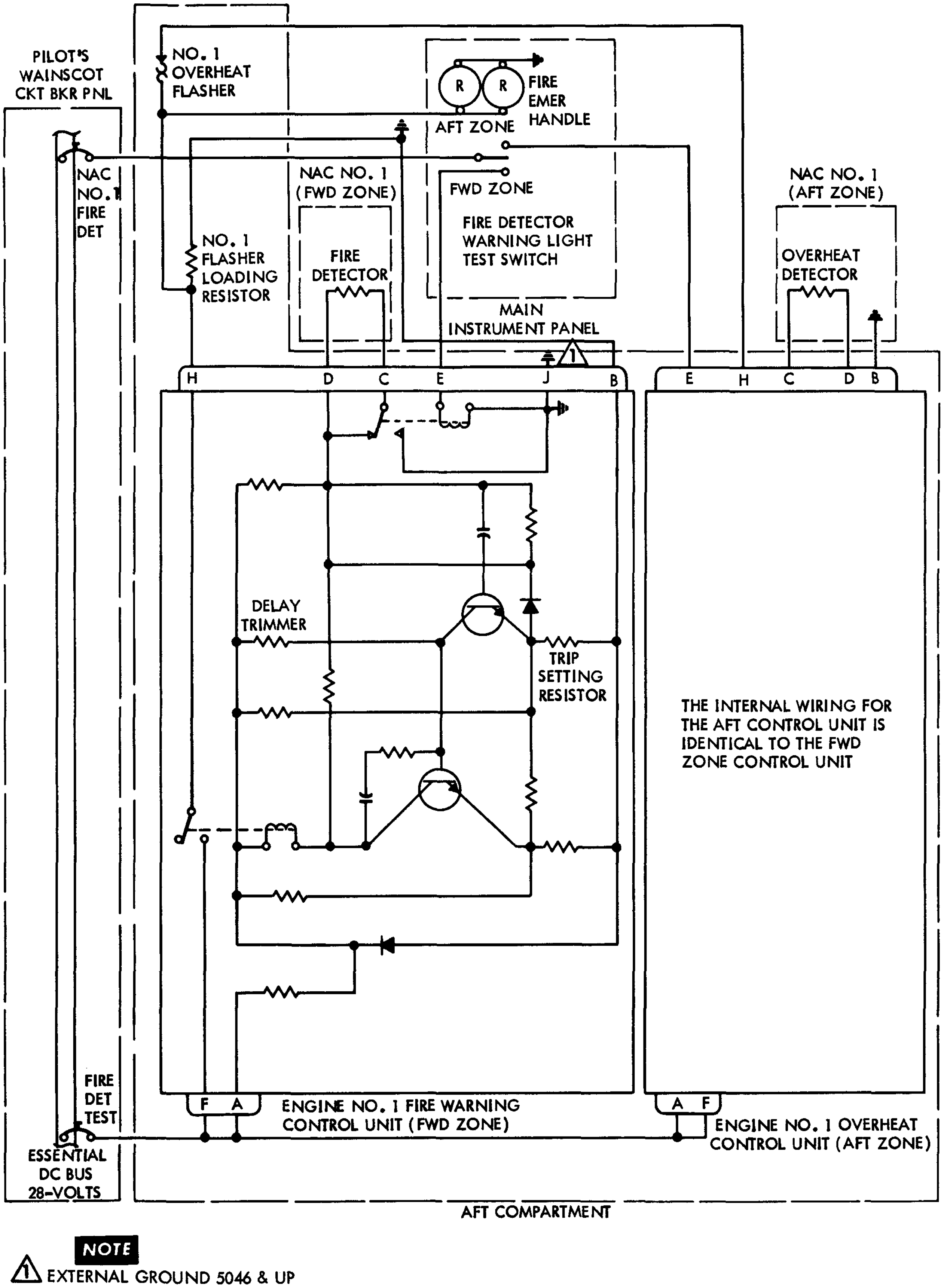

FIRE EXTINGUISHER AGENT BOTTLES
DIRECTIONAL CONTROL VALVE
TWO WAY CHECK VALVE
DISCHARGE SQUIBS
DIRECTIONAL CONTROL VALVE
PRESSURE GAGE
INBOARD ENGINE FIRE EXTINGUISHER AGENT DISCHARGE TUBES
BOX BEAM DISCHARGE TUBE
OUTBOARD ENGINE DISCHARGE TUBES
FIRE OVERHEAT DETECTOR CONTROL UNITS
OVERHEAT DETECTOR SENSING ELEMENT (ZONE 1)
FIRE DETECTOR SENSING ELEMENT (ZONE II)
JS-2-2-2601

Figure 26-1. Fire Protection System Components


PILOT'S WAINSCOT CKT BKR PNL
NAC NO. 1 FIRE DET
NO. 1 OVERHEAT FLASHER
NO. 1 FLASHER LOADING RESISTOR
R
R
FIRE EMER HANDLE
AFT ZONE
FWD ZONE
NAC NO. 1 (FWD ZONE)
FIRE DETECTOR
FIRE DETECTOR WARNING LIGHT TEST SWITCH
MAIN INSTRUMENT PANEL
NAC NO. 1 (AFT ZONE)
OVERHEAT DETECTOR
H
D
C
E
J
B
E
H
C
D
B
DELAY TRIMMER
TRIP SETTING RESISTOR
THE INTERNAL WIRING FOR THE AFT CONTROL UNIT IS IDENTICAL TO THE FWD ZONE CONTROL UNIT
F
A
ENGINE NO. 1 FIRE WARNING CONTROL UNIT (FWD ZONE)
A
F
ENGINE NO. 1 OVERHEAT CONTROL UNIT (AFT ZONE)
FIRE DET TEST
ESSENTIAL DC BUS 28-VOLTS
AFT COMPARTMENT
1

NOTE

⚠1 EXTERNAL GROUND 5046 & UP

Figure 26-2. Fire and Overheat Detector System Schematic Diagram

JS-2-2-2602

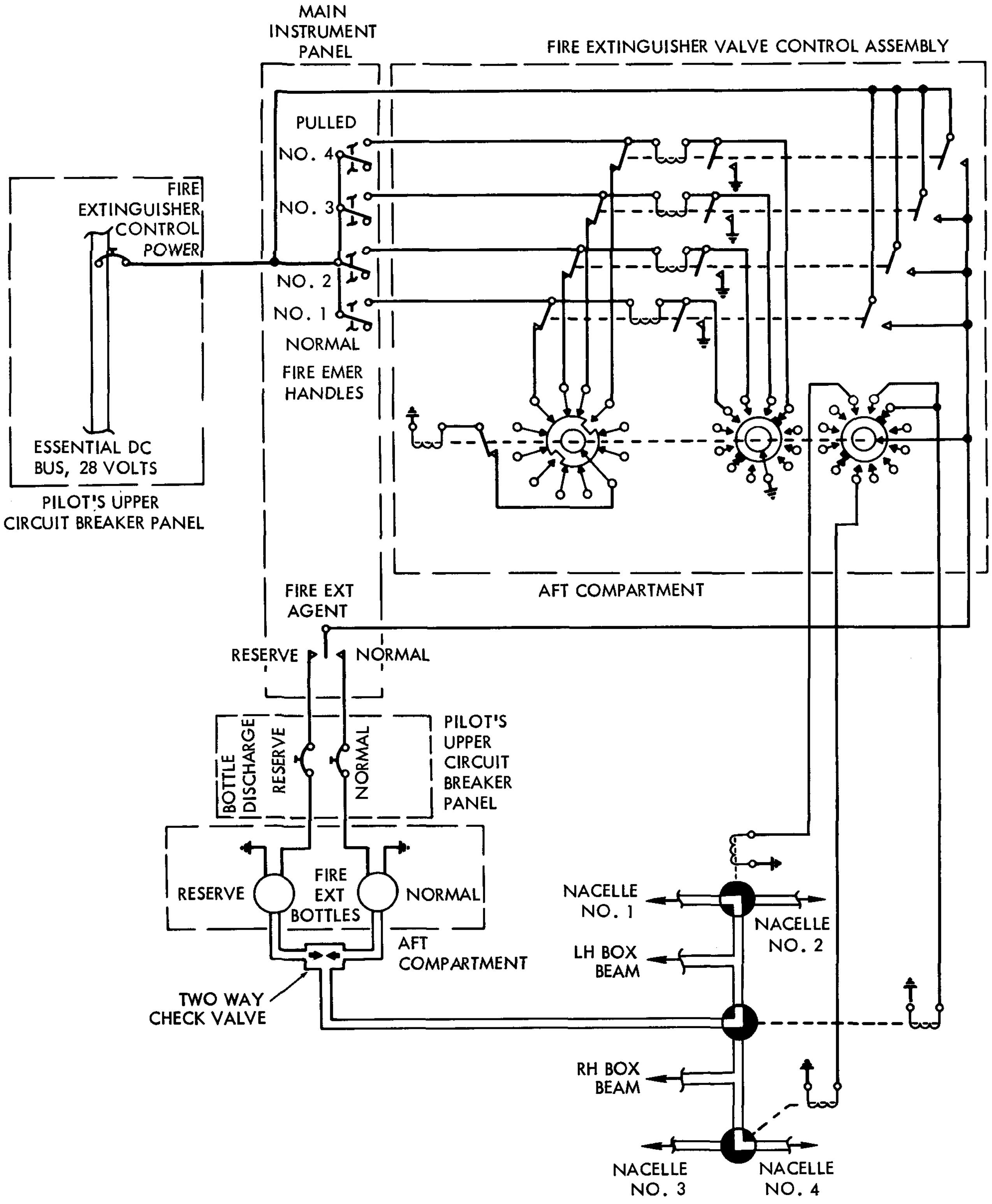


Figure 26-3. Fire Extinguishing System Schematic Diagram

CHAPTER 27

FLIGHT CONTROLS

TABLE OF CONTENTS

Section | Page
27.0 AILERON CONTROL SYSTEM 27-3
27.1 ELEVATOR CONTROL SYSTEM 27-15
27.2 RUDDER CONTROL SYSTEM 27-27
27.3 TRIM CONTROL AND MACH TRIM SYSTEM 27-35
27.4 LEADING AND TRAILING EDGE WING FLAP SYSTEMS 27-59
27.5 SPEED BRAKE SYSTEM 27-103
27.6 STALL WARNING SYSTEM-DASH 6 JETSTAR 27-111
27.7 STALL WARNING SYSTEM-DASH 8 JETSTAR 27-123
27.8 CONTROL CABLES-GENERAL REQUIREMENTS 27-133

SECTION 27.0

AILERON CONTROL SYSTEM

Contents Page

DESCRIPTION AND OPERATION 27-3
  General 27-3
  Operation 27-3
  Control Wheels 27-4
  Control Cables 27-4
  Autopilot Servo Drive 27-4
  Torque Tube 27-4
  Cable Tension Regulator 27-4
  Booster Assembly 27-4
  Control Switches 27-4
  Booster Assembly Filters and Screens 27-5
  Booster Assembly Solenoids 27-5
  Pushrod Mechanism 27-5

OPERATIONAL CHECKOUT 27-5
  Aileron Control System 27-5
  Aileron Booster Assembly Leakage Check 27-6

TROUBLESHOOTING 27-7
  Aileron Control System 27-7

REMOVAL AND REPLACEMENT 27-8
  Removal of the Aileron Booster Assembly 27-8
  Replacement of the Aileron Booster Assembly 27-8
  Removal of the Aileron Booster Tension Regulator 27-8
  Replacement of the Aileron Booster Tension Regulator 27-8
  Removal of the Aileron 27-9
  Replacement of the Aileron 27-9
  Removal and Replacement of Aileron Booster Filters 27-9
  Bolt Torque Requirements 27-9

ADJUSTMENT 27-10
  Aileron Control System Rigging Instructions 27-10

INSPECTION 27-10
  Aileron Control Cable Connector Inspection 27-10
  Inspection For Excessive Looseness In Ailerons 27-10

FIGURES

27-1. Aileron Control System Schematic Diagram 27-11
27-2. Aileron Control System Rigging Instructions 27-12
27-3. Aileron Lockclad Cable Connector Inspection 27-14

27.0.1 DESCRIPTION AND OPERATION

27.0.1.1 GENERAL. (See figures 27-1 and 27-2.) Lateral control of JetStar airplanes is accomplished through use of a hydraulically boosted aileron control system. Dual control wheels in the pilots' compartment act through a cable system to rotate a torque tube under the pilots' floor. This torque tube interconnects the control wheels for dual control reliability, and drives another cable system to position the aileron booster assembly control valve. During normal operation, this valve routes both normal and standby hydraulic system power to a tandem actuating cylinder. The actuating cylinder operates a system of pushrods and bellcranks to position the ailerons. See Section 59.8 for typical repairs to the ailerons and aileron control items.

27.0.1.2 OPERATION. (See figures 27-1 and 27-2.) Rotating either control wheel drives a cable system to rotate the torque tube under the pilots' floor. As the torque tube rotates, it moves dual cables, which turn a tension regulator on top of the aileron booster assembly. Turning this tension regulator positions

27.0.1.2 (Cont).

the tandem servo (control) valve to admit or block hydraulic power to the proper sides of the tandem actuating cylinder. As the actuating piston moves, it positions a pushrod and bellcrank mechanism to actuate the ailerons according to control wheel rotation. Normal and standby hydraulic system fluids are kept separate in the booster assembly. Either system can be used independently of the other to power the aileron booster assembly. A mechanical followup linkage normally follows the hydraulic boosting action of the booster; in the absence of hydraulic power, this linkage takes over to position the ailerons manually.

27.0.1.3 CONTROL WHEELS. There are two U-shaped control wheels: the pilot's and the copilot's. Each wheel is mounted on a shaft which rotates in bearings at the top of the control column. Each shaft has a pulley attached to it with a single cable looped over the pulley and locked in place. The ends of the cable attach to a sector on the aileron torque tube. The grips of the control wheels have controls for normal pitch trim, an autopilot disconnect button, a trim disconnect button, and a transmit/intercom switch.

27.0.1.4 CONTROL CABLES. There are 12 control cables in the aileron control system. One cable for each control wheel is routed over the control wheel pulley with its ends connected to a sector on the torque tube. A dual set of cables connects to a sector on the torque tube and runs aft to join another set between fuselage stations 243 and 256. The latter cables run aft and connect to the cable tension regulator on top of the aileron booster assembly. Two autopilot servomotor cables wrap around a servomotor drum and drive a sector on the aileron torque tube. Fairleads, pulleys, and other anti-fouling devices maintain proper cable routing. Both Lockclad and flexible, prestretched, steel-wire cables are used in the aileron control system. (Refer to Section 27.8.)

27.0.1.5 AUTOPILOT SERVO DRIVE. During autopilot operation, the servomotor, in response to signals from the autopilot system, positions the aileron torque tube by two cables connecting a servomotor drum to a sector on the aileron torque tube. The servomotor drum is driven through a clutch assembly which automatically disengages when the servomotor is not operating. The aileron servomotor is attached to the airplane structure aft of the aileron torque tube and under the pilot's floor.

27.0.1.6 TORQUE TUBE. The aileron torque tube is under the cockpit floor. It is made up of three sections: a left section, a center section, and a right section. These sections are connected by sprocket and chain assemblies. Where the sections meet, sprockets butt together and chain assemblies cover them to form unions. There are three sectors on the left section of the torque tube, one sector on the right section, and no sectors on the center section. The outboard sector of the left section connects to cables from the autopilot servomotor. The center sector of the left section connects to cables from the aileron booster assembly. The inboard sector of the left section connects to a cable from the pilot's control wheel pulley. The sector on the right section connects to a cable from the copilot's control wheel pulley. The center section of the torque tube connects the left and right sections for dual control reliability. The ends of the left and right aileron torque tube sections ride in bearings and attachment fittings attached to the airplane structure.

27.0.1.7 CABLE TENSION REGULATOR. A spring-loaded cable tension regulator on top of the aileron booster assembly maintains proper cable tension through a temperature range of -60 to +165°F. The degree of cable tension is shown on a tension indicator on top of the tension regulator. This indicator scale is from "C" (cold) to "H" (hot) with 10 divisions in between. A chart in the aileron rigging instructions shows the proper indicator readings for different temperatures.

27.0.1.8 BOOSTER ASSEMBLY. (See figure 27-1.) The aileron booster assembly contains a manifold and servo (control) valve sub-assembly, an actuator sub-assembly, and a mechanical followup linkage. The manifold and servo valve sub-assembly contains a servo valve, two filters, six split screens, four check or poppet valves, two shutoff and bypass valves, and fluid passages. The actuator sub-assembly contains a tandem actuating cylinder and fluid passages. A mechanical followup linkage connects the control valve input linkage to the actuating cylinder output arm. Normally, both normal and standby hydraulic system power is used for aileron booster operation, but either system can be used independently of the other with no increase in the amount of effort required to control the ailerons. In the absence of hydraulic power, the mechanical followup linkage positions the ailerons manually.

27.0.1.9 CONTROL SWITCHES. Hydraulic power to the booster assembly is controlled by two BOOST HYD PRESS switches an AILERON SYS 1 switch and an AILERON SYS 2 switch. These switches control the shutoff and bypass valves in the manifold and servo valve assembly of the booster. The SYS 1 switch controls the normal hydraulic system shutoff and bypass valve, and the SYS 2 switch controls the standby hydraulic system shutoff and bypass valve. Placing either switch in the "OFF" position energizes the corresponding valve to block hydraulic power from the booster control valve and at the same time open a bypass for the fluid within that portion of the assembly to prevent hydraulic locks within the booster. Placing either switch in the "ON" position de-energizes the corresponding valve to admit hydraulic power to that portion of the booster. These switches are guarded in the "ON" position, and are on the pilot's control column cover with the elevator booster switches.

27.0.1.10 BOOSTER ASSEMBLY FILTERS AND SCREENS. (See figure 27-1.) Normal hydraulic system fluid enters the booster assembly through a micronic filter assembly which contains a bypass valve. This bypass valve relieves fluid around clogged filter elements. Fluid from this filter then passes through a screen. After the fluid passes through the screen, it goes through the control valve and actuating cylinder. Filters can be removed for cleaning, but screens can be removed only by disassembling the booster assembly. The screens and filters in the standby hydraulic system portion of the booster assembly are identical to those of the normal system portion. Filters should be removed and cleaned every 200 flight hours.

Note

The late model booster assembly is provided with filter units that do not contain a bypass valve as used in some earlier units.

CAUTION

When reinstalling filters, use MIL-S-8660 compound or equivalent to position the O-ring on filter shoulder. This will prevent the O-ring from dropping down around the threads and getting damaged when filter is installed.

27.0.1.11 BOOSTER ASSEMBLY SOLENOIDS. The booster assembly contains two solenoids, one in system No. 1 and one in system No. 2. When installing a solenoid, use MIL-S-8660 compound or equivalent to position the O-ring on the solenoid shoulder. This will prevent the O-ring from dropping down around the threads and getting damaged when solenoid is installed.

27.0.1.12 PUSHROD MECHANISM. A pushrod mechanism positions the ailerons according to movement of the booster output arm. This mechanism is made up of pushrods, bellcranks, and four sets of roller assemblies mounted along each wing rear beam. Each set of rollers is provided with one roller mounted on an eccentric bolt which can be used to adjust rod clearance.

27.0.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use And Application
Portable Aircraft Hydraulic System Test Stand	15 GPM at 2700 PSI Capacity (MIL-H-5606 hydraulic fluid)	To pressurize hydraulic systems.

Tools Required

Nomenclature	Description	Use And Application
Controls Surfaces Rigging Kit	Lockheed P/N 11019-MEU	To determine the faired position of the ailerons.
Aileron and Elevator Protractor	Lockheed P/N 11069-MEU	To measure aileron travel.

27.0.2.1 AILERON CONTROL SYSTEM

a. Connect 28-volt, DC power to the airplane service connections.

b. Make sure that the following circuit breakers are engaged:

Boost Control - No. 1 and No. 2
Inverters - No. 1 Main Cont
Hyd Press - Sys 1 and Sys 2

c. Connect external hydraulic power to the normal hydraulic system ground test connections. Adjust the power source to deliver 3,000 PSI at no flow, and 2,700 PSI at 10 gallons per minute.

d. Make sure the AILERON SYS 1 BOOST HYD PRESS switch on the pilot's control column cover is in the "ON" position.

e. Center the control wheels according to the instructions in the aileron rigging procedures, and check the positions of the ailerons.

Note

Use an aileron contour template from the control surfaces rigging kit to determine the aileron faired position. Free play in aileron should not exceed 3/4°.

RESULT: The ailerons should come within 0.10 inch of the aileron faired position. For example, if one aileron is faired to the wing, the other should come within 0.10 inch of the faired position.

f. Place the MAIN INV switch on the AC CONT panel in the "ON" position. Make sure the AC SYS switch is in the "NORMAL" position.

g. Apply 3,000 PSI at no flow to the normal hydraulic system, and observe the dual-pressure indicator on the main instrument panel.
RESULT: The No. 1 pointer should indicate 3,000 (±150) PSI, and No. 2 pointer should indicate 0 (±50) PSI.

h. Attach an aileron and elevator protractor to each aileron. Attach the protractors at the inboard edges of the ailerons, and zero them at the aileron faired position.

i. Place a modified deep well, 1-1/16 inch socket on the retaining nut of the pilot's control wheel, being careful not to damage the wire bundle passing through the nut. Connect a 150 inch-pound torque wrench to the socket.

Note

Modify the socket by cutting or grinding a notch in one side sufficient to fit over the wires running through the control wheel retaining nut without damaging them.

j. Rotate the pilot's wheel clockwise with the torque wrench, and observe the amount of torque required to start wheel rotation, and the amount required for full aileron deflection. Measure the aileron deflection at full travel.
RESULT: The amount of torque required to start control wheel rotation should not exceed 35 inch-pounds. The amount of torque required to obtain full aileron travel should not exceed 70 inch-pounds. At full aileron travel, the left aileron should deflect 18.1 (±1/2) degrees down, and the right aileron should deflect 22-1/2 (±1/2) degrees up from the faired position.

k. Return the control wheel to the centered position, and repeat step (j), rotating the control wheel counter-clockwise.
RESULT: The amount of torque required to start rotation of the control wheel should not exceed 35 inch-pounds. The amount of torque required to obtain full aileron travel should not exceed 70 inch-pounds. At full aileron travel, the left aileron should deflect 22.3 (±1/2) degrees up, and the right aileron should deflect 18 (±1/2) degrees down from the faired position.

l. Return the control wheel to the centered position, and repeat steps (i) through (k), using the copilot's control wheel. Results should be the same.

m. Turn one of the control wheels by hand and note the amount of force required to turn it; place the AILERON SYS 1 switch in the "OFF" position and again note the amount of force required to turn it.
RESULT: The amount of force required to turn the control wheel when the switch is in the "ON" position should be much less than that required with the switch "OFF."

n. Reduce the hydraulic pressure to zero, disconnect the hydraulic power source from the normal hydraulic system ground test connections, and connect it to the standby hydraulic system ground test connections.

o. Apply 3,000 PSI to the standby hydraulic system, and observe the dual-pressure indicator on the main instrument panel.
RESULT: The No. 1 pointer should indicate 0 (±50) PSI, and the No. 2 pointer should indicate 3,000 (±150) PSI.

p. With hydraulic pressure applied to the standby hydraulic system, place the AILERON SYS 2 switch on the pilot's control column cover in the "ON" position.

q. Repeat step (l). Results should be the same.

r. Rotate one of the control wheels by hand, noting the amount of force required to turn it, and then place the AILERON SYS 2 switch in the "OFF" position and note the amount of force required to rotate it with boost pressure off.
RESULT: The amount of force required to rotate the control wheel with boost pressure on should be much less than that required with boost pressure off.

s. Perform a booster leakage check according to the instructions in this section.

t. Reduce hydraulic pressure to zero, and disconnect the power sources from the airplane.

u. Cycle the controls until all hydraulic pressure is relieved, and then return the control wheels to the centered position.

v. Measure the amount of force required to rotate the control wheels with no boost pressure.
RESULT: The amount of force required to start wheel rotation should not exceed 90 inch-pounds. The amount of force required to obtain full aileron travel should not exceed 140 inch-pounds.

w. Remove the aileron protractors, and the torque wrench and socket. Return the aileron control system to normal.

27.0.2.2 AILERON BOOSTER ASSEMBLY LEAKAGE CHECK.

a. Obtain two sources of external hydraulic power and connect one to the normal hydraulic system ground test connections and one to the standby hydraulic system ground test connections. Adjust both hydraulic power sources to deliver 2,700 PSI at 10 gallons per minute.

b. Connect 28-volt, DC power to the airplane service connections, and energize the main 28-volt, DC bus.

27.0.2.3 (Cont).

c. **Make sure the No. 1 and No. 2 boost control circuit breakers are engaged, and that the AILERON SYS 1 and AILERON SYS 2 BOOST HYD PRESS switches on the pilot's control column cover are in the "ON" (guarded) position.**

d. With hydraulic power applied to both portions of the booster assembly, cycle the aileron booster 40 complete cycles, closely observing leakage from seals during the last 25 cycles.

RESULT: During the last 25 cycles, leakage should not exceed one drop from either of the actuator rod end seals, and it should not exceed two drops from the center drain of the actuator. There should be no leakage from the static seals.

e. Upon completion of this check, reduce hydraulic pressure to zero, disconnect the power sources, and return the aileron control system to normal.

27.0.3 TROUBLESHOOTING.

27.0.3.1 AILERON CONTROL SYSTEM.

PROBABLE CAUSE	ISOLATION	REMEDY
AILERONS FAIL TO FAIR TO THE WING WITHIN ACCEPTABLE TOLERANCES WHEN THE CONTROL WHEELS ARE CENTERED WITHIN ACCEPTABLE TOLERANCES.		
Control cables from control wheel pulleys to torque tube incorrectly rigged.	Perform an operational checkout of the aileron control system according to the instructions in this section.	Re-rig the aileron control system as needed. Rigging instructions are included in this section.
Control cables from torque tube to tension regulator incorrectly rigged.	Perform an operational checkout of the aileron control system according to the instructions in this section.	Re-rig the aileron control system as needed. Rigging instructions are included in this section.
Pushrod mechanism improperly adjusted.	Perform an operational checkout of the aileron control system according to the instructions in this section.	Re-rig the aileron control system as needed. Rigging instructions are included in this section.
EXCESSIVE TORQUE REQUIRED TO ROTATE CONTROL WHEELS WITH BOOST HYDRAULIC PRESSURE ON.		
One or both of the booster shutoff and bypass valves improperly positioned.	Check that the BOOST HYD PRESS switches are correctly positioned.	Position the BOOST HYD PRESS switches correctly.
	Check operation of the shutoff and bypass valve.	If shutoff and bypass valves are defective, replace the booster assembly and rig the aileron control system.
Hydraulic pressure and flow insufficient.	Make sure that 2,700 PSI with 10 gallons per minute flow is available.	Adjust the external hydraulic power source as necessary.
Binding in pushrod mechanism or cable system.	Check the cable and pushrod systems for binding.	Eliminate binding as needed.
Defective booster assembly.	Check the operation of the booster assembly.	Replace the defective booster assembly.
PILOT'S AND COPILOT'S CONTROL COLUMNS NOT CENTERED AT SAME TIME.		
Cable from control wheel pulley to torque tube improperly rigged.	Perform an operational checkout of the aileron control system to determine which control wheel is improperly adjusted.	Rig the aileron control system according to instructions in this section

27.0.4 REMOVAL AND REPLACEMENT.

27.0.4.1 REMOVAL OF THE AILERON BOOSTER ASSEMBLY.

WARNING

Depressurize the normal and standby hydraulic systems, de-energize the auxiliary hydraulic pump circuits, and warn personnel not to move the controls.

a. Remove the left aileron booster and trailing edge flap motor access door (see figure 52-5), and the small booster access door above the left wing fillet.

b. Disconnect the rod which connects the tension regulator input lever to the booster feel lever, at the input lever. Do not disturb cable rigging.

c. Disconnect the aileron pushrods from the booster output arm.

d. Disconnect and remove the trailing edge flap torque tube and the support fitting below the booster.

e. Disconnect the electrical connectors from the booster and secure them to avoid damage.

f. Disconnect and remove all hydraulic lines from the area through which the booster has to be removed, including bulkhead fittings and the unions and angle fittings from the boost control. Cap all tube and fitting openings.

g. Disconnect the booster input rod from the tension regulator.

h. Remove booster attaching links and bolts from the fuselage attaching points near the fuselage tunnel.

i. Support the outboard side of the booster, loosen the two internal wrenching bolts and raise or remove bolts until the two barrel spacers can be taken out. Note shims that may be used here for replacement on reinstallation.

j. In some cases it may be necessary to disconnect the tension regulator support bracket from the booster. Note spacer shims between the bracket and the booster.

k. Move the output lever of the booster inboard then work the booster outboard and down until it clears the structures.

CAUTION

Any time a booster assembly is removed or a control surface is disconnected from a hydraulic booster assembly, or the hydraulic systems are drained of fluid, an external contour-type clamp shall be installed to lock the control surface to the adjacent structure. This will prevent damage if the airplane is subjected to wind gusts. As long as the hydraulic systems are properly serviced and the control surfaces are connected to the booster assembly, normal ground gusts will not damage the controls.

27.0.4.2 REPLACEMENT OF THE AILERON BOOSTER ASSEMBLY.

a. Reverse the removal procedure.

b. Bleed the aileron control system.

c. Check aileron control system operation, and rig the system as needed.

Note

Prior to installation, the boost package should be placed on a bench with the damper fill plugs up. These are the two small slotted pan head plugs lock-wired together on the same level as the base of the solenoids. The plugs should be removed and the cavity below filled to overflowing with clean MIL-H-5606 fluid while moving the pilot lever back and forth by hand. Reinstall plug, seals (MS28775-007) and lockwire.

27.0.4.3 REMOVAL OF THE AILERON BOOSTER TENSION REGULATOR.

a. Gain access to the booster assembly by removing the access panel, aft of the left main wheel well.

b. Loosen the cables between the aileron torque tube and the tension regulator on top of the booster assembly (inside the airplane). Refer to aileron rigging instructions for locations or turnbuckles.

c. Remove the bolts from the torque tube input arm. These bolts hold the regulator shaft to the booster assembly.

d. With no tension on the cables to the tension regulator, pull the cable guard pins, and lift the regulator clear of the airplane structure.

e. Disconnect the cables from the tension regulator, and remove the tension regulator from the airplane.

27.0.4.4 REPLACEMENT OF THE AILERON BOOSTER TENSION REGULATOR.

a. Inspect the upper and lower tension regulator support bearings. Replace if necessary.

b. Reverse the removal procedures.

c. Rig the aileron control system as needed.

d. Perform an operational checkout of the aileron control system.

e. Secure all connections. Make sure all guard pins are properly installed and all cables are riding correctly on their pulleys.

f. Replace all access panels.

27.0.4.5 REMOVAL OF THE AILERON.

Note

Step (c) is necessary only when removing the left aileron.

WARNING

Refer to Chapter 51 for precautions to be observed when working around depleted uranium counter-balance weights.

a. Open the access door on the underside of the wing trailing edge near the inboard end of the aileron, and disconnect the aileron pushrod from the aileron.

b. Disconnect all bonding straps from the aileron.

c. Disconnect the wiring from the aileron trim actuator (left aileron only), and remove the clamps as necessary to free the wiring from the aileron.

d. Remove the hinge segments at aileron stations 24 and 71 to expose the ends of the three hinge pins.

e. While supporting the aileron, remove the aileron hinge pins, and remove the aileron.

27.0.4.6 REPLACEMENT OF THE AILERON.

a. Insert a small amount of Lubriplate No. 130A in the holes of the hinge nodes so that insertion of the hinge pin will carry the lubricant into the hinges.

b. Reverse the removal procedures; after replacement of the aileron hinge pin, apply Specification MIL-L-7870 lubricant to the hinges while moving the aileron through the full range of travel. Wipe off excess lubricant with clean cloths.

c. Rig the aileron control system as needed.

d. Perform an operational checkout of the system.

e. Secure all connections, and replace the access doors.

27.0.4.7. REMOVAL AND REPLACEMENT OF AILERON BOOSTER FILTERS. Before removing filter caps, thoroughly clean adjacent area with an approved cleaning solvent. Apply steady pressure when removing filter elements. A sudden thrust of torque may snap the element stem. Use an Allen wrench to remove filter elements.

a. Visually inspect filter elements for breaks or cracks, and discard elements which show signs of damage.

b. Clean elements using one of the following methods:

(1) Send to an FAA Approved Commercial Filter Cleaning facility.

(2) Clean filter element using SAE Specification ARP725 as a guide.

Note

The filter element is rated at 10 microns nominal, 25 microns absolute.

(3) Return filter element to manufacturer for cleaning,

Aircraft Porous Media, Inc.
6301 49th St. N.
Pinellas Park, Florida 33565
Phone: 813-522-3111

c. Reassemble filter elements in reverse order of disassembly.

Note

Discard all used packing, and backup rings and replace with new parts.

Place new packing MS28778-10 on filter elements. Install elements in manifold filter cavities, and tighten to 60 to 75 inch-pounds torque, using an Allen wrench. Place new packing MS28775-128 and new backup rings MS27595-128 on filter caps. Install filter caps on manifold housing and tighten to 300 to 348 inch-pounds torque, using a 1-7/8-inch wrench.

27.0.4.8 BOLT TORQUE REQUIREMENTS. Bolts used with castellated nuts and cotter pins at the pushrod and bellcrank ends should be tightened to a torque value of at least 5 to 10 inch-pounds to a position where the cotter pin can be installed. When possible, such installations should be made with the nut down. Other torque values should be in accordance with the values referenced in Chapter 51, in this manual, unless otherwise specified.

27.0.5 ADJUSTMENT.

Test Equipment Required

Nomenclature	Description	Use and Application
Portable Aircraft Hydraulic System Test Stand	15 GPM at 2,700 PSI Capacity (MIL-H-5606 hydraulic fluid)	To pressurize hydraulic systems.

Tools Required

Nomenclature	Description	Use and Application
Control Surfaces Rigging Kit	Lockheed P/N 11019-MEU	To determine the faired position of the controls.
Aileron and Elevator Protractor	Lockheed P/N 11069-MEU	To measure aileron travel.
Cable Tensiometer	Type T-5 Pacific Scientific Co. 45402 or equal.	To measure cable tension.

27.0.5.1 AILERON CONTROL SYSTEM RIGGING INSTRUCTIONS. (See figure 27-2.)

27.0.6 INSPECTION.

27.0.6.1 AILERON CONTROL CABLE CONNECTOR INSPECTION (AIRPLANES 5001 THRU 5045). After each reassembly of the aileron lockclad control cable connectors, the following inspection should be made to determine whether the cable pin is installed.

WARNING

When the retainer is safety-wired in position, the pin location is not visible. The safety-wire may sustain the rigging tension loads without the pin. Therefore, it is possible that the pin could be omitted and its absence remain undetected until the connector separated under loads imposed by flight operation.

a. Measure the distance the male end protrudes out of the female end.

b. Compare the measurement with the dimensions shown in figure 27-3 to make certain the pin has been installed.

The projection of the male end from the female end will be approximately 0.31 inch greater without the pin than it would be with the pin installed.

27.0.6.2 INSPECTION FOR EXCESSIVE LOOSENESS IN AILERONS. If total aileron motion, measured at the trailing edge with the aileron in the faired position, exceeds 0.57 inch when the aileron is moved by hand, further inspection of the individual joints and bearings is required.

Note

The hole in the hinge halves should not exceed 0.110 inch diameter. The hinge pins should not be less than 0.070 inch diameter.

Note

Any bearings with radial play or axial play more than 0.015 inch should be replaced.

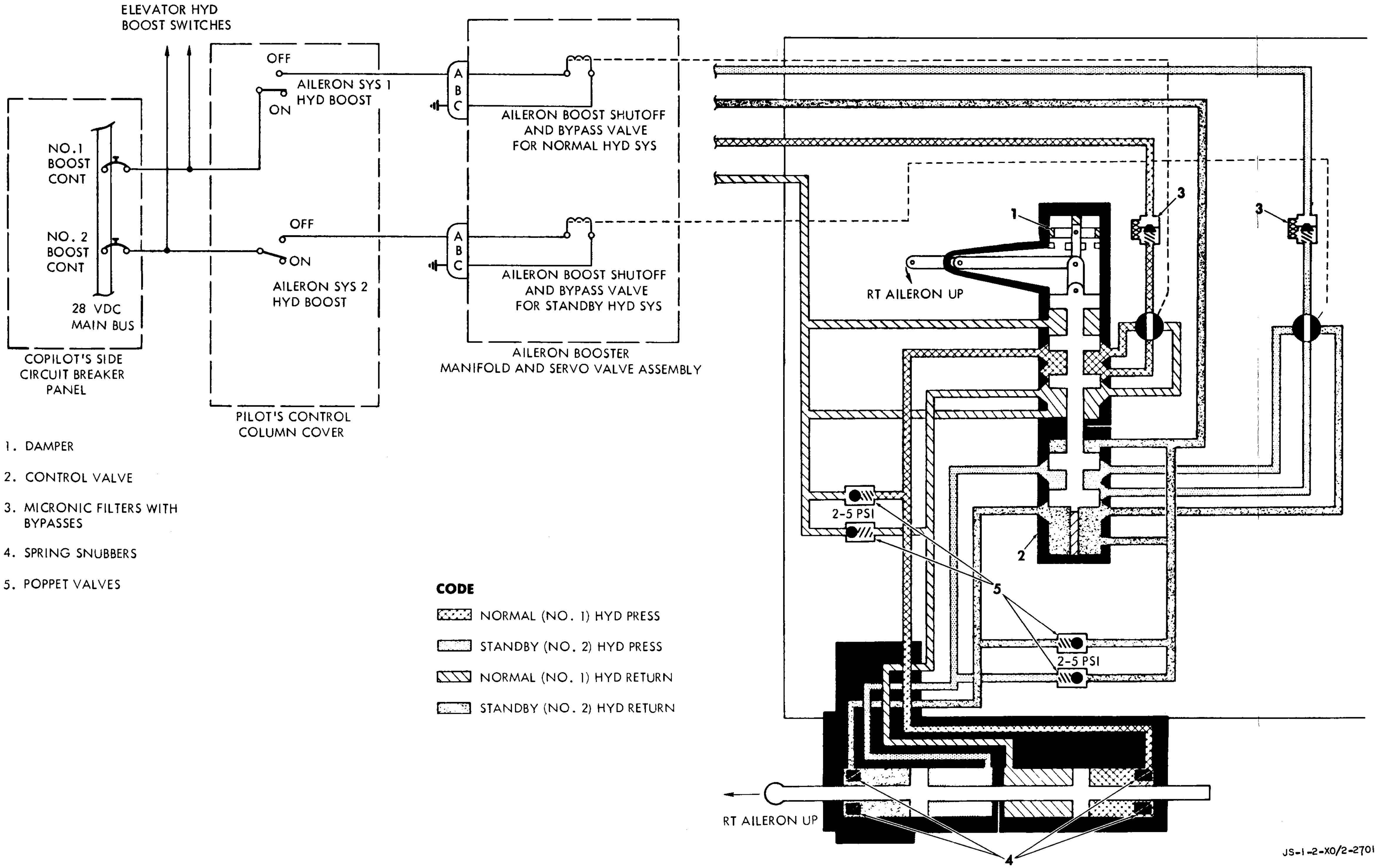

ELEVATOR HYD BOOST SWITCHES
OFF
AILERON SYS 1 HYD BOOST
ON
A
B
C
AILERON BOOST SHUTOFF AND BYPASS VALVE FOR NORMAL HYD SYS
NO. 1 BOOST CONT
NO. 2 BOOST CONT
28 VDC MAIN BUS
COPILOT'S SIDE CIRCUIT BREAKER PANEL
OFF
ON
AILERON SYS 2 HYD BOOST
PILOT'S CONTROL COLUMN COVER
AILERON BOOST SHUTOFF AND BYPASS VALVE FOR STANDBY HYD SYS
AILERON BOOSTER MANIFOLD AND SERVO VALVE ASSEMBLY
1
RT AILERON UP
3
3
2-5 PSI
2
5
2-5 PSI
RT AILERON UP
4
1. DAMPER
2. CONTROL VALVE
3. MICRONIC FILTERS WITH BYPASSES
4. SPRING SNUBBERS
5. POPPET VALVES
CODE
NORMAL (NO. 1) HYD PRESS
STANDBY (NO. 2) HYD PRESS
NORMAL (NO. 1) HYD RETURN
STANDBY (NO. 2) HYD RETURN
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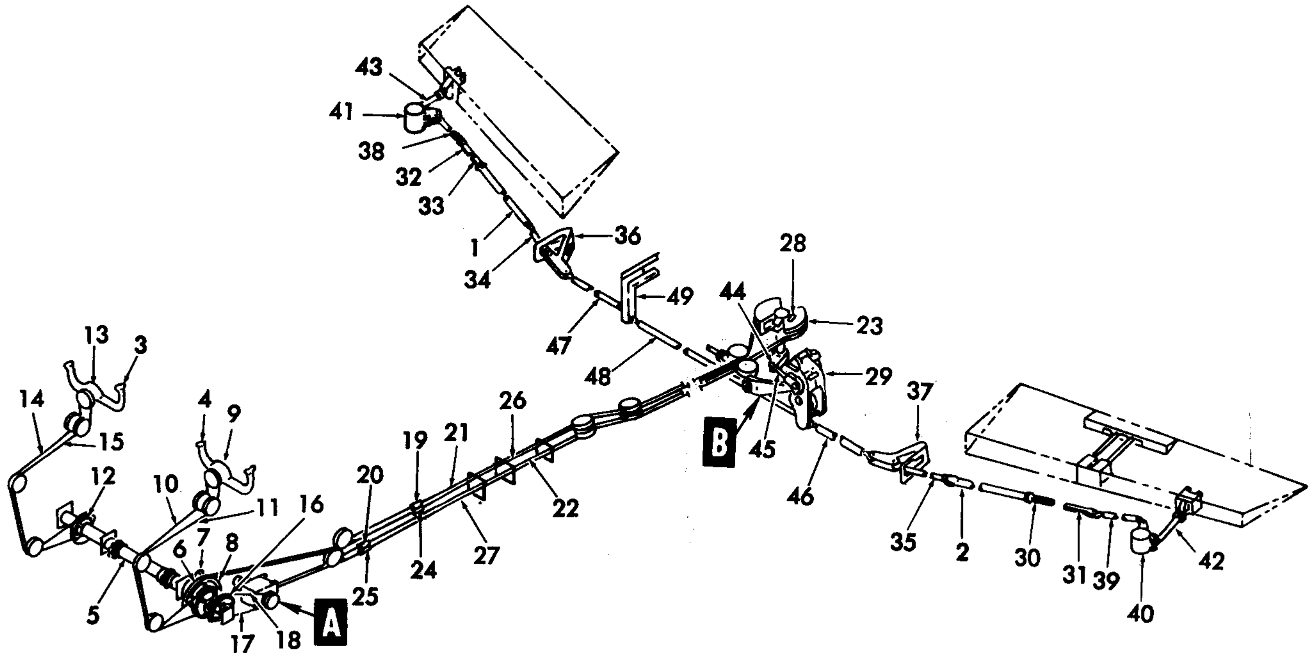
Figure 27-1. Aileron Control System Schematic Diagram



PREPARATION

STEP 1

INSPECT ALL AILERON CONTROL CABLES, PULLEYS, AND FAIRLEADS FOR EVIDENCE OF EXCESSIVE FRICTION; MAKE SURE ALL GUARD PINS ARE IN PLACE, AND THAT NO RUBBING OR INTERFERENCE EXISTS IN THE SYSTEM.

STEP 2

INSPECT THE CONTROL OUTPUT SYSTEM (FROM THE BOOSTER ASSEMBLY TO THE AILERON CONTROL SURFACES) TO MAKE SURE ALL RODS ARE INSTALLED CORRECTLY AND ARE RIDING ON THEIR ROLLERS. MAKE SURE THE ROLLERS ARE FREE TO ROTATE. ROLLER WEAR IN THE RODS (1) AND (2) SHOULD NOT EXCEED A DEPTH OF 0.010 INCH AT ANY ONE POINT.

NOTE

CHECK ROLLER ECCENTRICS TO ENSURE THAT THE FORCE REQUIRED TO MOVE THE RODS (1) AND (2) DOES NOT EXCEED 2 POUNDS PER ROD. DISCONNECT THE ROD ASSEMBLIES (1) AND (2) AT BOTH ENDS. ADJUST THE INBOARD ROLLERS SO THAT THE CLEARANCE IS 0.005 TO 0.010 INCH AS THE ROD IS PUSHED ALTERNATELY BETWEEN BOTH ROLLERS. SET THE TWO OUTBOARD ROLLERS SNUGLY AND THEN LOOSEN THEM AS REQUIRED UNTIL THE FORCE REQUIRED TO MOVE EACH ROD IS LESS THAN 2 POUNDS. RECONNECT THE RODS.

STEP 3

WITH NO HYDRAULIC PRESSURE ON THE AILERON BOOSTER SYSTEM, MOVE THE CONTROL WHEELS (3) AND (4) THROUGH SEVERAL COMPLETE CYCLES TO MAKE SURE THERE IS NO INTERFEARENCE IN EITHER THE BOOSTER ASSEMBLY OR THE CCNTROL OUTPUT SYSTEM FROM THE BOOSTER ASSEMBLY TO THE AILERON CONTROL SURFACES.

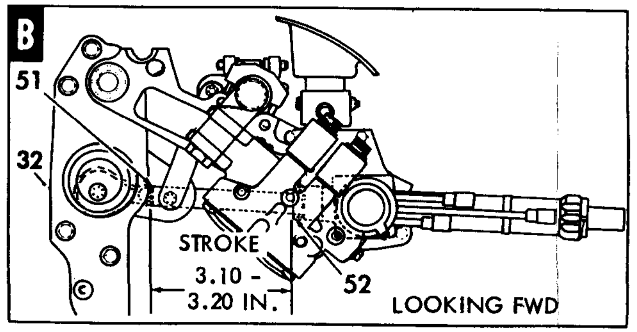
RIGGING THE BOOSTER INPUT SYSTEM

STEP 4

FIND THE MID-TRAVEL POSITION OF THE BOOSTER. THIS POSITION IS FOUND BY MEASURING THE TOTAL TRAVEL OF THE ACTUATOR, AND THEN, WITH THE ACTUATOR FULLY EXTENDED IN EITHER DIRECTION, MOVE THE ACTUATOR IN THE OPPOSITE DIRECTION ONE HALF OF THE TOTAL TRAVEL DIMENSION.

NOTE

TOTAL TRAVEL OF THE ACTUATOR IS FOUND BY ROTATING EITHER OF THE CONTROL WHEELS CLOCKWISE AND THEN COUNTERCLOCKWISE UNTIL THE WHEEL REACHES ITS FULL LIMIT OF TRAVEL. MEASURE THE DISTANCE FROM THE FACE OF THE ACTUATOR END CAP (52) TO THE END OF THE ACTUATOR ROD (51) WITH THE CONTROL WHEEL IN THE FULL COUNTER-CLOCKWISE POSITION, AND THEN IN THE FULL CLOCKWISE POSITION. THE TOTAL TRAVEL OF THE ACTUATOR ROD SHOULD BE BETWEEN 3.10 AND 3.20 INCHES. (SEE VIEW 'B').



STEP 5

LOCK THE TENSION REGULATOR (23) IN THE MID-TRAVEL POSITION WHICH WAS DETERMINED IN THE PRECEEDING STEP, BY CHOCKING THE TENSION REGULATOR SECURELY.

NOTE

THE LENGTH OF THE ROD ASSY (45) SHOULD BE BETWEEN 9.80 AND 9.90 INCHES FROM CENTER TO CENTER AFTER ADJUSTMENT.

STEP 6

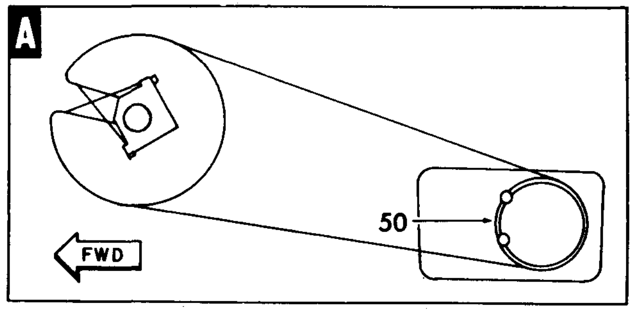
INSERT A PIN THROUGH THE RIG PIN HOLE (6) OF THE LEVER (7) AND THE RIG PIN HOLE IN THE ADJACENT STRUCTURE. ALTERNATELY ADJUST THE CABLE TERMINALS AT THE SECTOR (8) TO ALIGN THE PILOT'S WHEEL CONTROL INDEX MARK (9) WITH THE INDEX MARK ON THE TOP OF THE CONTROL COLUMN. THE CABLES MUST BE ADJUSTED UNTIL THE INDEX MARKS ALIGN TO WITHIN 0.03 INCH. SIMULTANEOUSLY, DEFLECT THEN RELEASE CABLES (10) AND (11) SEVERAL TIMES TO EQUALIZE CABLE TENSION TO A VALUE OF 50±5 POUNDS AT 70 DEGREES FAHRENHEIT. (SEE CHART A.)

STEP 7

REPEAT STEP 6 FOR THE COPILOT'S CONTROL WHEEL.

STEP 8

WITH THE AILERONS IN THE FAIRED POSITION ALTERNATELY ADJUST THE CABLE TERMINALS AT THE SECTOR (16) TO OBTAIN THE APPROXIMATE CABLE WRAPAROUND SHOWN IN VIEW "A." RIG CABLES (17) AND (18) TO A TENSION VALUE OF 50±5 POUNDS AT 70 DEGREES FAHRENHEIT. (SEE CHART B.)



NOTE

EACH CABLE SHOULD BE WRAPPED AROUND THE SERVO DRUM (50) 1-1/2 TIMES, AND THE CABLE ENDS SHOULD BE NEAR THE EDGES OF THE SERVO DRUM WITH THE CABLES WRAPPING TOWARD EACH OTHER.

STEP 9

WITH THE TENSION REGULATOR (23) STILL IN THE LOCKED POSITION (STEP 5) ALTERNATELY ADJUST THE TURNBARRELS (20) AND (24) (UNDER THE FLOOR BETWEEN FUSELAGE STATIONS 240 AND 250) TO TIGHTEN CABLES (22) AND (26) UNTIL THE TENSION INDICATOR (28) ON THE TENSION REGULATOR (23) IS POSITIONED TO WITHIN 3 TO 5 MARKS FROM "H" (HOT) AT 70 DEGREES FAHRENHEIT (SEE CHART C).SIMULTANEOUSLY DEFLECT,THEN RELEASE CABLES (22) AND (26) SEVERAL TIMES TO EQUALIZE CABLE TENSION.

STEP 10

REPEAT THE PROCEDURE OF STEP 9 FOR THE INBOARD AND OUTBOARD CABLES (21) AND (27) BY ADJUSTMENT OF TURNBARRELS (19) AND (25). KEEP THE TENSION INDICATOR POINTER IN THE SAME POSITION AS IN STEP 9.

STEP 11

TAKE TENSIOMETER READINGS OF ALL FOUR CABLES AND MAKE ANY ADJUSTMENTS NECESSARY TO HAVE THEM WITHIN A 10 POUND TOLERANCE BAND. REMOVE THE RIG PIN FRCM LEVER (7). SAFETY ALL TURNBARRELS.

STEP 12

WITH THE BOOSTER STILL IN THE MID-TRAVEL POSITION (REF STEPS 4 AND 5), ADJUST RODS (46) AND (47) FOR INSERTION OF RIG PINS THROUGH IDLERS (36) AND (37) AND ADJACENT AIRCRAFT STRUCTURE. CHECK THE ROD END WITNESS HOLES FOR THREAD ENGAGEMENT.

NOTE

THE USE OF THE RIG PINS IS OPTIONAL IN THIS STEP.

STEP 13

CHECK ROD ASSEMBLIES (34) AND (35) FOR LENGTHS OF 20.86 ± .03 INCHES CENTER TO CENTER, ADJUST ROD ASSEMBLIES (42) AND (43) TO LENGTHS OF 10.06 INCHES CENTER TO CENTER. ADJUST ROD ASSEMBLIES (38) AND (39) TO FAIR THE AILERON CONTROL SURFACE WITH THE WING.

NOTE

TO DETERMINE AILERON CONTOUR WITH THE WING, USE CONTOUR TEMPLATES FROM THE CONTROL SURFACES RIGGING KIT. LOCKHEED P/N 11019-MEU.

STEP 14

REMOVE THE RIG PIN FROM THE TENSION REGULATOR (23). WITH THE BOOST OFF, MOVE THE CONTROL WHEEL THROUGH THE FULL RANGE AND CHECK FOR INTERFERENCES. WITH THE BOOST ON, CHECK FOR FULL TRAVEL DEFLECTIONS ON BOTH AILERONS OF 22 1/2 DEGREES UP, AND 18 DEGREES DOWN.

STEP 15

IF FULL AILERON DEFLECTIONS ARE NOT WITHIN ONE-HALF OF A DEGREE OF THE SPECIFIC LIMITS, ADJUST RODS (42) AND (43) TO OBTAIN PROPER LIMITS.

NOTE

WHEN ADJUSTING RODS (42) AND (43), IT WILL BE NECESSARY TO READJUST RODS (38) AND (39) TO MAINTAIN FAIRED CONDITION.

STEP 16

TURN HYDRAULIC BOOST ON. CHECK COMPLETE SYSTEM FOR PROPER CLEARANCES, THROUGH THE FULL TRAVEL.

STEP 17

TURN HYDRAULIC BOOST OFF. INSPECT AND SAFETY ALL ROD ENDS AND CHECK ALL WITNESS HOLES FOR THREAD ENGAGEMENT.

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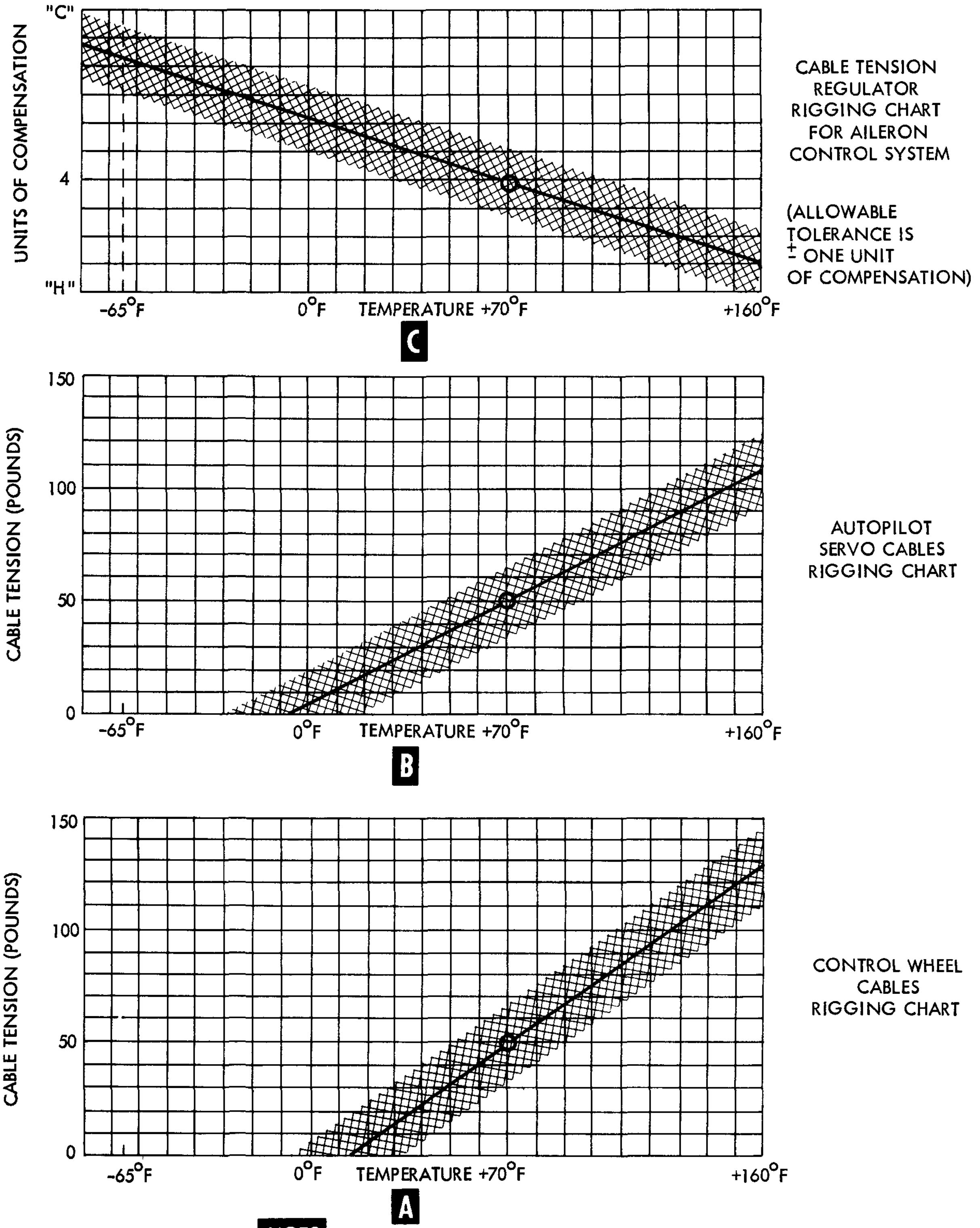
Figure 27-2. Aileron Control System Rigging Instructions (Sheet 1 of 2)



NOTE

THIS CHART IS VALID ONLY WHEN THE TEMPERATURE OF THE CABLES AND THE STRUCTURE HAVE STABILIZED.

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Figure 27-2. Aileron Control System Rigging Instructions (Sheet 2 of 2)

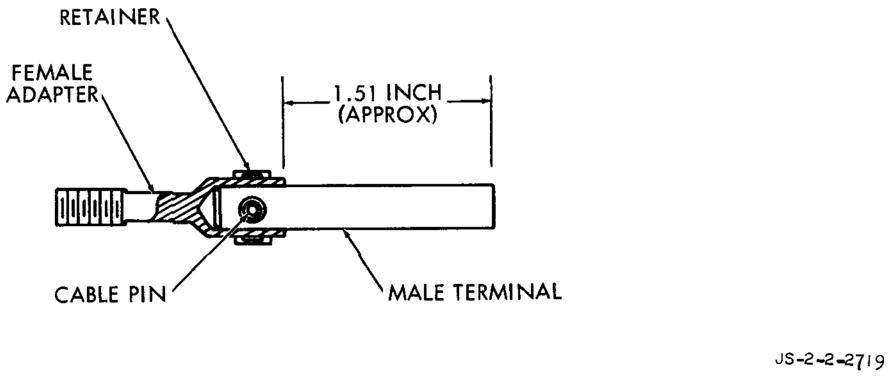


Figure 27-3. Aileron Lockclad Cable Connector Inspection

SECTION 27.1

ELEVATOR CONTROL SYSTEM

Contents Page

DESCRIPTION AND OPERATION 27-15
- General 27-15
- Operation 27-15
- Control Columns 27-16
- Control Column Covers 27-16
- Control Cables 27-16
- Autopilot Servo Drive 27-16
- Torque Tube Assembly 27-16
- Cable Tension Regulator 27-16
- Booster Assembly 27-16
- Control Switches 27-16
- Booster Assembly Filters and Screens 27-17
- Booster Assembly Solenoids 27-17
- Pushrod Mechanism 27-17
- Elevator Centering Spring 27-17

OPERATIONAL CHECKOUT 27-18
- Elevator Control System 27-18
- Elevator Booster Assembly Leakage Check 27-18A

TROUBLESHOOTING 27-19
- Elevator Control System 27-19

REMOVAL AND REPLACEMENT 27-20
- Removal of the Elevator Booster Assembly 27-20
- Replacement of the Elevator Booster Assembly 27-20
- Removal of the Elevator Booster Tension Regulator 27-20
- Replacement of the Elevator Booster Tension Regulator 27-20
- Removal of the Elevator 27-20A
- Replacement of Elevator 27-20A
- Removal of Elevator Torque Tube 27-20A
- Replacement of Elevator Torque Tube Bearing 27-20A
- Replacement of Elevator Torque Tube 27-21
- Removal and Replacement of Elevator Booster Filters. 27-21
- Bolt Torque Requirements 27-21

ADJUSTMENT 27-21
- Elevator Rigging Instructions 27-21

INSPECTION 27-21
- Elevator Control Cable Connector Inspection 27-21

FIGURES
- 27-4. Elevator Hinge Pin Clearance 27-22
- 27-5. Elevator Lockclad Cable Connector Inspection 27-22
- 27-6. Elevator Control System Schematic Diagram 27-23
- 27-7. Elevator Control System Rigging Instructions 27-24

27.1.1 DESCRIPTION AND OPERATION.

27.1.1.1 GENERAL. (See figures 27-6 and 27-7.) Airplane pitch is controlled by a hydraulically boosted elevator control system. This system is made up of a pilot's control column, a copilot's control column, an elevator torque tube assembly, dual cable systems, a cable tension regulator, an elevator booster assembly, and a pushrod mechanism connected to two elevator control surfaces. See Section 51.5 for instructions on rebalancing the elevator and Section 59.8 for typical repairs to the elevator and elevator control items.

27.1.1.2 OPERATION. (See figures 27-6 and 27-7.) Moving either the pilot's or copilot's control column rotates the torque tube under the cockpit flooring. Rotating this torque tube positions the control valve of the booster assembly to operate the actuating

27.1.1.2 (Cont).

cylinder. An output arm on the actuator assembly actuates the pushrod mechanism to position the elevator control surfaces according to pilot commands. A mechanical followup linkage connects the control valve input linkage to the output arm of the actuator and normally follows the boosted movement of the booster assembly. In the absence of hydraulic power, this followup linkage takes over to manually position the pushrod mechanism for mechanical control of the elevators. During normal operation of the elevators, both the normal and the standby hydraulic systems power the elevator booster assembly, but either system can be used independently of the other with no increase in the amount of effort required. In the absence of normal and standby hydraulic system power, the auxiliary hydraulic pump can be used to power the booster assembly. In the absence of any hydraulic power, the mechanical followup linkage takes over to provide manual control of the elevators. A centering spring attached to the elevator booster assembly returns the elevators to the neutral position when no input commands are present.

27.1.1.3 CONTROL COLUMNS. (See figure 27-7.) There are two control columns: one for the pilot and one for the copilot. Moving either control column directly rotates the elevator torque tube assembly under the cockpit flooring. Mechanical stops attached to the airplane structure contact a lever on the copilot's side of the torque tube assembly to limit control column travel by limiting torque tube rotation.

27.1.1.4 CONTROL COLUMN COVERS. Each control column is partially covered by a stationary cover which hides the upper section of the control columns but allows the heads of the columns to move fore and aft. These control column covers also provide a mounting area for switches such as the BOOST HYD PRESS switches.

27.1.1.5 CONTROL CABLES. (See figure 27-7.) Dual cable systems connect the elevator torque tube assembly to the tension regulator on the elevator booster assembly. On the pilot's side, two cables attach to a lever on the torque tube assembly, and run aft approximately to fuselage station 480 where they connect to two more cables. The latter cables continue aft to between fuselage stations 664 and 700. At this point, two more cables connect to them and continue to the tension regulator on the elevator booster assembly. The cables on the pilot's side are similar to those on the copilot's side, except that an autopilot servomotor is connected into the system between the torque tube assembly and the tension regulator. Both lockclad and flexible, prestretched, steel-wire cables are used in the elevator control system. (Refer to Section 27.8.)

27.1.1.6 AUTOPILOT SERVO DRIVE. An autopilot servomotor drives the cable system on the copilot's side to position the elevators in response to autopilot system signals. This servomotor drives a drum connected to the elevator control cables. A clutch mechanism automatically disengages the drum from the servomotor when the servomotor is not operating.

27.1.1.7 TORQUE TUBE ASSEMBLY. An elevator torque tube assembly under the cockpit flooring is made up of three sections. The section on the pilot's side drives one cable system, and the section on the copilot's side drives another cable system. The center section interconnects the control columns for dual control reliability. Where the sections join, sprockets butt together and chains cover them to form a union. Four bearing assemblies support the torque tube assembly.

27.1.1.8 CABLE TENSION REGULATOR. A cable tension regulator is attached to the bottom of the elevator booster assembly to keep the proper cable tension between the booster assembly and the elevator torque tube. Cables from the elevator torque tube connect to the tension regulator. Moving the cables rotates the regulator to position the control valve in the booster assembly. The tension regulator is a spring-loaded device which keeps the proper cable tension between -60 and +165°F. An indicator on top of the tension regulator shows cable tension. This indicator has a scale of 10 divisions with "H" and "C" as the limits. "H" is for high or hot, and "C" is for low or cold. A chart in the elevator rigging instructions gives the proper reading for temperatures between -65 and +160°F.

27.1.1.9 BOOSTER ASSEMBLY. (See figure 27-6.) The elevator booster assembly consists of a manifold and servo valve sub-assembly, an actuator sub-assembly, and a mechanical followup linkage. The manifold and servo valve sub-assembly contains a tandem servo (control) valve, two shutoff and bypass valves, two filters, six split screens, four check or poppet valves, and interconnecting fluid passages. The actuator sub-assembly consists of a tandem actuating cylinder and fluid passages. A mechanical followup linkage connects the control valve input linkage to the actuating cylinder output arm. Normally this linkage follows the hydraulic boosting action of the booster assembly, but in the absence of hydraulic power it takes over to position the elevators manually.

27.1.1.10 CONTROL SWITCHES. Hydraulic power to the booster assembly is controlled by two BOOST HYD PRESS switches on the pilot's control column: an ELEVATOR SYS 1 switch and an ELEVATOR SYS 2 switch. These switches control the shutoff and bypass valves in the booster assembly. The SYS 1 switch controls the normal system power, and the SYS 2 switch controls the standby system power. Placing either switch in the "ON" position de-energizes the corresponding valve to the normally open position. Placing either switch in the "OFF" position energizes the corresponding valve to block hydraulic power from the booster assembly and at the same time opens a run-around circuit to prevent hydraulic

27.1.1.10 (Cont).

locks in the portion of the booster assembly which is not receiving hydraulic power. These switches are in a group of four BOOST HYD PRESS switches. All of these switches are guarded in the normal "ON" position.

27.1.1.11 BOOSTER ASSEMBLY FILTERS AND SCREENS. (See figure 27-6.) Normal hydraulic fluid entering the booster assembly passes through a micronic filter assembly which contains a bypass valve. This bypass valve opens to relieve hydraulic pressure around clogged filter elements. Fluid from this filter passes through a screen, through the control valve, and through the actuating cylinder. Filters and screens in the standby hydraulic system portion of the booster assembly are the same as those for the normal hydraulic system. Filters are accessible for cleaning or replacement, but screens cannot be removed unless the booster assembly is disassembled. Filters should be removed and cleaned every 200 flight hours.

Note

The late model booster assembly is provided with filter units that do not contain a bypass valve as used in some earlier units.

CAUTION

When removing filter, use extreme caution to prevent any metal chips or foreign matter that are trapped on the filter from falling back into the open cavity. After filter is removed, clean filter cavity with a clean, lint-free cloth. When reinstalling filter, use MIL-S-8660 compound, or equivalent to position O-ring on filter shoulder. This will prevent the O-ring from dropping down around the threads and getting damaged when the filter is installed.

27.1.1.12 BOOSTER ASSEMBLY SOLENOIDS. The booster assembly contains two solenoids, one in system No. 1 and one in system No. 2. When installing a solenoid, use MIL-S-8660 compound or equivalent to position the O-ring on the solenoid shoulder. This will prevent the O-ring from dropping down around the threads and getting damaged when solenoid is installed.

27.1.1.13 PUSHROD MECHANISM. The output arm of the elevator booster assembly connects to a pushrod which in turn connects to a bellcrank. This bellcrank is attached to two pushrods. Each of these pushrods connects to a horn on one of the elevators. As the booster output arm moves, the pushrod positions the bellcrank and thus moves horns and elevators.

27.1.1.14 ELEVATOR CENTERING SPRING. (See figure 27-7, view B.) An elevator centering spring assembly is attached between the airplane structure and the shaft of the tandem servo control valve. This spring assembly centers the booster assembly in the absence of input commands, to fair the elevators.

27.1.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Portable Aircraft Hydraulic System Test Stand	15 GPM at 2,700 PSI Capacity (MIL-H-5606 hydraulic fluid)	To pressurize hydraulic systems.

Tools Required

Nomenclature	Description	Use and Application
Control Surfaces Rigging Kit	Lockheed P/N 11019-MEU	To determine the position of the elevators.
Aileron and Elevator Protractor	Lockheed P/N 11069-MEU	To measure elevator travel.

27.1.2.1 ELEVATOR CONTROL SYSTEM.

a. Connect 28-volt DC power to the airplane service connections.

b. Make sure the hydraulic system reservoirs and accumulators are properly serviced. Make sure the No. 1 and No. 2 boost control circuit breakers are engaged.

c. Connect external hydraulic power to the normal hydraulic system external hydraulic connections. Adjust the power source to deliver no flow at 3,000 PSI and 10 gallons per minute at 2,700 PSI.

d. Make sure the ELEVATOR SYS 1 and ELEVATOR SYS 2 BOOST HYD PRESS switches on the pilot's control column cover are in the "OFF" position.

e. Set the pitch trim actuator in the zero trim position by setting the actuator length at 28.37 (±0.03) inches. Measure the length according to the instructions in Section 27.3 on timing the actuator halves.

f. Connect a force-measuring device to the center of the hub of the pilot's control wheel.

g. Apply 3,000-PSI hydraulic pressure to the normal (No. 1) hydraulic system, and place the ELEVATOR SYS 1 switch in the "ON" position.

h. Move the pilot's control column to the extreme forward position, and measure the distance between the top of the control column and the face of the main instrument panel.
RESULT: There should be clearance of 2.70 (+ 0.30, -0.10) inches between the top of the control column and main instrument panel.

i. Return the control column to the elevator-faired position. Use an elevator contour board to fair one elevator; then, while holding it faired, measure the amount of deflection of the other elevator. Measure the deflection at the inboard, trailing edge of the elevators.
RESULT: With one elevator faired, the other elevator should be no more than 0.06 inch from the faired position.

j. Attach protractors to the elevators and zero them at the elevator-faired position.

k. Use the force-measuring device to move the control column to the extreme aft position. Measure the amount of force required to start the column moving and the amount necessary to obtain full elevator travel. Measure the amount of elevator deflection above the faired position.
RESULT: The amount of force required to start the control column moving from the elevator-faired position should not exceed 7 pounds. This force should increase as column travel increases, but it should not exceed 20 pounds at full elevator travel. With the control column in the extreme aft position, the elevators should be 13 (±1/4) degrees up from the faired position. Check both elevators for travel.

l. Return the control column to the elevator-faired position; then use the force-measuring tool to move the control column to the extreme forward position. Measure the amount of force required to start the control column moving and the amount required to obtain full elevator travel. Measure the amount of elevator deflection below the faired position.

RESULT: The amount of force required to start the control column moving should not exceed 6.5 pounds and should increase as control column travel increases. The amount of force required to obtain full elevator travel should not exceed 20 pounds. With the control column in the extreme forward position, both elevators should be 16-3/4 (±3/4) degrees down from the faired position.

m. Repeat steps (f), (h), (i), (k), and (l), using the copilot's control column. Results should be the same.

27.1.2.1 (Cont)

n. Fair the elevators, and check for vertical play in the control surfaces.
RESULT: There should be no more than 0.05 inch of free play in either elevator as measured at the inboard, trailing edge.

o. Depressurize the normal hydraulic system, disconnect the external hydraulic power from the normal hydraulic system external hydraulic connections, and connect it to the standby (No. 2) hydraulic system ground test connections.

p. Apply 3,000-PSI hydraulic pressure to the standby hydraulic system, and place the ELEVATOR SYS 1 switch in the "OFF" position and the ELEVATOR SYS 2 switch in the "ON" position.

q. Repeat steps (f), (h), (i), (k), and (l) using the standby hydraulic system. The results should be the same.

r. Place the ELEVATOR SYS 2 switch in the "OFF" position.

s. Perform a booster assembly leakage check according to the instructions in this section.

t. Depressurize the standby hydraulic system, and disconnect the external hydraulic power source.

u. Use the force measuring device to measure the amount of force required to move the control column through its full range of travel.
RESULT: The force required to obtain full elevator travel should not exceed 25 pounds.

v. Disconnect the 28-volt, DC power source from the airplane; remove all contour boards, protractors, and force measuring devices; return the elevator control system to normal.

27.1.2.2 ELEVATOR BOOSTER ASSEMBLY LEAKAGE CHECK.

a. Obtain two sources of external hydraulic power and connect one to the normal hydraulic system ground test connections and one to the standby hydraulic system ground test connections. Adjust both hydraulic power sources to deliver 2,700 PSI at 10 gallons per minute.

b. Connect 28-volt, DC power to the airplane service connections, and energize the main 28-volt, DC bus.

c. Make sure the No. 1 and No. 2 boost control circuit breakers are engaged, and that the ELEVATOR SYS 1 and ELEVATOR SYS 2 BOOST HYD PRESS switches on the pilot's control column cover are in the "ON" (guarded) position.

d. With hydraulic power applied to both portions of the booster assembly, cycle the elevator booster 40 complete cycles, closely observing leakage from the seals during the last 25 cycles.
RESULT: During the last 25 cycles, leakage should not exceed one drop from either of the rod end seals, and it should not exceed two drops from the center drain of the actuator. There should be no leakage from the static seals.

e. Upon completion of this check, reduce hydraulic pressure to zero, disconnect the power sources, and return the elevator control system to normal.

27.1.3 TROUBLESHOOTING.

27.1.3.1 ELEVATOR CONTROL SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INSUFFICIENT CLEARANCE BETWEEN CONTROL COLUMN HEAD ASSEMBLY AND THE FACE OF THE MAIN INSTRUMENT PANEL.		
Control system improperly rigged.	Perform an operational checkout of the system.	Rig the elevator control system as necessary.
ELEVATORS FAIL TO FAIR WITHIN PRESCRIBED LIMITS.		
Pushrod system improperly adjusted.	Perform an operational checkout of the system.	Rig the control system as necessary.
ELEVATORS FAIL TO TRAVEL THE PRESCRIBED AMOUNT.		
Control system improperly rigged.	Perform an operational checkout of the system.	Rig the control system as necessary.
Booster assembly defective.	Check the operation of the booster assembly.	Replace the defective booster assembly, and then rig the system.
No hydraulic boosting action.	Check the shutoff and bypass valve solenoids and control circuits.	Repair or replace solenoids and circuits as necessary.
	Check the operation of the boost shutoff and bypass valves.	Replace defective booster assembly.
Binding, excessive friction, or improper cable tension.	Check the tension indicator on the tension regulator. If tension is correct, look for excessive friction or binding.	Correct binding and/or excessive friction condition(s), and then rig the system as necessary.
EXCESSIVE UP-AND-DOWN PLAY IN THE ELEVATOR OUTPUT SYSTEM.		
Worn pushrods or bellcrank.	Check the pushrods and the bellcrank for excessive wear.	Replace excessively worn components, and then rig the system.
EXCESSIVE AMOUNT OF FORCE REQUIRED TO CONTROL THE ELEVATORS AT HIGH ALTITUDE OR IN COLD WEATHER.		
Water or ice trapped under the elevator booster assembly.	Check for water or ice under the elevator booster assembly and around the centering spring.	Clean and dry the area thoroughly, and make sure that drain holes are open.
EXCESSIVE BOOSTER LEAKAGE.		
Defective seals in booster assembly.	Perform a leakage check according to the instructions in this section.	Replace excessively leaking booster assemblies.

27.1.4 REMOVAL AND REPLACEMENT.

27.1.4.1 REMOVAL OF THE ELEVATOR BOOSTER ASSEMBLY.

WARNING

Depressurize the normal and standby hydraulic systems, and de-energize the auxiliary hydraulic pump circuits. Warn personnel not to move the controls.

a. If installed, remove the drag chute according to the instructions in Chapter 39 and then disconnect all wiring and cables from the tail cone.

b. Remove the tail cone to gain access to the booster assembly.

c. Disconnect the hydraulic lines from the booster assembly. Cap or plug all openings.

d. Disconnect the electrical connectors from the booster assembly and secure them to avoid damage.

e. Disconnect the elevator pushrods from the output arm of the booster assembly.

f. Disconnect the rod connecting the tension regulator input lever to the elevator feel lever at the input lever. Do not disturb cable rigging.

g. Remove horizontal diaphram from structure.

h. Support the booster assembly and remove the two internal wrenching bolts holding the front of the booster assembly to the airplane structure.

i. Remove the two upper bolts, nuts, and washers holding the rear of the booster assembly to the airplane structure, and remove the booster assembly from the airplane. Leave the tension regulator in the airplane.

CAUTION

Any time a booster assembly is removed or a control surface is disconnected from a hydraulic booster assembly or the hydraulic systems are drained of fluid, an external contour-type clamp shall be installed to lock the control surface to the adjacent structure. This will prevent damage if the airplane is subjected to wind gusts. As long as the hydraulic systems are properly serviced and the control surfaces are connected to the booster assembly, normal ground gusts will not damage the controls.

27.1.4.2 REPLACEMENT OF THE ELEVATOR BOOSTER ASSEMBLY.

Note

Prior to installation, the boost package should be placed on a bench with the damper fill plugs up. These are the two small slotted pan head plugs lockwired together on the same level as the base of the solenoids. The plugs should be removed and the cavities below filled to overflowing with clean MIL-H-5606 hydraulic fluid while moving the pilot lever back and forth by hand. Reinstall plugs, seals (MS28775-007), and lockwire.

a. Reverse the removal procedures.

b. Bleed the elevator control system.

c. Check elevator control system operation, and rig the system as needed.

27.1.4.3 REMOVAL OF THE ELEVATOR BOOSTER TENSION REGULATOR.

a. Gain access to the booster assembly by removing the drag chute mechanism according to the instructions in Chapter 39.

b. Loosen the cables between the elevator torque tube and the tension regulator on the bottom of the booster assembly.

c. Disconnect the centering spring bracket from the torque tube input shaft.

d. Remove the pulley bracket assembly below the tension regulator to provide enough clearance for removing the tension regulator.

e. With no tension on the cables to the tension regulator, remove the two bolts holding the tension regulator shaft to the input torque arm, and lower the tension regulator.

f. Disconnect the cables from the tension regulator, and remove the tension regulator from the airplane.

27.1.4.4 REPLACEMENT OF THE ELEVATOR BOOSTER TENSION REGULATOR.

a. Inspect the upper and lower tension regulator support bearings. Replace if necessary.

b. Reverse the removal procedures.

c. Rig the elevator control system as needed.

d. Perform an operational checkout of the elevator control system.

27.1.4.4 (Cont).

e. Secure all connections. Make sure that all cables are riding correctly in their pulleys and that all guard pins are in place.

f. Replace the access panels.

27.1.4.5 REMOVAL OF THE ELEVATOR. Follow these instructions for removal of either half of the elevator.

WARNING

Refer to Section 51.9 for precautions to be observed when working around depleted uranium counterbalance weights.

a. Disconnect the elevator from the torque tube at the inboard end. On airplanes 5046 and subsequent and airplanes modified to service bulletin 329-128 disconnect the elevator from outboard hinge pin.

CAUTION

Support the elevator when removing the attaching bolts to prevent dropping the elevator.

b. Remove the elevator by pushing aft and outboard.

c. Disconnect the bonding jumpers from the horizontal stabilizer.

d. Remove the elevator from the airplane.

27.1.4.6 REPLACEMENT OF ELEVATOR.

a. Lubricate the hinge pins with a small amount of Specification MIL-G-23827 grease, and then reverse the removal procedures. Check the hinge points for the clearances shown in figure 27-4. If necessary, on airplanes not modified by Service Bulletin 329-116, remove the torque tube and add or remove washers at the inboard end of the torque tube to obtain the 0.030- to 0.090-inch clearance. On airplanes modified by Service Bulletin 329-116, install spacers as necessary between the inboard end of the elevator and the outboard end of the torque tube to obtain the 0.030- to 0.090-inch clearance. Torque the attaching bolts to 70 to 90 inch-pounds.

b. Rig the elevator control system as needed.

c. Perform an operational checkout of the system.

d. Secure all connections, and replace the access doors.

27.1.4.7 REMOVAL OF ELEVATOR TORQUE TUBE.

a. Remove the elevator as outlined in paragraph 27.1.4.5.

b. Remove access covers, trailing edge fairing, and lower vertical stabilizer fairing.

c. Disconnect the fuel jettison line flexible hose.

d. Disconnect the elevator pushrod from the horn fitting.

e. Remove the bearing retainer at the inboard end of the torque tube and remove the torque tube from the airplane.

27.1.4.7.1 REPLACEMENT OF ELEVATOR TORQUE TUBE BEARING.

a. Remove torque tube per paragraph 27.1.4.7.

b. Remove cotter pin through 1/2-inch holes near inboard end of torque tube. Loosen and remove nut using socket and extension from outboard end. Use care not to lose washers under nut.

c. Reassemble using new bearing (KSBG-8-18) and cotter pin (MS24665-302) with same bolt, bearing retainer, washer, and nut. Torque nut to 150 to 200 inch-pounds and align cotter pin hole with one of the 1/2-inch access holes. Do not exceed 200 inch-pounds torque. Install and safety new cotter pin.

27.1.4.8 REPLACEMENT OF ELEVATOR TORQUE TUBE.

a. Reverse the removal procedures. On airplanes not modified by Service Bulletin 329-116, install the elevator temporarily and check for clearance of 0.030 to 0.090 inch between the shoulder on the outboard hinge pin and the ball on the outboard elevator hinge bearing. (See figure 27-4.) If necessary, remove the torque tube and add or remove washers at the inboard end of the torque tube to obtain clearance. On airplanes modified by Service Bulletin 329-116 and airplanes 5046 and up, no adjustment is required at this point.

b. Torque the outboard attaching bolts and install shims as indicated in paragraph 27.1.4.6.a.

c. Secure all connections and replace the access covers and fairings.

27.1.4.9 REMOVAL AND REPLACEMENT OF ELEVATOR BOOSTER FILTERS. Before removing filter caps, thoroughly clean adjacent area with an approved cleaning solvent.

Note

Before removing the elevator filter elements, disconnect the pressure lines to the actuator, and allow all oil to drain from the filter cavities.

CAUTION

Apply steady pressure when removing filter elements. A sudden thrust of torque may snap the element stem. Use an Allen wrench to remove filter elements. Remove packing from filter elements.

a. Visually inspect filter elements for breaks or cracks, and discard elements which show signs of damage.

b. Clean elements using one of the following methods:

(1) Send to an FAA Approved Commercial Filter Cleaning facility.

(2) Clean filter element using SAE Specification ARP725 as a guide.

Note

The filter element is rated at 10 microns nominal, 25 microns absolute.

(3) Return filter element to manufacturer for cleaning,

Aircraft Porous Media, Inc.
6301 49th St. N.
Pinellas Park, Florida 33565
Phone: 813-522-3111

c. Reassemble filter elements in reverse order of disassembly.

Note

Discard all used packing, and backup rings and replace with new parts.

Place new packing MS28778-10 on filter elements. Install elements in manifold filter cavities, and tighten 60 to 75 inch-pounds torque, using an Allen wrench. Place new packing MS28775-128 and new backup rings MS27595-128 on filter caps. Install filter caps on manifold housing and tighten 300 to 348 inch-pounds torque, using a 1-7/8-inch wrench.

27.1.4.10 BOLT TORQUE REQUIREMENTS. Bolts used with castle nuts and cotter pins at push rod and bell crank ends should be torqued at least 5 but no more than 10 inch-pounds to a position where the cotter pin can be installed. When possible, such installations should be made with the nut down. Other torques should be as indicated in Chapter 51 unless otherwise specified.

27.1.5 ADJUSTMENT. (See paragraph 27.1.2 for tools and test equipment required.)

27.1.5.1 ELEVATOR RIGGING INSTRUCTIONS. (See figure 27-7).

27.1.6 INSPECTION.

27.1.6.1 ELEVATOR CONTROL CABLE CONNECTOR INSPECTION (AIRPLANES 5001 thru 5045). After each reassembly of the elevator lockclad control cable connectors, the following inspection should be made to determine if the cable pin is installed:

WARNING

When the retainer is safety-wired in position, the pin location is not visible. The safety-wire may sustain the rigging tension loads without the pin. Therefore, it is possible for the pin to be omitted and its absence remain undetected until the connector separated under loads imposed by flight operation.

a. Measure the distance that the male end protrudes out of the female end.

b. Compare the measurement with the dimensions shown in figure 27-5 to make certain the cable pin has been installed.

Note

The projection of the male end from the female will be approximately 0.31 inch greater without the pin than with the pin installed.

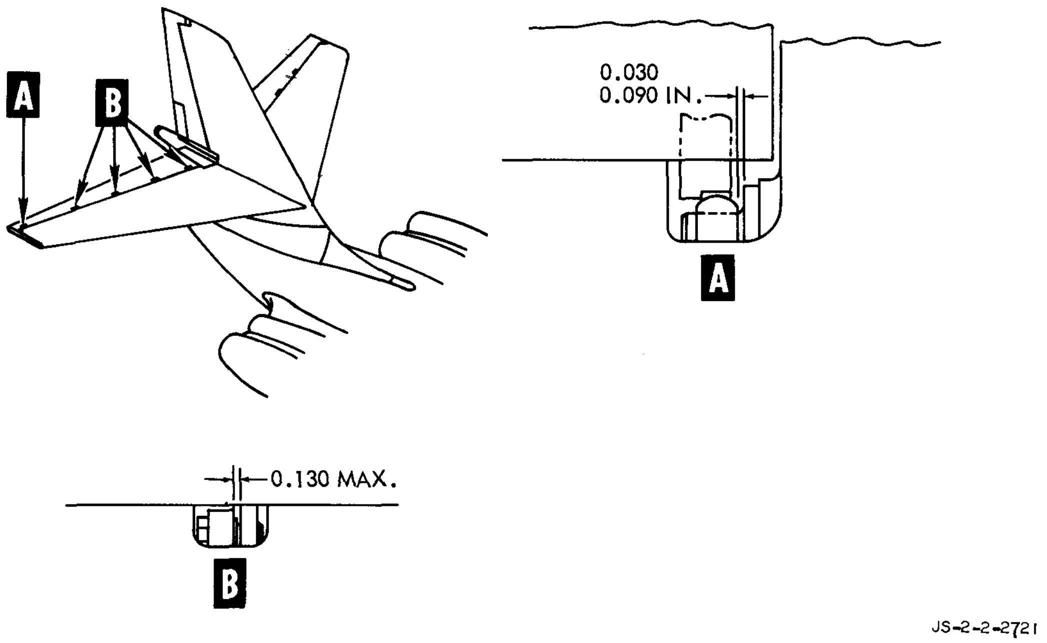


Figure 27-4. Elevator Hinge Pin Clearance

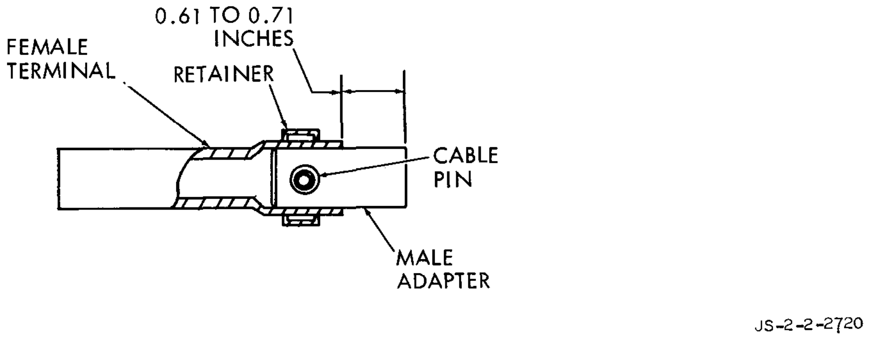
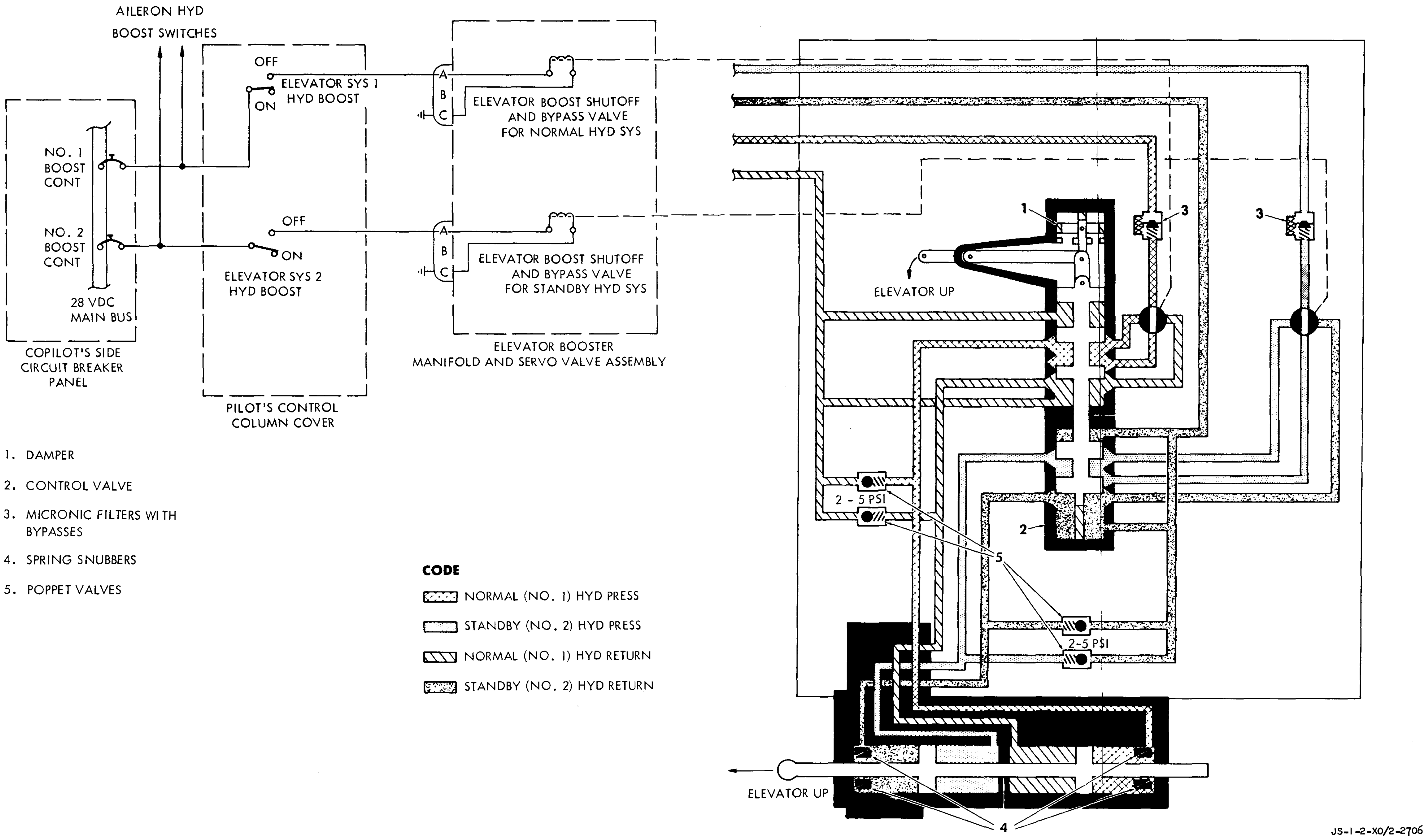


Figure 27-5. Elevator Lockclad Cable Connector Inspection



1. DAMPER
2. CONTROL VALVE
3. MICRONIC FILTERS WITH BYPASSES
4. SPRING SNUBBERS
5. POPPET VALVES

CODE

▨ NORMAL (NO. 1) HYD PRESS

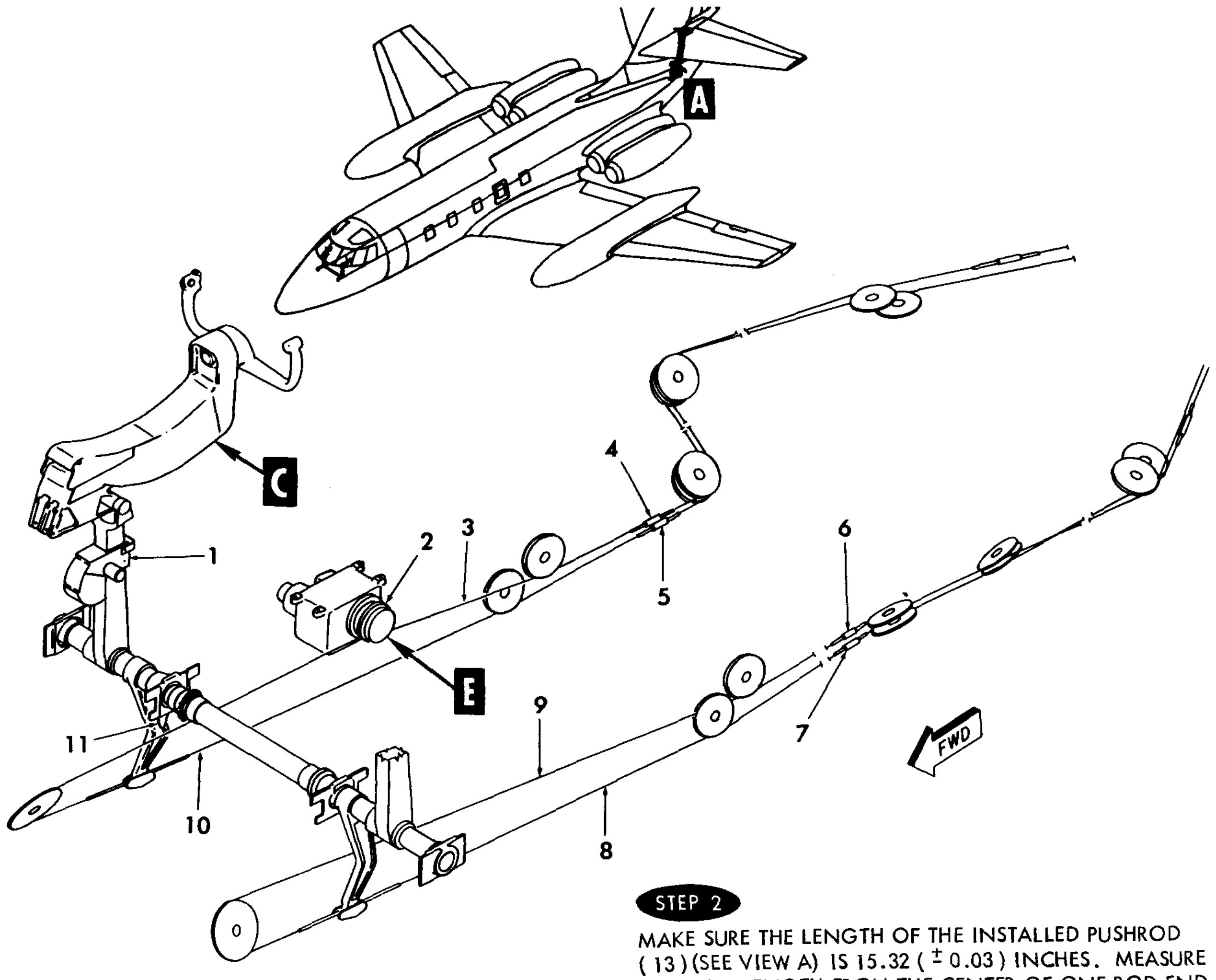
▭ STANDBY (NO. 2) HYD PRESS

▧ NORMAL (NO. 1) HYD RETURN

▩ STANDBY (NO. 2) HYD RETURN

Figure 27-6. Elevator Control System Schematic Diagram

Changed 31 August 1983



NOTE

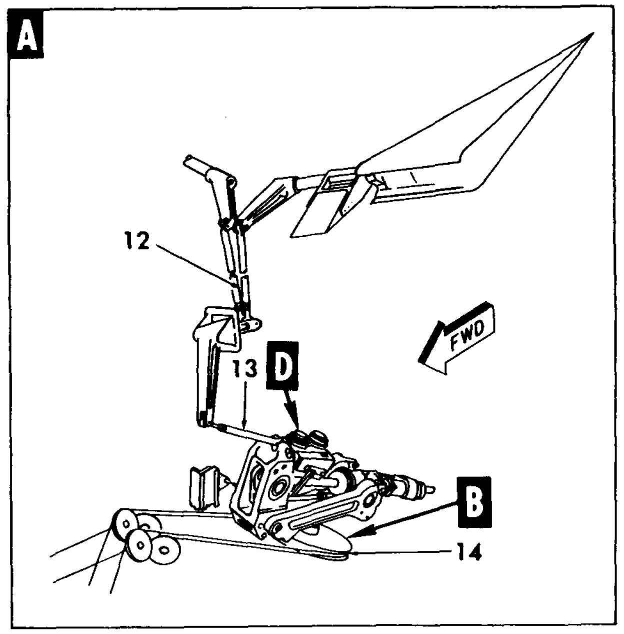
REMOVE THE FOLLOWING ACCESS PANELS AND FLOORING TO RIG THE ELEVATOR CONTROL SYSTEM: THE LEFT AND RIGHT ACCESS PANELS ON THE AFT, EMPENNAGE FAIRING UNDER THE RUDDER; THE LEFT AND RIGHT ACCESS PANELS JUST BELOW THESE AND ON THE FUSELAGE; THE COPILOT'S FLOORING AFT OF THE CONTROL COLUMN AND UNDER THE COPILOT'S SEAT; AND THE KICK PANEL ON THE LOWER LEFT SIDE OF THE COPILOT'S CONTROL COLUMN COVER. IT MAY ALSO BE NECESSARY TO REMOVE SOME OF THE ACCESS PANELS INSIDE THE ACCESS OPENING FORWARD OF THE NOSE GEAR DOORS.

STEP 1

SET THE HORIZONTAL STABILIZER AT ZERO PITCH TRIM BY ACTUATING THE PITCH TRIM ACTUATOR TO A LENGTH OF 28.37 (± 0.03) INCHES. MEASURE THE PITCH TRIM ACTUATOR LENGTH ACCORDING TO THE INSTRUCTIONS ON TIMING THE PITCH TRIM ACTUATOR HALVES.

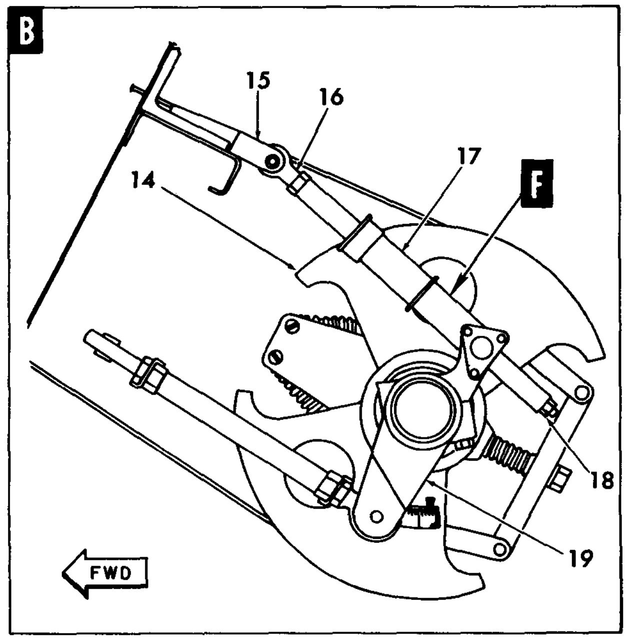
STEP 2

MAKE SURE THE LENGTH OF THE INSTALLED PUSHROD (13) (SEE VIEW A) IS 15.32 (± 0.03) INCHES. MEASURE PUSHROD LENGTH FROM THE CENTER OF ONE ROD END TO THE CENTER OF THE OTHER ROD END.



STEP 3

DISCONNECT THE ELEVATOR CENTERING SPRING (17) FROM THE BULKHEAD FITTING (15) AND FROM THE TENSION REGULATOR (14), AND REMOVE IT FROM THE AIRPLANE. (SEE VIEW B).



STEP 4

WITH BOTH CONTROL COLUMNS IN THE EXTREME FORWARD POSITION (SEE VIEW C), ADJUST THE DOWN-STOP (21) TO OBTAIN $2.70\,(^{+0.30}_{-0.10})$ INCHES CLEARANCE BETWEEN THE CONTROL COLUMNS AND THE MAIN INSTRUMENT PANEL. WITH THE COLUMNS IN THIS POSITION, ADJUST THE SHAKER ASSEMBLY (1) TO CLEAR THE PAN (22) BY AT LEAST .75 INCH.

STEP 5

WITH NO HYDRAULIC PRESSURE ON THE ELEVATOR BOOSTER, MOVE THE CONTROL COLUMNS TO DRIVE THE BOOSTER ASSEMBLY THROUGHOUT ITS FULL RANGE OF TRAVEL AND CHECK FOR INTERFERENCE AROUND THE BOOSTER ASSEMBLY AND THE CONTROL SYSTEM FROM THE BOOSTER ASSEMBLY TO THE ELEVATOR.

NOTE

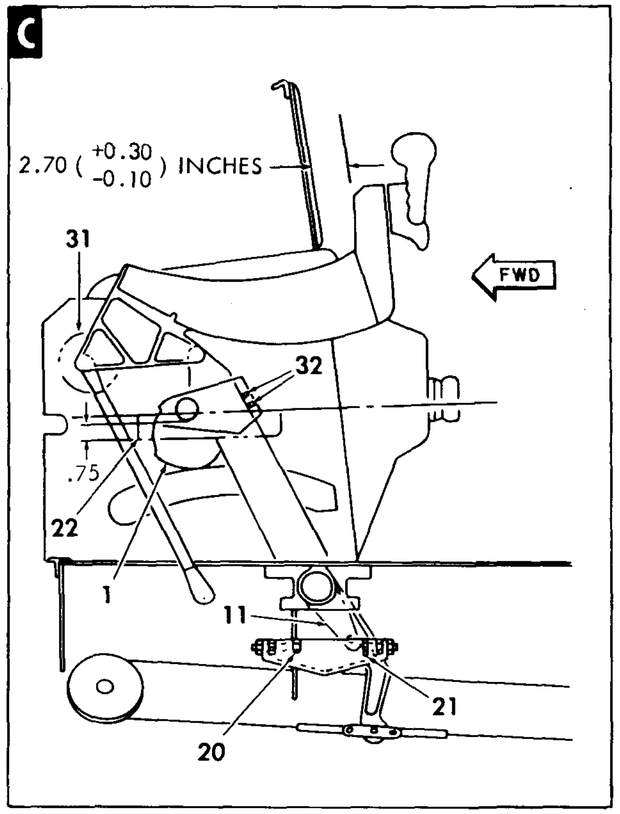
WITH HYDRAULIC PRESSURE ON THE ELEVATOR BOOSTER, THE ELEVATOR WILL TRAVEL APPROXIMATELY 1 DEGREE FARTHER. ALLOW FOR THIS EXTRA TRAVEL.

STEP 6

CONNECT EXTERNAL HYDRAULIC POWER TO BOTH THE NORMAL (NO. 1) AND THE STANDBY (NO. 2) HYDRAULIC GROUND TEST CONNECTIONS, AND PRESSURIZE BOTH SIDES OF THE ELEVATOR BOOSTER TO 2,700 TO 3,000 PSI. WITH THIS PRESSURE ON THE BOOSTER ASSEMBLY, MOVE THE CONTROL COLUMNS THROUGH SIX COMPLETE CYCLES AND MAKE SURE THERE IS NO INTERFERENCE THROUGHOUT THE FULL RANGE OF TRAVEL.

STEP 7

WITH THE STOP LEVER (11) 0.032 (± 0.010) INCH FROM THE DOWN-STOP (21), MAKE SURE THE BOOSTER PISTON IS BOTTOMED AND THE SERVO VALVE INPUT ARM (25) IS BOTTOMED AGAINST THE EXTERNAL STOP (24). (SEE VIEW "D"). WITH THE BOOSTER HELD IN THIS POSITION, AND WITH CABLES (3) AND (10) SLACK, ALTERNATELY ADJUST TURNBUCKLES (6) AND (7) IN CABLES (8) AND (9) TO SET THE CABLE TENSION REGULATOR (14) ACCORDING TO THE TEMPERATURE-TENSION CONVERSION CHART.



STEP 8

WITH THE CONTROL COLUMNS IN THE EXTREME AFT POSITION, THE BOOSTER PISTON BOTTOMED, AND THE SERVO VALVE INPUT ARM (25) AGAINST THE EXTERNAL STOP (23); ADJUST THE UP-STOP (20) TO WITHIN 0.164 (± 0.030) INCH OF THE STOP LEVER (11). ADJUST THE VERTICAL PUSHRODS (12) (SEE VIEW "A") TO OBTAIN AN ELEVATOR-UP ANGLE OF 13 (± 1/4) DEGREES. RETURN THE ELEVATOR TO THE FAIRED POSITION AND PLACE AN ELEVATOR CONTOUR BOARD FROM THE CONTROL SURFACES RIGGING KIT ON THE LEFT SIDE OF THE ELEVATOR. WITH THE LEFT SIDE OF THE ELEVATOR IN THIS POSITION, THE RIGHT SIDE OF THE ELEVATOR SHOULD BE WITHIN 0.06 INCH OF THE FAIRED POSITION. MAKE SURE THE SHAKER ASSEMBLY (1) IS NO LESS THAN .45 INCH FROM THE PULLEY (31) WHEN THE COPILOT'S CONTROL COLUMN IS IN THE EXTREME AFT POSITION WITH THE STOP LEVER (11) AGAINST THE UP-STOP (20) (SEE VIEW "C").

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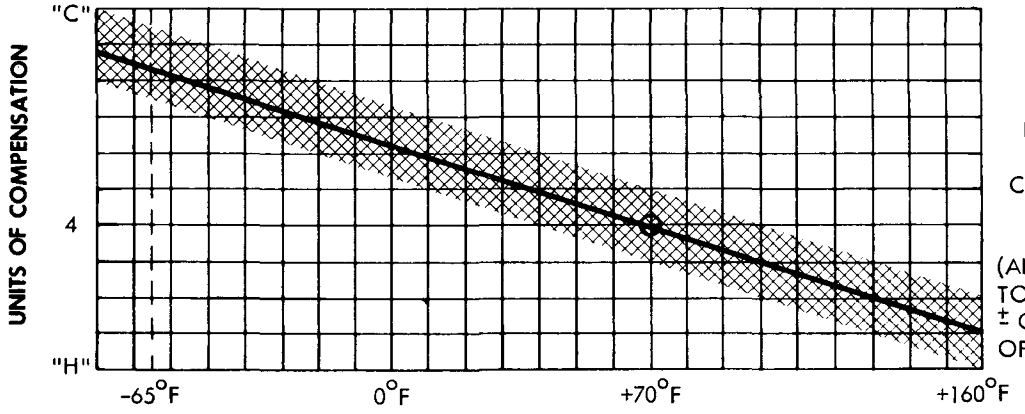
Figure 27-7. Elevator Control System Rigging Instructions (Sheet 1 of 2)



CABLE TENSION REGULATOR RIGGING CHART FOR ELEVATOR CONTROL SYSTEM

(ALLOWABLE TOLERANCE IS ± ONE UNIT OF COMPENSATION)

NOTE

THIS CHART IS VALID ONLY WHEN THE TEMPERATURES OF THE CABLES AND THE STRUCTURE HAVE STABILIZED.

NOTE

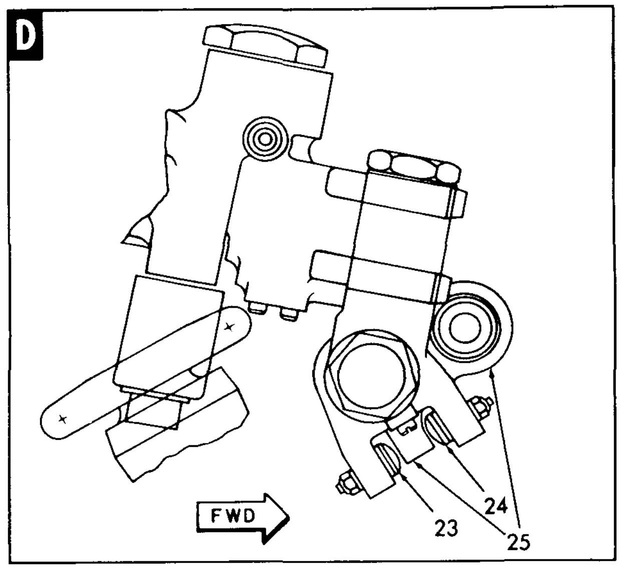
TO OBTAIN SHAKER ASSEMBLY CLEARANCE IT MAY BE NECESSARY TO READJUST THE EXTREME FORWARD POSITION AND/OR REPOSITION THE SHAKER ASSEMBLY.

STEP 9

WITH THE BOOSTER ASSEMBLY STILL PRESSURIZED, MOVE THE CONTROL COLUMNS TO THE EXTREME FORWARD POSITION AND MAKE SURE THE SERVO VALVE INPUT ARM (25) IS AGAINST THE EXTERNAL STOP (24). (SEE VIEW D). WITH THE BOOSTER ASSEMBLY IN THIS POSITION, THE ELEVATOR SHOULD BE 16-3/4 (± 3/4) DEGREES DOWN FROM THE FAIRED POSITION.

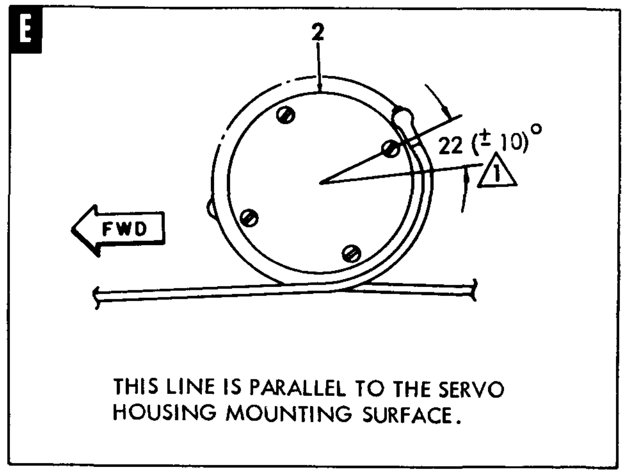
NOTE

MAKE SURE THE SHAKER ASSEMBLY (1) IS NO CLOSER THAN .75 INCH TO THE PAN (22); THEN MAKE SURE THE SHAKER ADJUSTMENT SCREWS (32) ARE SAFETIED.



STEP 10

WITH THE CONTROL COLUMN FORWARD AND THE BOOSTER PISTON BOTTOMED, MAKE SURE THE ELEVATOR AUTOPILOT SERVO IS POSITIONED ACCORDING TO VIEW E WITH CABLES WRAPPED AROUND THE DRUM (2) AS SHOWN.



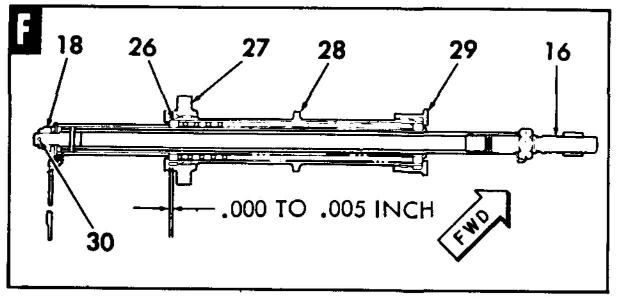
THIS LINE IS PARALLEL TO THE SERVO HOUSING MOUNTING SURFACE.

STEP 11

ATTACH TENSIOMETERS TO CABLES (3), (8), (9), AND (10); THEN ADJUST TURNBUCKLES (4), (5), (6), AND (7) TO OBTAIN CABLE TENSIONS WITHIN 5 POUNDS WHILE MAINTAINING THE PROPER TENSION ON THE TENSION REGULATOR.

STEP 12

REDUCE HYDRAULIC PRESSURE TO ZERO



STEP 13

RECONNECT THE SPRING ASSEMBLY (17) (SEE VIEW B) TO THE BOOSTER ASSEMBLY; THEN APPLY 3,000-PSI HYDRAULIC PRESSURE TO BOTH SIDES OF THE ELEVATOR BOOSTER, AND ACTUATE THE ELEVATOR TO THE FAIRED POSITION. WITH THE ELEVATOR IN THIS POSITION, ADJUST THE ROD END (16) TO ATTACH THE SPRING ASSEMBLY (17) TO THE BRACKET (15) WITHOUT PUTTING ANY LOAD ON THE CENTERING SPRING ASSEMBLY.

STEP 14

REDUCE HYDRAULIC PRESSURE TO ZERO; THEN DISCONNECT THE HYDRAULIC POWER SOURCE FROM THE AIRPLANE.

STEP 15

INSPECT ALL WITNESS HOLES TO SEE THAT THREADED CONNECTORS ARE SUFFICIENTLY ENGAGED, PERFORM AN OPERATIONAL CHECKOUT OF THE ELEVATOR CONTROL SYSTEM, THEN SECURE AND SAFETY ALL TURNBUCKLES AND CONNECTIONS AS REQUIRED AND REPLACE ALL ACCESS PANELS AND FLOORING.

JS-2-2-2724-2

Figure 27-7. Elevator Control System Rigging Instructions (Sheet 2 of 2)

SECTION 27.2

RUDDER CONTROL SYSTEM

Contents Page

DESCRIPTION AND OPERATION 27-27
General 27-27
Operation 27-27
Rudder Pedals 27-27
Rudder Pedal Adjustment 27-27
Torque Tube Assembly 27-28
Autopilot Servo Drive 27-28
Gust Lock Mechanism 27-28

OPERATIONAL CHECKOUT 27-28
Rudder Control System 27-28
Rudder Gust Lock Mechanism 27-29

TROUBLESHOOTING 27-29
Rudder Control System 27-29
Gust Lock Mechanism 27-30

REMOVAL AND REPLACEMENT 27-30
Removal of Rudder 27-30
Replacement of Rudder 27-31

ADJUSTMENT 27-31
Rudder Control System Rigging Instructions 27-31
Rudder Pedal Legroom-Adjustment Mechanism 27-31
Gust Lock Mechanism 27-31

FIGURES
27-8 Gust Lock Assembly 27-33
27-9 Rudder Control System Rigging Instructions 27-34

27.2.1 DESCRIPTION AND OPERATION.

27.2.1.1 GENERAL. Directional control of JetStar airplanes is accomplished through use of a cable actuated rudder with a balance tab mechanism. Rudder pedals connect to a rudder torque tube assembly under the cockpit floor by pushrods and levers. A lever on the torque tube assembly operates a cable system to the rudder control surface when the torque tube assembly is rotated. See Section 51.5 for instructions on rebalancing rudder and Section 59.8 for typical repairs to the rudder and rudder control items.

27.2.1.2 OPERATION. A mechanical linkage transmits movement of the pilot's and copilot's rudder pedals to a rudder torque tube below the cockpit floor. Both sets of rudder pedals are adjustable for pilot comfort. A bellcrank on the rudder torque tube is connected to a single cable system which is routed to the rudder torque arms on the vertical torque tube. Pilot effort to control the rudder is reduced by the balance tab action of the rudder trim tab. The tab actuator is attached to the vertical stabilizer, and the tab is attached to the rudder. Linkage between the actuator and the tab swings the tab during rudder movement to provide an aerodynamic boosting action during flight.

27.2.1.3 RUDDER PEDALS. There are two sets of rudder pedals in JetStar airplanes: one set for the pilot, and one set for the copilot. Each set consists of two pedals or levers. Depressing the left pedal positions the rudder for a left turn; depressing the right pedal positions the rudder for a right turn. A brake pedal is hinged to the top of each rudder pedal. Pivoting the brake pedals operates the brakes. The distance from the pilots' seats to the pedals is adjustable; the control for each pedal adjustment is on the respective control column cover.

27.2.1.4 RUDDER PEDAL ADJUSTMENT. A RUDDER PEDAL ADJUST knob on the aft side of each control column cover adjusts the distance from the pedals to the seats. Each knob has an indicator showing a neutral position and several reference

27.2.1.4 (Cont).

points. Either set of rudder pedals can be adjusted independently of the other set. Rotating the adjustment knob rotates a flexible shaft which connects to a worm gear near the base of the pedals under the flooring. Rotating this worm gear positions a sector to change the relative position of the pedals with respect to the position of the torque tube assembly.

27.2.1.5 TORQUE TUBE ASSEMBLY. The rudder torque tube assembly under the cockpit floor is made up of three sections: a left section, a center section, and a right section. The left section receives pilot commands and transfers them to the right section by means of the interconnecting center section. The right section also positions a lever to actuate the cable system which connects to the torque arms on the rudder torque tube. A sector on the right section receives autopilot servomotor commands. The center section of the torque tube assembly contains a cam which is engaged by a rudder gust lock latching mechanism to lock the rudder control surface when the RUDDER LOCK handle is pulled. Where the rudder torque tube sections meet, sprockets butt together and chain assemblies cover them to form unions.

27.2.1.6 AUTOPILOT SERVO DRIVE. A sector on the right section of the rudder torque tube assembly receives autopilot commands from a rudder servomotor. This servomotor receives autopilot signals and drives a servomotor drum to position cables which connect to the sector on the rudder torque tube assembly. This servomotor drum is driven by the servomotor through a clutch mechanism. This clutch

27.2.1.6 (Cont).

mechanism disengages the drum from the servomotor when the servomotor is not operating.

27.2.1.7 GUST LOCK MECHANISM. (See figure 27-8.) A RUDDER LOCK tee handle on the right side of the throttle quadrant controls both a rudder lock and a throttle lock. The rudder lock holds the rudder control surface in the faired position to prevent damage due to sudden external forces upon the rudder surface. The throttle lock mechanism restricts throttle movement to prevent flying the airplane with the rudder surface locked. Neither of these locks can be engaged unless the rudder is near the faired position and the throttles are near the "IDLE" position. Pulling the RUDDER LOCK tee handle actuates a latching mechanism to engage a cam on the center section of the rudder torque tube assembly under the pilots' floor. This latching mechanism engages with two slots in the cam to lock the rudder torque tube and thus the rudder surface in the faired position. At the same time that the rudder is locked, two throttle-lock levers are raised between the throttle pulleys for engines No. 1 and No. 2, and No. 3 and No. 4. These stop levers contact stop strips attached to the throttle pulleys when the throttles are advanced approximately 1-1/4 inch forward of the "IDLE" position. These lock levers cannot be raised to restrict throttle movement unless all throttles are retarded below this stop-contact position. When the RUDDER LOCK tee handle is released and returned to the normal (in) position, mechanical linkage opens the latching mechanism from around the cam on the torque tube and returns the throttle-lock levers to the lowered or unlocked position.

27.2.2 OPERATIONAL CHECKOUT.

Tools Required

Nomenclature	Description	Use and Application
Controls Surface Rigging Kit	Lockheed Aircraft Co. 11019-MEU	To determine the faired position of the rudder.
Rudder and Trim Travel Kit	Lockheed Aircraft Co. 110668-TG	To measure rudder travel.

27.2.2.1 RUDDER CONTROL SYSTEM.

a. Engage the rudder gust lock, then use a rudder contour template to make sure the rudder surface is faired to the vertical stabilizer within 0.06 inch.

b. With the rudder in the position of step (a), actuate the trim tab to fair with the rudder surface within 0.03 inch as measured from the top of the trailing edge of the tab to the rudder surface, and attach a rudder and trim travel gage to the airplane.

27.2.2.1 (Cont).

c. Make sure the rudder control cables are rigged to the proper tension as recommended in the rudder rigging instructions.

d. Disengage the rudder gust lock.

e. Using a spring scale, measure the amount of force required to obtain full left rudder deflection. Measure also the trim tab deflection and the pedal travel.

27.2.2.1 (Cont).

RESULT: The rudder should deflect 30(±0.5) degrees to the left; the rudder trim tab should deflect 3.15(±0.20) inches to the right of the neutral rig position at full left rudder deflection; the amount of force required to start the pedal moving should not exceed 10 pounds, and the amount of force required to obtain full rudder deflection should not exceed 16 pounds; the left rudder pedal should travel 3.25(±0.12) inches forward from the neutral position, and the right pedal should travel the same amount in the opposite direction.

f. Repeat step (e) for full right rudder. Results should be the same though for opposite directions.

g. Return the rudder to the faired position, then adjust the pilot's pedal adjustment knob to obtain full forward pedal adjustment.

h. With the pedals adjusted fully forward, check pedal travel by commanding full left then full right rudder.
RESULT: The pedals should travel 3.25(±0.12) inches in both directions with no interference.

i. Adjust the pilot's pedals to the extreme aft adjustment position measuring the amount of adjustment available.
RESULTS: There should be approximately 5.0 inches of pedal adjustment available from fully forward to fully aft as measured on a line parallel to the airplane floor.

j. With the pedals adjusted to the fully aft position, check the rudder pedal travel for both full left rudder and full right rudder.
RESULT: The pedals should travel 3.25(±0.12) inches in both directions with no interference.

k. Repeat steps (g) through (j) for the copilot's pedals. Results should be the same.

27.2.2.1 (Cont).

l. Remove all test equipment and tools from the rudder and tab, and return the rudder control system to normal.

27.2.2.2 RUDDER GUST LOCK MECHANISM.

a. With the rudder pedals aligned and in the neutral adjustment position, and with the throttles in the "OFF" position, use a force measuring device to pull the RUDDER LOCK tee handle out approximately 3 inches. Rotate the handle approximately 70 degrees clockwise and lock it in the extended position.
RESULT: The force required to pull the handle should not exceed 40 pounds. The throttles should be restricted to 1-1/4-inch movement forward of the "IDLE" position. Measure this distance from the forward edge of the throttle levers at the point where the throttles protrude from the quadrant. The rudder should lock in the faired position.

Note

It may be necessary to fair the rudder before the gust lock can engage.

b. Unlock the gust lock by rotating the RUDDER LOCK HANDLE approximately 70 degrees clockwise and then releasing it.
RESULT: The handle should return to the normal (in) position with no assistance, and the rudder and throttles should be free to move.

c. Pull the RUDDER LOCK handle out approximately 0.4 inch on airplanes 5041 & up (0.16 inch on airplanes 5001 thru 5040), and try to operate the throttles and rudder.
RESULT: Both the throttles and the rudder controls should operate with no interference.

d. Remove all test equipment and tools.

27.2.3 TROUBLESHOOTING.

27.2.3.1 RUDDER CONTROL SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
EXCESSIVE PEDAL FORCE REQUIRED TO CONTROL RUDDER.		
Binding or interference in the rudder and rudder trim systems.	Check the rudder and the rudder trim systems for binding and interference.	Relieve interference and binding, and then rig the systems as required.
Improper cable tension.	Check the cable tension.	Adjust the cable tension.

27.2.3.1 RUDDER CONTROL SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INSUFFICIENT PEDAL ADJUSTMENT AVAILABLE.		
Adjustment mechanism improperly adjusted.	Perform an operational checkout of the rudder control system to determine if the rest of the control system is properly rigged.	Adjust the mechanism, and rig the control system as necessary.
INCORRECT RUDDER AND TAB TRAVEL.		
Systems improperly rigged.	Perform an operational checkout of the rudder control system.	Rig the rudder system as required.
Pushrods or bellcranks worn excessively.	Check the pushrods and bellcranks for excessive wear.	Replace excessively worn components and then rig the system as needed.
Control system stops improperly adjusted.	Check the adjustment of the mechanical stops both on the torque tube assembly under the cockpit floor and on the vertical rudder torque tube.	Adjust the stops according to the instructions in the rigging instructions.
Binding or interference in the rudder or rudder tab system.	Check the tab linkages and all rudder control cables and linkages for interference and binding.	Relieve binding and interference as necessary, and then check the rudder rigging.

27.2.3.2 GUST LOCK MECHANISM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
RUDDER LOCK HANDLE FAILS TO RETURN TO THE UNLOCKED (IN) POSITION UNASSISTED WHEN UNLOCKED AND RELEASED.		
Binding, excessive friction, or broken springs in gust lock mechanism.	Perform an operational checkout of the gust lock mechanism. If the amount of force required to pull the handle is not excessive, check for broken springs.	Relieve binding and excessive friction as necessary. Replace broken springs.
THROTTLES AND RUDDER DO NOT MOVE FREELY WHEN THE RUDDER LOCK HANDLE IS PULLED OUT LESS THAN 0.16 INCH ON AIRPLANES 5001-5040 AND 0.4 INCH ON AIRPLANES 5041 & UP.		
Gust lock mechanism improperly adjusted.	Perform an operational checkout of the gust lock mechanism.	Adjust the gust lock mechanism as needed.

27.2.4 REMOVAL AND REPLACEMENT.

27.2.4.1 REMOVAL OF THE RUDDER.

WARNING

Refer to Section 51.9 for precautions to be observed when working around depleted uranium counterbalance weights.

27.2.4 (Cont).

a. Disconnect all bonding straps from the rudder.

b. Disconnect the rudder trim pushrods from the bellcrank near the vertical stabilizer.

c. Disconnect the rudder from the rudder torque tube.

d. Lift the rudder up and off the airplane.

27.2.4.2 REPLACEMENT OF THE RUDDER.

a. Lubricate the hinge pins with a small amount of Specification MIL-G-23827 grease, and then reverse the removal procedures.

27.2.4.2 (Cont).

b. Perform an operational checkout of the rudder and rudder trim systems, and rig the systems as needed.

c. Secure all connections, and replace the access doors.

27.2.5 ADJUSTMENT.

Tools Required

Nomenclature	Description	Use and Application
Control Surfaces Rigging Kit	Lockheed Aircraft Co. 11019-MEU	To determine the faired position of the rudder.
Rudder and Trim Travel Gage	Lockheed Aircraft Co. 11068-TG	To determine rudder travel.
Type T-5 Cable Tensiometer.	Pacific Scientific Co. 45402	To measure cable tension.

27.2.5.1 RUDDER CONTROL SYSTEM RIGGING INSTRUCTIONS.

(Refer to figure 27-9.)

27.2.5.2 RUDDER PEDAL LEGROOM-ADJUSTMENT MECHANISM.

a. Using the RUDDER PEDAL ADJUST knobs on the control column covers, adjust the pedals to a position midway between adjustment extremes.

b. With the pedals in the position of step (a), disconnect the flexible cables from the knob assemblies, and rotate the knobs until the indicators on the knob assemblies show the neutral-adjustment position.

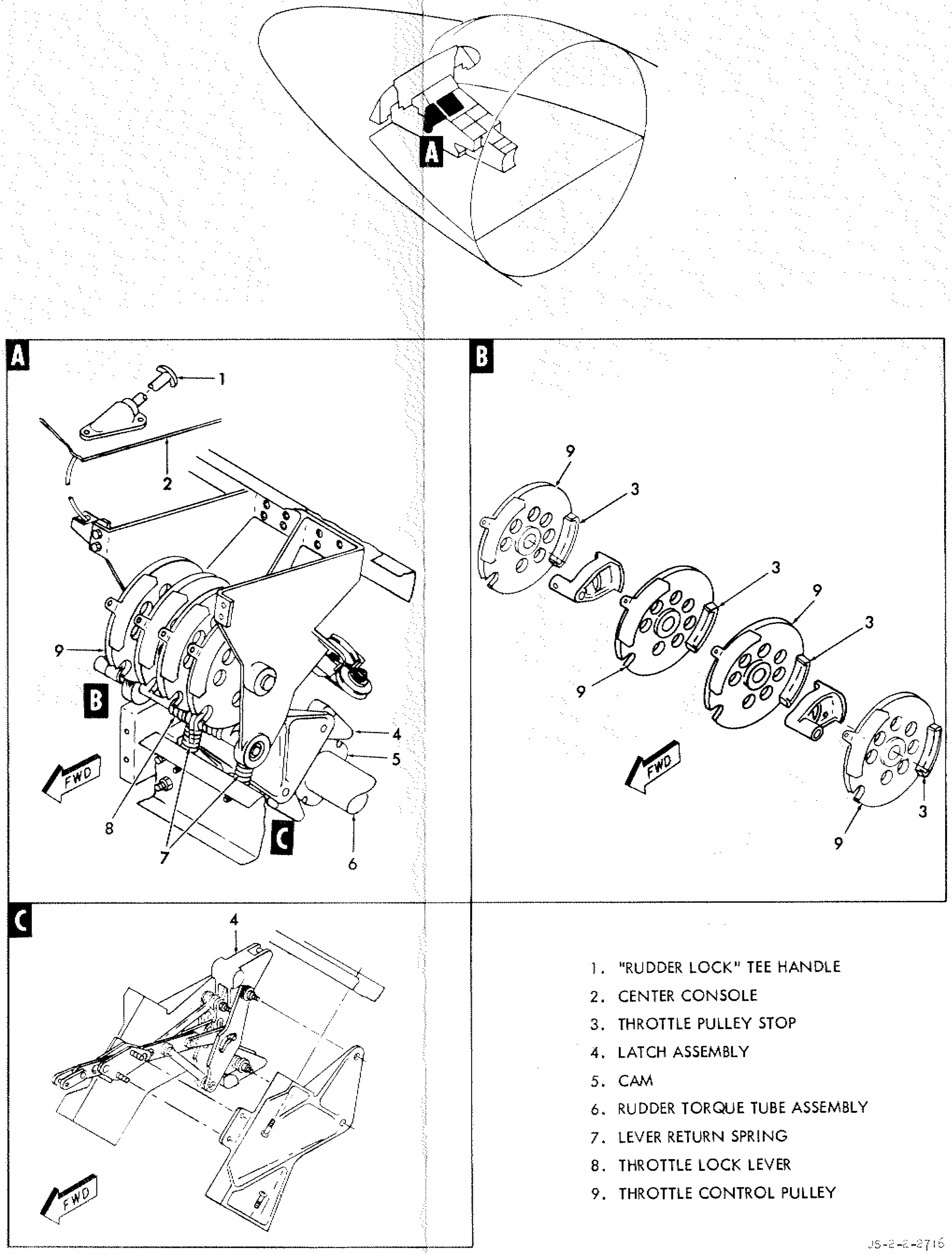
c. Reconnect the flexible cables to the knob assemblies, and make sure at least 2.9 inches of pedal travel is available in the forward direction and 1.0 inch less in the aft direction, measured on a line parallel with the airplane floor.

d. Return the pedal adjustment mechanism to normal after completing this procedure.

27.2.5.3 GUST LOCK MECHANISM ADJUSTMENT.

a. Adjust the gust lock mechanism to lock both the throttles and the rudder controls when the RUDDER LOCK handle is pulled out and locked in the extended position.

b. Adjust the gust lock mechanism to fully unlock the throttles and rudder controls when the RUDDER LOCK handle is 0.4 inch from the fully-in position on airplanes 5041 and up, and 0.16 inch from the fully-in position on airplanes 5001 through 5040.



1. "RUDDER LOCK" TEE HANDLE
2. CENTER CONSOLE
3. THROTTLE PULLEY STOP
4. LATCH ASSEMBLY
5. CAM
6. RUDDER TORQUE TUBE ASSEMBLY
7. LEVER RETURN SPRING
8. THROTTLE LOCK LEVER
9. THROTTLE CONTROL PULLEY

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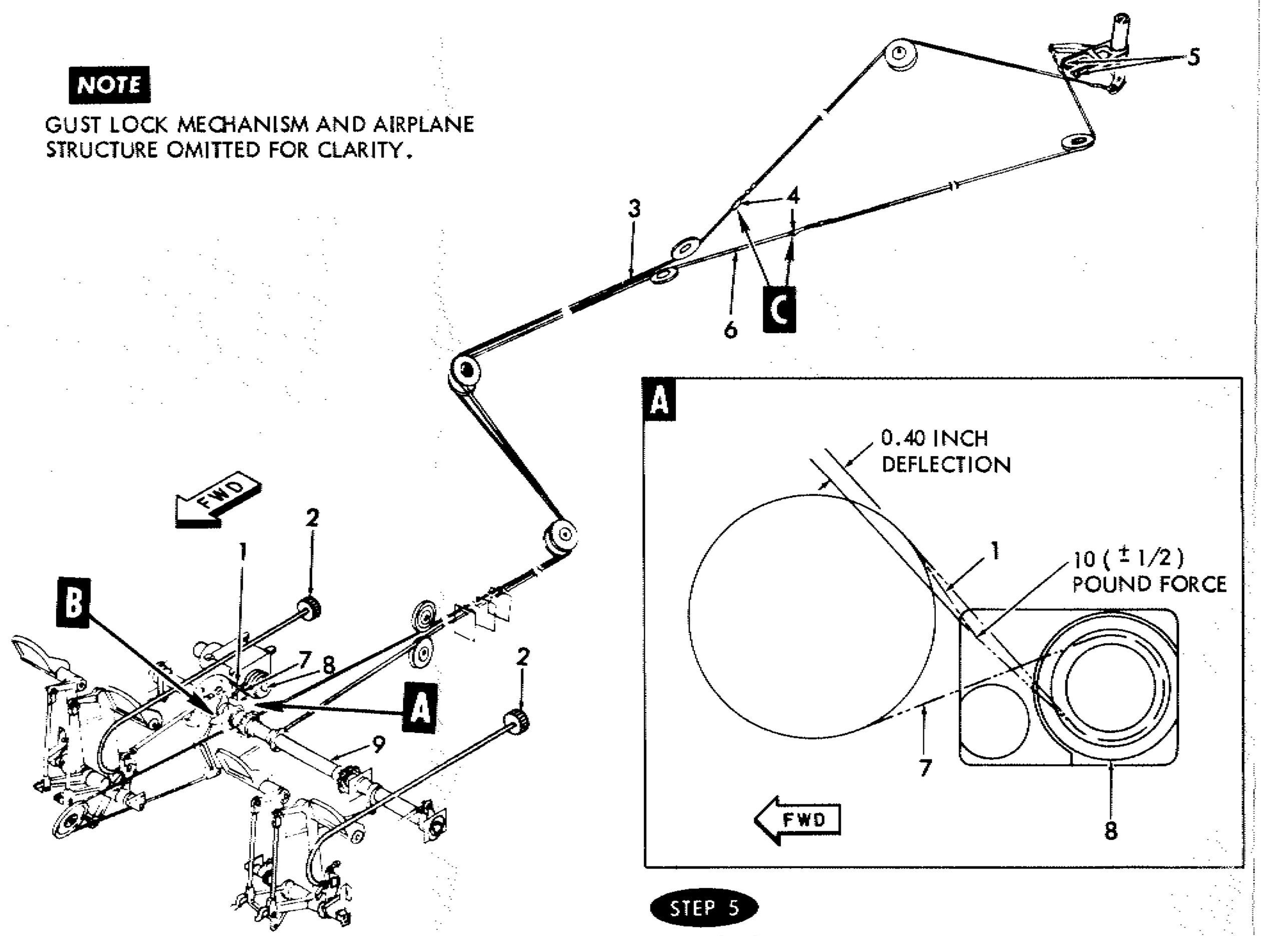
Figure 27-8. Gust Lock Assembly

NOTE

GUST LOCK MECHANISM AND AIRPLANE STRUCTURE OMITTED FOR CLARITY.



STEP 1

DISCONNECT CABLES (3) AND (6) AT THE QUICK DISCONNECTS (4) AT APPROXIMATE FUSELAGE STATION 682. (AIRPLANES 5045 AND UP DO NOT HAVE A QUICK DISCONNECT BUT HAVE TURNBARRELS AT THIS POINT.)

STEP 2

LOCK THE TORQUE SHAFT (9) IN THE NEUTRAL POSITION BY ENGAGING THE RUDDER GUST LOCK.

STEP 3

ADJUST BOTH THE PILOT'S AND THE COPILOT'S PEDALS TO THE NEUTRAL POSITION BY USING THE RUDDER PEDAL ADJUST KNOBS (2).

STEP 4

ADJUST CABLES (1) AND (7) TO OBTAIN A TENSION EQUIVALENT TO 50 (± 10) POUNDS AT 70 DEGREES FAHRENHEIT. MAKE SURE THAT CABLE-WRAP AROUND THE SERVO DRUM (8) IS EQUAL TO WITHIN 1/4 INCH. TO DETERMINE CABLE TENSION, APPLY A 10 (± 1/2) POUND FORCE AS SHOWN IN VIEW "A". WHEN THE CABLE DEFLECTION IS 0.40 (± 0.030) INCH, THE CABLE TENSION IS 50 (± 10) POUNDS.

STEP 5

DISENGAGE THE RUDDER GUST LOCK AND MAKE SURE THAT 3.8 INCHES OF PEDAL TRAVEL IS AVAILABLE IN BOTH DIRECTIONS WITH NO INTERFERENCE. CHECK RUDDER PEDAL TRAVEL IN BOTH EXTREMES OF PEDAL ADJUSTMENT.

STEP 6

BACK OFF THE MECHANICAL STOPS (5). DISCONNECT THE RUDDERTRIM ACTUATOR FROM THE TRIM SURFACE, AND MAKE SURE THAT ATLEAST 31 DEGREES OF RUDDER TRAVEL IS AVAILABLE IN BOTH DIRECTIONS WITH NO INTERFERENCE.

STEP 7

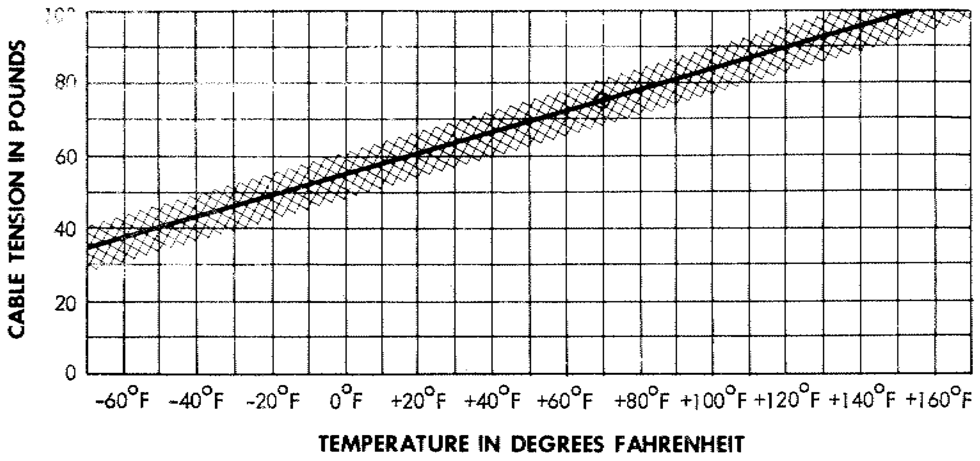
RECONNECT CABLES (3) AND (6).(NOTE: CAUTION SHOULD BE OBSERVED WHEN RECONNECTING QUICK DISCONNECTS. SEE DETAIL C.)

STEP 8

ENGAGE THE RUDDER GUST LOCK, THEN ALTERNATELY ADJUST CABLES (3) AND (6) TO OBTAIN TENSIONS EQUIVALENT TO 75 (± 5) POUNDS AT 70 (± 5) DEGREES FAHRENHEIT WITH THE RUDDER CONTROL SURFACE FAIRED TO THE VERTICAL STABILIZER.

NOTE

USE THE RUDDER CONTOUR TEMPLATE FROM THE CONTROL SURFACES RIGGING KIT LOCKHEED PART NO. 11019-MEU TO DETERMINE THE FAIRED POSITION OF THE RUDDER.



STEP 9

RECONNECT THE RUDDER TRIM ACTUATOR AND CHECK THE RUDDER TRIM RIGGING. RE-RIG THE TRIM SYSTEM IF NECESSARY.

STEP 10

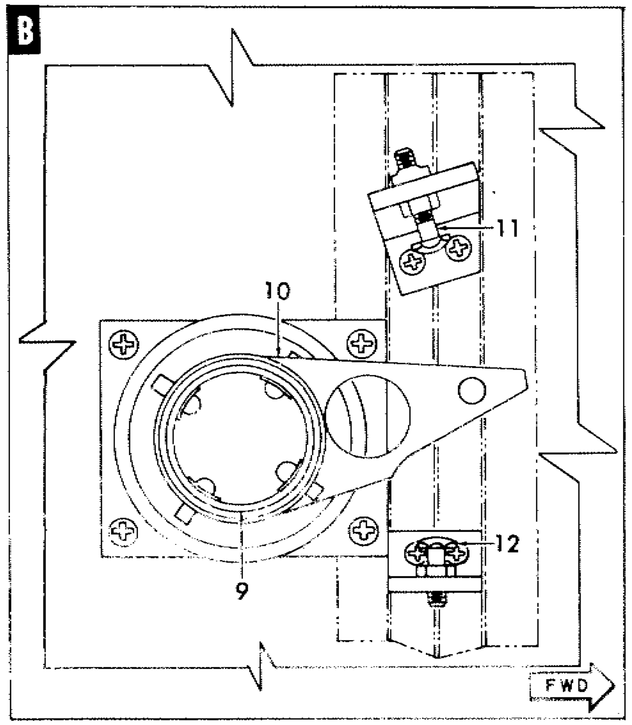
DISENGAGE THE RUDDER GUST LOCK AND CHECK THE RUDDER TRAVEL. THE RUDDER SHOULD TRAVEL 30 (± 1/2) DEGREES IN BOTH DIRECTIONS. ADJUST THE MECHANICAL STOPS (5) TO OBTAIN THIS TRAVEL.

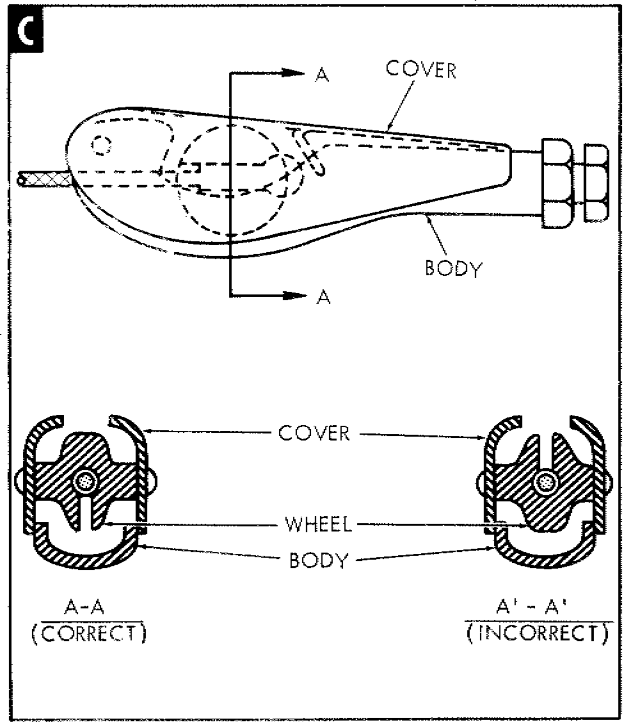
STEP 11

WHILE HOLDING THE RUDDER PEDAL IN THE EXTREME LEFT-TURN POSITION, ADJUST THE UPPER TORQUE TUBE STOP (11) TO OBTAIN A GAP OF 0.138 (± 0.030) INCH BETWEEN THE STOP (11) AND THE STOP LEVER (10). REPEAT THIS ADJUSTMENT FOR THE RIGHT-TURN POSITION USING THE LOWER STOP (12).

STEP 12

PERFORM AN OPERATIONAL CHECKOUT OF THE RUDDER CONTROL SYSTEM AND SAFETY ALL CONNECTIONS AS REQUIRED.





JS-2-2-2725

Figure 27-9. Rudder Control System Rigging Instructions

SECTION 27.3
TRIM CONTROL AND MACH TRIM SYSTEM

Contents Page

DESCRIPTION AND OPERATION 27-36
  General 27-36
  Aileron Trim System 27-36
  Aileron Trim Operation 27-36
  Aileron Trim Actuator Assembly 27-36
  Aileron Trim Control Switch and Position Indicator 27-36
  Rudder Trim System 27-36
  Rudder Trim Operation 27-36
  Rudder Trim Actuator Assembly 27-36A
  Rudder Trim Control Knob and Position Indicator 27-36A
  Pitch Trim System 27-36A
  Normal Pitch Trim Operation 27-36A
  Emergency Pitch Trim Operation 27-37
  Pitch Trim Actuator Assembly 27-37
  Pitch Trim Control Switches 27-37
  Horizontal Stabilizer Position Indicator 27-37
  Trim Disconnect System 27-37
  Trim Disconnect System Operation 27-37
  Mach Trim Compensator System Operation 27-38
  Control Panel 27-38
  Control Switch 27-38
  Test Switch 27-38
  Warning Light 27-39
OPERATIONAL CHECKOUT 27-39
  Aileron Trim System 27-39
  Rudder Trim System 27-39
  Pitch Trim System 27-40
  Trim System Disconnect Circuits 27-41
  Mach Trim Compensator System 27-41

TROUBLESHOOTING 27-42A
  Aileron Trim System 27-42A
  Rudder Trim System 27-43
  Pitch Trim System 27-43
  Mach Trim Compensator System 27-44

REMOVAL AND REPLACEMENT 27-45
  Aileron and Rudder Trim Actuator Handling 27-45
  Pitch Trim Actuator Assembly 27-45
  Drive Motor 27-45
  Removal 27-45
  Replacement 27-46

ADJUSTMENT 27-46
  Aileron Trim System Adjustment Procedures 27-46
  Rudder Trim System Adjustment Procedures 27-46
  Timing the Pitch Trim Actuator Halves 27-46
  Mach Trim Compensator System 27-46A

MINOR REPAIR, INSPECTION, AND SERVICING 27-47
  Empennage Attachment Point Wear Tolerances 27-47

FIGURES
  27-10. Aileron and Rudder Trim Systems Schematic Diagram 27-49
  27-11. Pitch Trim Switch Assembly (Part No. JR 2002-3) Terminal Locking Instructions 27-50
  27-12. Pitch Trim System Schematic Diagram 27-51
  27-13. Trim Disconnect System Schematic Diagram (Airplanes 5001 thru 5069) 27-52
  27-13A. Trim Disconnect System Schematic Diagram (Airplanes 5070 and Up) 27-52A

Figures — Page

27-14. Aileron Trim Actuator Adjustment 27-53
27-15. Rudder Trim Actuator Adjustment 27-54
27-16. Mach Trim Compensator Schematic Diagram (Airplanes 5001 thru 5085) 27-55
27-16A. Mach Trim Compensator Schematic Diagram (Airplanes 5086 and Up) 27-56
27-17. Pitch Trim Actuator-and-Empennage Attach Points 27-56A
27-18. Removal and Replacement of the Pitch Trim Actuator Assembly 27-57
27-18A. Pitch Trim Actuator Electrical Schematic 27-58C

27.3.1 DESCRIPTION AND OPERATION.

27.3.1.1 GENERAL. Lateral, directional, and pitch trim are controlled by electric-motor driven actuators. Lateral trim is accomplished by actuating a trim tab on the left aileron. Directional trim is accomplished by actuating a trim tab on the rudder. Pitch trim is accomplished by pivoting the empennage. A trim disconnect circuit, controlled by pushbuttons on the control wheels, breaks the trim circuits. A mach trim compensator system automatically trims the airplane nose-up as speed increases through a range of from 0.8 to 0.89 mach, and trims the airplane nosedown as speed decreases through the same range. When the autopilot is engaged, the mach trim signals are isolated from the pitch trim system, but the mach trim compensator unit remains synchronized to autopilot trimming, so when the autopilot is disengaged the mach trim compensator system will not produce unwanted pitch signals. The mach trim compensator unit in the nose electronic equipment area is controlled by pitot and static pressures. Power for the system is 28-volt, DC and 115-volt, AC.

27.3.1.2 AILERON TRIM SYSTEM. (See figure 27-10.) The aileron trim system is made up of an electric-motor driven actuator mounted in the left aileron, linkage from the actuator to the trim tab, a control switch on the center console, and a position indicator next to the control switch to show the degree of aileron trim. The position transmitter is part of the actuator assembly.

27.3.1.3 AILERON TRIM OPERATION. (See figure 27-10.) Placing the trim switch in the left-wing-up position completes a circuit through the de-energized aileron-rudder trim disconnect relay and the trim switch to the extension windings of the aileron trim actuator motor. Extending the actuator moves the aileron tab up to move the left aileron down. The actuator motor operates until the control switch is released or until the actuator reaches its travel limit to open a limit switch in the motor assembly.

27.3.1.4 AILERON TRIM ACTUATOR ASSEMBLY. The aileron trim actuator assembly is made up of an electric motor, two drive assemblies, a position transmitter, and limit switches. The motor is reversible and connects to a reduction gear train which drives a driver sprocket. This driver sprocket drives two driven sprockets attached to jackscrews which connect to the actuator rod ends. The sprockets are connected by chain assemblies, and the driven sprockets operate the limit switches and position transmitter.

27.3.1.5 AILERON TRIM CONTROL SWITCH AND POSITION INDICATOR. An aileron trim switch and an aileron trim indicator are on the center console. Moving the switch to the right operates the actuator motor to extend the jackscrews and thus move the trim tab up. The control switch is spring loaded to return to the off position when released. The tab position indicator shows the amount of trim in degrees of tab travel.

27.3.1.6 RUDDER TRIM SYSTEM. (See figure 27-10.) The rudder trim system is made up of an electric-motor driven actuator assembly, linkage from the actuator to the rudder tab, a control switch, and a position indicator. The actuator assembly contains an electric motor and a tab position transmitter. Motor operation is controlled by the control switch on the center console, and tab position is shown by the indicator near the switch. The actuator assembly is attached to the structure of the vertical stabilizer with its jackscrew shafts connected to a bellcrank mechanism. The bellcrank mechanism connects to pushrods which actuate the rudder tab. The bellcrank and pushrod mechanism is attached to the structure so as to provide a balance tab mechanism in addition to a trim tab actuating mechanism. The balance tab operation in no way effects the trimming operation.

27.3.1.7 RUDDER TRIM OPERATION. Rotating the rudder trim control knob on the center console operates contacts in the switch to complete a circuit through the de-energized aileron-rudder trim disconnect relay, through the trim switch, and to the reversible motor in the trim actuator assembly. This motor drives two jackscrews to position the rudder tab. Rotating the knob left actuates the tab to the right. Once initiated, a trimming operation continues until the knob is released or until the tab reaches its travel limit where a limit switch in the actuator assembly opens to break the actuating circuit. A linkage and bellcrank mechanism connect the jackscrews of the actuator assembly to the rudder trim tab. Trimming the direction of the airplane changes the relative position of the tab with the rudder and has no effect on the balance tab effect described in the rudder control system description.

27.3.1.8 RUDDER TRIM ACTUATOR ASSEMBLY. The rudder trim actuator assembly is made up of an electric motor, two drive assemblies, a position transmitter, and limit switches. The motor is reversible and connects to a reduction gear train which drives a driver sprocket. This driver sprocket drives two driven sprockets by chain assemblies. The driven sprockets drive the jackscrews, the position transmitter, and the limit switch cam.

27.3.1.9 RUDDER TRIM CONTROL KNOB AND POSITION INDICATOR. A rudder trim control knob and a position indication are on the center console. The trim knob is spring loaded to the off position. Rotating this knob to the right energizes the actuating motor to drive the rudder trim tab to the left of the rudder surface. The position indicator shows tab trim in degrees. The position indicator does not show the actual tab position due to the action of the balance mechanism; it shows only what the tab position would be if the rudder were in the faired position.

27.3.1.10 PITCH TRIM SYSTEM. (See figure 27-12.) The pitch trim system consists of a dual actuator that pivots the entire empennage to trim the airplane for nose-up or nose-down trim. The actuator contains two electric motors, one for normal actuation and one for emergency actuation. Both motors are drive screwjacks attached to the front of the empennage.

27.3.1.11 NORMAL PITCH TRIM OPERATION. The pitch trim system is capable of trimming the airplane a maximum of 1 degree nose-down or 9 degrees nose-up. While using the normal actuating motor, the maximum nose-up trim with the flaps fully up is 4 degrees.

27.3.1.11.1 Selecting nose-up trim on either the pilot's or the copilot's HOR STAB TRIM switches allows 28-volt, DC power to go from the main bus, through the de-energized nose-down stabilizer trim relay, through the de-energized autopilot transfer relay, to the normal stabilizer actuator ground contactor. This contactor provides a ground to the normal actuating motor.

27.3.1.11.2 Power also goes through the de-energized nose-down stabilizer trim relay to energize the nose-up stabilizer trim relay, which in turn energizes the normal power stabilizer actuator contactor. This contactor sends power from the main bus to retract the normal actuator and thus trim the airplane for nose-up. As the actuator reaches 4 degrees nose-up trim, if the flap control lever is in the fully-up position, the 4-degree limit switch opens to break the circuit. If the flap control lever is not in the fully-up position, the circuit is maintained through the 9-degree limit switch until the actuator reaches the 9-degree nose-up position.

27.3.1.11.3 Normal nose-down trim actuation is simiiar to nose-up trim actuation except power energizes the nose-down stabilizer relay instead of the nose-up stabilizer relay, and there is just one limit of travel. The nose-down trim limit is 1 degree, and is unaffected by the position of the flap control lever.

27.3.1.11.4 During operation, power is applied to the clutch coil and through the normally closed contacts of the brake release power transfer relay to the brake release coil to engage the primary motor clutch. When power is removed the drive spindle is separated from the primary motor and is mechanically braked to prevent overtravel of the pitch trim.

27.3.1.12 EMERGENCY PITCH TRIM OPERATION. (See figure 27-13.) Placing the HOR STAB TRIM SYS switch in the "EMERGENCY" position selects emergency power from the essential bus. Selecting "NOSE-UP" or "NOSE-DOWN" trim with the EMER HOR STAB TRIM switch transmits power through the emergency power stabilizer actuator contactor. A relay in the contactor energizes and allows essential bus power to reach the emergency motor of the actuator. As the limit of 1 degree nose-down or 9 degrees nose-up trim is reached, actuator limit switches break the circuit to the actuator motor.

27.3.1.12.1 During operation, power is applied to the clutch coil and the brake release coil to engage the emergency motor clutch and also through the brake release power transfer relay normally open contacts to operate the brake release coil in the primary motor. When power is removed the primary motor drive spindle is braked, there is no brake in the emergency motor.

27.3.1.13 PITCH TRIM ACTUATOR ASSEMBLY. The pitch trim actuator assembly is made up of two halves, with an electric motor and jackscrew mechanism in each half. These halves are connected by gears to insure equal jackscrew travel. One motor drives the empennage for normal operation, and the other drives the empennage for emergency operation. Pitch trim control circuits prevent both motors from operating simultaneously. Mechanical stops and electrical limit switches limit jackscrew travel. The mechanical stops limit jackscrew travel between 20 and 29.41 inches as measured from the centerlines of the attachment fittings. The limit switches limit jackscrew travel by limiting motor operation according to flap control lever position and pitch trim system selection. For the actuator lengths at which these switches operate, refer to the procedures for timing the pitch trim actuator halves. As the halves of the actuator can become separated during handling, a handling fixture should be used to hold the halves together when the actuator is not installed in the airplane. If the halves become separated, they must be retimed according to the instructions in this section.

27.3.1.14 PITCH TRIM CONTROL SWITCHES. Normal pitch trim is manually controlled by a HOR STAB TRIM switch assembly on each control wheel. Each assembly is made up of two switches designed so both must be in the same trim position to energize the trim control circuits. This switch configuration prevents accidental actuation of the pitch trim system by touching one of the switches. When the HOR STAB TRIM SYS switch on the center console is in the "NORMAL" position, pitch trim is controlled by the trim switches on the control wheels. When the HOR STAB TRIM SYS switch is in the "EMERGENCY" position, pitch trim is controlled by the EMER HOR STAB TRIM switch on the center console.

27.3.1.15 HORIZONTAL STABILIZER POSITION INDICATOR. A horizontal stabilizer position indicator provides for horizontal stabilizer nose-up or nose-down position indications transmitted by the horizontal stabilizer normal trim actuator position transmitter. The horizontal stabilizer position indicator shows the degree of nose-up or nose-down movement in relationship to the empennage control surface. When the front of the empennage is moved down a nose-up indication is shown on the indicator; when the front of the empennage is moved up a nose-down indication is shown on the indicator.

27.3.1.16 TRIM DISCONNECT SYSTEM. (See figure 27-13.) Two TRIM DISC pushbuttons, one on each control wheel, control the aileron, rudder, and pitch trim disconnect circuits. A TRIM SYS RESET pushbutton on the center console is used to reset the trim systems. Engaging the autopilot automatically disconnects the manual trim circuits.

27.3.1.17 TRIM DISCONNECT SYSTEM OPERATION. (See figure 27-13.) With the HOR STAB TRIM SYS switch in the "NORMAL" position, pressing the TRIM DISC pushbutton on either control wheel energizes the aileron-rudder trim disconnect relay, the stabilizer trim disconnect relay, the trim system disconnect relay, and a solenoid in the normal power

27.3.1.17 (Cont).

stabilizer trim contactor. Energizing the stabilizer trim disconnect relay completes a holding circuit to keep the other relays in the energized position after the TRIM DISC pushbutton is released. With these relays energized, power is isolated from the aileron, rudder, and normal pitch trim circuits.

27.3.1.17.1 Placing the HOR STAB TRIM SYS switch in the "EMERGENCY" position energizes the trim system disconnect relay and a solenoid in the normal power stabilizer actuator contactor to cut off power to the normal pitch trim control circuits, and readies the emergency pitch trim circuits for operation. With the HOR STAB TRIM SYS switch in this position, pressing either of the TRIM DISC pushbuttons energizes the stabilizer trim disconnect relay, the aileron-rudder trim disconnect relay, the trim lockout relay, and a solenoid in the emergency power stabilizer actuator contactor. Energizing the trim lockout relay completes a holding circuit to keep the relays in the energized positions after the TRIM DISC pushbutton is released. With the circuits in this configuration, aileron and rudder trim circuits are broken, and both normal and emergency pitch trim control circuits are broken.

27.3.1.17.2 Pressing the TRIM SYS RESET pushbutton on the center console when the HOR STAB TRIM SYS switch is in the "NORMAL" position returns aileron, rudder, and normal pitch trim control to the pilots. Pressing the TRIM SYS RESET pushbutton when the HOR STAB TRIM SYS switch is in the "EMERGENCY" position returns aileron, rudder, and emergency pitch trim control to the pilots.

27.3.1.17.3 A TRIM DISC pushbutton on each control wheel can be pressed to disconnect aileron, rudder, and pitch trim control circuits. To reset the trim and thus return trim capabilities to the pilots, the TRIM SYS RESET pushbutton can be pressed. The TRIM SYS RESET pushbutton is on the center console near the throttle quadrant.

27.3.1.18 MACH TRIM COMPENSATOR SYSTEM OPERATION. (See figure 27-16.) The mach trim compensator system trims the airplane using the normal pitch trim motor. Manual trim overrides mach trim, and the autopilot isolates mach trim from the pitch trim motor so pitch trim commands come from but one source at a time. When the mach trim system is on and the autopilot is disengaged, pitch trim is always the resultant of manual and mach trim commands. For example, if the manual trim calls for one degree nose-down trim and the mach trim system calls for three degrees nose-up trim, the resultant trim will be two degrees nose up.

27.3.1.18.1 Output trim signals from the mach trim system are automatically isolated from the pitch trim motor when the autopilot is engaged, but the mach trim system remains synchronized to autopilot trimming operations. The autopilot controls pitch through the elevator control surfaces, and then follows up by trimming the horizontal stabilizer until the elevator control surfaces are again in a neutral position. This assures a trimmed condition at all times.

27.3.1.18.2 The mach trim compensator system warning light operates when the airplane is flying above 0.8 mach and mach trim signals are not being fed to the pitch trim system. The warning light operates when the airplane enters the mach trim range at too rapid a rate, and when manual trim is used (autopilot disengaged) above 0.8 mach to override the mach trim compensator.

27.3.1.18.3 A mach trim compensator system test switch on the mach trim control panel activates a test circuit in the mach trim system to run the mach trim system through a preset test cycle. This cycle consists of trimming the horizontal stabilizer approximately two to three degrees nose-up and then back to the neutral trim position.

27.3.1.18.4 CONTROL PANEL. A mach trim compensator control panel on the center console contains a two-position ("ON," "OFF") control switch; a momentary, two-position ("TEST," "OFF") MACH TRIM COMP TEST switch, and a MACH TRIM COMP OUT warning light. The test switch is labeled GROUND ONLY.

WARNING

Once a test cycle is initiated by using the MACH TRIM COMP TEST switch, the test cycle must be allowed to complete itself. If AC power to the system is turned off before the test cycle is complete, the remainder of the test cycle will occur the next time the system is turned on.

27.3.1.18.5 CONTROL SWITCH. A two-position ("ON," "OFF") control switch controls the power to the mach trim warning light (or lights), 28-volt, DC power to the mach trim compensator unit and horizontal stabilizer trim functions.

27.3.1.18.6 TEST SWITCH. A momentary, two-position ("TEST," "OFF") MACH TRIM COMP TEST switch on the mach trim control panel controls the test cycle of the mach trim system. This switch is labeled for GROUND ONLY operation. Placing the mach trim control switch in the "OFF" position before a test cycle is completed will stop the test cycle, but the remainder of the cycle will occur the next time the mach trim control switch is placed in the "ON" position. This test switch receives 28-volt, DC power from the main bus through the mach trim control circuit breaker.

27.3.1.18.7 WARNING LIGHT. A MACH TRIM COMP. OUT warning light on the center console goes on to show that the mach trim compensator system is not operating properly. In addition to this light on the center console, another warning light operates in the same manner on the copilot's side inclined panel on airplanes 5026 through 5085. For airplanes 5085 and up, a second light is provided on the annunciator panel.

27.3.2 OPERATIONAL CHECKOUT.

Tools Required

Nomenclature	Description	Use and Application
Control surfaces rigging kit.	Lockheed P/N 11019-MEU.	To determine the faired position of the rudder.
Rudder and trim travel gage.	Lockheed P/N 11068-TG.	To measure rudder tab travel.
Aileron and elevator protractor.	Lockheed P/N 11069-MEU.	To measure horizontal stabilizer travel.

27.3.2.1 AILERON TRIM SYSTEM. A means of measuring tab travel in inches or degrees is needed to perform this test. Inch measurements should be taken from the inboard trailing edge of the tab to the adjacent point on the aileron.

Note

> Maximum free play in trim tab is 0.08 inch, measured at longest chord.

a. Make sure the aileron trim and the trim disconnect breakers are engaged.

b. Connect 28-volt, DC power to the airplane.

c. Actuate the aileron trim tab to the neutral rig position and observe the aileron trim indicator on the center console.
RESULT: The trim indicator should show 0 (±0.5) degree.

d. Hold the aileron trim switch on the center console in the left-wing-up position until trim operation.
RESULT: The trim tab should move 1.37 (±0.09) inches, 15 (±1) degrees, up from the neutral rig position and the aileron trim indicator should show 15 (±2) degrees left wing up.

e. Repeat step (d) for right-wing-up operation. Results should be the same, although for opposite motion.

f. Actuate the tab to the faired position, releasing the trim switch several times during tab operation.
RESULT: The tab should stop moving each time the switch is released.

g. Hold the aileron trim switch in either trim position and, while the tab is moving, press the TRIM DISC pushbutton on the pilot's control wheel. After tab movement stops, press the TRIM SYS RESET pushbutton on the center console near the No. 4 throttle.
RESULT: Tab movement should stop when the TRIM DISC pushbutton is pressed and should resume when the TRIM SYS RESET pushbutton is pressed.

h. Repeat step (g) using the TRIM DISC switch on the copilot's control wheel. Results should be the same.

i. Return the tab to the neutral rig position, disconnect power from the airplane, and return the aileron trim system to normal.

27.3.2.1.1 FIXED TRIM TAB. Airplanes 5046 and up. Also airplanes 5001 through 5045 with Service Bulletin 329-289 incorporated.

Note

> Maximum allowable rig position is .25 inch offset of outboard trailing edge of tab from adjacent fixed trailing edge of aileron.

Use and/or disclosure is governed by the statement on the title page of this document.

27.3.2.1.1 (Cont).

Adjustment of the right hand tab may be as follows:

a. Remove the bolt at the forward end of the link and lower trunnion fitting out of aileron fitting.

b. Loosen jam nut on rod end fitting and rotate trunnion fitting as required.

c. Rotate link an additional 1/2 to 1 turn after correct position is established to create a slight preload.

d. Check that all connections bolts, nuts and adjustment screws are secure and safetied.

e. Perform an operational checkout of the system.

27.3.2.2 RUDDER TRIM SYSTEM. A means of measuring rudder trim tab travel in inches or degrees is needed to perform this test. When measuring tab travel in inches, measure from the top trailing edge of the tab to a point adjacent to it on the rudder.

Note

Maximum free play in trim tab is 0.12 in., measured at the longest chord.

a. Make sure the rudder trim and the trim disconnect circuit breakers are engaged.

b. Connect 28-volt, DC power to the airplane.

c. Hold the rudder trim knob on the center console in the left trim position until tab actuation ceases.
RESULT: The rudder trim tab should travel 1.27 (±0.07) inches, 9.75 (±1) degrees, to the right of the rudder tab rig angle and trim indicator should show 9.75 (±1) degrees left trim.

d. Hold the rudder trim knob in either trim position and, while the tab is moving, press the TRIM DISC pushbutton on the pilot's control wheel. After tab movement stops, press the TRIM SYS RESET pushbutton on the center console near the No. 4 throttle.
RESULT: Tab movement should stop when the TRIM DISC pushbutton is pressed, and should resume when the TRIM SYS RESET pushbutton is pressed.

e. Repeat step (d) using the TRIM DISC pushbutton on the copilot's control wheel. Results should be the same.

f. Actuate the rudder trim tab to the rudder tab rig angle and observe the trim indicator.

g. Disconnect power from the airplane, and return the rudder trim system to normal.

27.3.2.3 PITCH TRIM SYSTEM.

a. Make sure the following circuit breakers are engaged:

STAB NOR TRIM CONT
NO. 1 MAIN INVERTER CONT
WARNING LIGHTS
MACH TRIM WARN LT
EMER TRIM CONT
MACH TRIM CONT DC
MACH TRIM CONT AC
TRIM DISC
STAB TRIM POS IND

b. Connect 28-volt, DC power to the airplane.

c. Make sure the HOR STAB TRIM SYS switch on the center console is in the "NORMAL" (guardded) position, and check that the flap control lever is in the flaps up position. It is not necessary to energize the flap systems during this test.

d. Place one of the pilot's HOR STAB TRIM switches in the "NOSE DOWN" position; then release it and place the other trim switch on the pilot's control wheel in the "NOSE DOWN" position.
RESULT: The trim system should not be affected.

e. Place and hold both of the pilot's HOR STAB TRIM switches in the "NOSE DOWN" position until trim actuation stops.

f. Attach a protractor to the horizontal stabilizer, and set it to read 1 degree stabilizer up. With the stabilizer in this position, observe the trim indicator on the center console.
RESULT: The trim indicator should show 1 (±0.5) degrees nose-down trim.

Use and/or disclosure is governed by the statement on the title page of this document.

27.3.2.3 (Cont).

g. Place one of the pilot's HOR STAB TRIM switches in the "NOSE UP" position; then release it and place the other switch on the pilot's control wheel in the "NOSE UP" position.
RESULT: There should be no effect on the trim system.

h. Place and hold both of the pilot's HOR STAB TRIM switches in the "NOSE UP" position until trim actuation ceases.
RESULT: The stabilizer should move until the protractor shows 4 (±0.5) degrees stabilizer down. The stabilizer trim indicator should show 4 (±0.5) degrees nose-up trim.

i. Repeat steps (d) through (h) using the copilot's HOR STAB TRIM switches. Results should be the same.

j. Place and hold the pilot's HOR STAB TRIM switches in the "NOSE DOWN" position; and while the trim-system is operating, place and hold the copilot's HOR STAB TRIM switches in the "NOSE UP" position.
RESULT: The stabilizer should trim for nose down with the copilot's switches having no effect on the system.

k. Place and hold the pilot's HOR STAB TRIM switches in the "NOSE UP" position; and while the trim system is operating, place the copilot's HOR STAB TRIM switches in the "NOSE DOWN" position.
RESULT: The stabilizer should trim for nose up with the copilot's switches having no effect on the system.

l. Place one of the pilot's HOR STAB TRIM switches in the "NOSE DOWN" position, and simultaneously place the other trim switch on the pilot's control wheel in the "NOSE UP" position.
RESULT: The trim system should not be affected.

m. Repeat step (l) with the switches in the opposite positions. The results should be the same.

n. Place one of the copilot's HOR STAB TRIM switches in the "NOSE UP" position, and place the other trim switch on the copilot's control wheel in the "NOSE DOWN" position.
RESULT: There should be no effect on the trim system.

o. Repeat step (n) with the switches in the opposite positions. The results should be the same.

Use and/or disclosure is governed by the statement on the title page of this document.

27.3.2.3 (Cont).

p. Disengage the WING FLAP LE AND TE CONTROL circuit breaker, and place the flap control lever in the fully-aft position.

q. Place and hold both of the pilot's HOR STAB TRIM switches in the "NOSE UP" position until trim actuation ceases.
RESULT: The stabilizer should actuate until the protractor indicates 9 (±0.5) degrees stabilizer down. At this position, the stabilizer trim indicator should show 9 (±0.5) degrees nose-up trim.

r. Place and hold the copilot's HOR STAB TRIM switches in the "NOSE DOWN" position; and while the stabilizer is changing position, momentarily place the HOR STAB TRIM SYS switch in the "EMERGENCY" (unguarded) position.
RESULT: The stabilizer should stop moving when the HOR STAB TRIM SYS switch in the "EMERGENCY" position, and it should resume movement when the switch is returned to the "NORMAL" (guarded) position.

s. While actuating the trim system in either direction, press the copilot's TRIM DISC pushbutton, and after trimming ceases, press the TRIM SYS RESET pushbutton on the center console near the No. 4 throttle.
RESULT: The trim system should cease movement when the TRIM DISC pushbutton is pressed, and should resume movement when the TRIM SYS RESET pushbutton is pressed.

t. Using the pilot's HOR STAB TRIM switch, position the horizontal stabilizer in the zero trim position.
RESULT: The protractor should show 0 (±0.5) degree stabilizer angle, and the stabilizer position indicator should show 0 (±0.5) degree trim.

u. Disconnect the normal trim system by pressing the pilot's TRIM DISC pushbutton, and then place the HOR STAB TRIM SYS switch in the "EMERGENCY" (unguarded) position.

v. Place and hold the EMER HOR STAB TRIM switch in the "NOSE DOWN" position until actuation ceases.
RESULT: The stabilizer should stop at 1 (±0.5) degrees nose-down trim as shown on the stabilizer trim indicator. The protractor should show 1 (±0.5) degrees stabilizer up.

w. Place and hold the EMER HOR STAB TRIM switch in the "NOSE UP" position until trimming stops.
RESULT: The stabilizer should stop at 9 (±0.5) degrees nose-up trim as shown on the stabilizer trim indicator. The protractor should show 9 (±0.5) degrees stabilizer down.

x. While trimming for nose-down, press the pilot's TRIM DISC pushbutton. After trimming action ceases, press the TRIM SYS RESET pushbutton.
RESULT: Pressing the TRIM DISC pushbutton should stop trim operation. Pressing the TRIM SYS RESET pushbutton should start the trimming operation again.

y. Return the HOR STAB TRIM SYS switch to the "NORMAL" (guarded) position, and then momentarily place the EMER HOR STAB TRIM switch in the "NOSE UP" position.
RESULT: There should be no effect on the trim system.

z. Return the stabilizer to the zero trim position, disconnect power from the airplane, return the flap control lever to the fully-forward position, and engage the WING FLAP LE AND TE CONTROL circuit breaker.

aa. Remove the protractor from the horizontal stabilizer, and return the pitch trim system to normal.

27.3.2.4 TRIM SYSTEM DISCONNECT CIRCUITS. The trim disconnect and reset circuits are tested during the aileron, rudder, and pitch trim operational checkouts.

27.3.2.5 MACH TRIM COMPENSATOR SYSTEM. A source of air to pressurize the pitot heads to a pressure corresponding to 0.80 to 0.89 mach and an external source of 28-volt, DC power are needed to perform this test.

Note

This test should also be performed when a pitch trim actuator or primary motor is replaced and/or there is a change in actuator preload.

a. Make sure the following circuit breakers are engaged: warning light, mach trim control (28-volt, DC), mach trim control (115-volt, AC), trim disconnect, and main inverter control. The mach trim and main inverter switches should be in the off position.

b. Make sure the LE and TE wing flap control circuit breaker is disengaged, and then place the flap control lever in the "TAKEOFF/APPROACH" position.

c. Connect external power to the airplane to start the engines in accordance with the instructions in Chapter 72. Start the engines. Obtain airplane power to operate the mach trim system.

d. Position the stabilizer in the zero trim position, using the stabilizer position indicator on the center console as a guide.

27.3.2.5 (Cont).

e. Place the MACH TRIM COMP CONT switch and the main inverter switch in the "ON" position. and observe the MACH TRIM COMP OUT lights and the master CAUTION lights.
RESULT: The MACH TRIM COMP OUT light and the master CAUTION lights should go out.

f. **Place the MACH TRIM COMP TEST switch in the "TEST" position, and then release it. Observe the MACH TRIM COMP OUT light and the master caution lights. Also observe the stabilizer position indicator, and check the rate of change of the stabilizer angle by using the clock on the instrument panel to time the change from zero to 2 degrees nose-up.**
RESULT: The MACH TRIM COMP OUT light should go on, and the master caution lights should remain off. The horizontal stabilizer should trim to 2.3 (±1.0) degrees nose up, as indicated by the stabilizer position indicator. At this point of the test cycle, the MACH TRIM COMP OUT light should go out momentarily, and the stabilizer should return to zero trim. The master caution lights should not go on during this test cycle. The trim rate should be between 0.10 and 0.13 degrees per second (7.7 to 10.0 seconds per degree of travel).

g. **Place the MACH TRIM COMP TEST switch in the "TEST" position, and when the stabilizer starts nose-up trim, press one of the TRIM DISC buttons on one of the control wheels.**
RESULT: The stabilizer should start trimming nose-up when the test switch is placed in "TEST," and it should stop when the TRIM DISC button is pressed.

h. In less than 3 seconds after depressing the trim disconnect switch, press the trim reset button.
RESULT: The stabilizer shall resume movement from interrupted position and may drive below the neutral position.

i. **Return the horizontal stabilizer to the zero trim position, and place the MACH TRIM COMP TEST switch in the "TEST" position during the down-trimming portion of the test cycle, press one of the TRIM DISC buttons.**
RESULT: The stabilizer trimming action should stop.

j. **Ten seconds after stabilizer trimming stops, press the trim reset button.**
RESULT: There should be no effect on the stabilizer trim system.

k. **Return the horizontal stabilizer to the zero trim position, and place the MACH TRIM COMP TEST switch in the "TEST" position.**

l. **During the up portion of the test cycle, place the MACH TRIM COMP CONT switch in the "OFF" position.**
RESULT: The test cycle should stop, and the MACH TRIM COMP OUT light should go on.

Note

The master caution lights may or may not go on during a test cycle, as operation of these lights is a function of relay operating time.

m. Place the mach trim comp control switch in the "ON" position.
RESULT: The MACH TRIM COMP OUT light should remain on, the master caution lights (if on) should go off, and the stabilizer should resume travel. Stabilizer travel may be in either direction.

n. **Using the pilot's trim switch, position the stabilizer at zero trim, and then place the test switch in the "TEST" position. When the stabilizer trim position indicator reaches 1 degree nose-up trim, hold the pilot's trim switch in the "NOSE UP" position until the stabilizer reaches the 3-degree nose-up trim position; then release it.**
RESULT: Stabilizer trim should reach 3 degrees nose-up, and then the trimming action should cease as the pilot's trim switch is released.

o. **Return the horizontal stabilizer to zero trim, using the pilot's trim switch.**

p. **Place the Autopilot & "Q" Switch Pitot Shutoff valve on the copilot's inclined panel in the "OPEN" position. Disconnect and plug the pitot system tubing at the following components:**

(1) Pilot's and copilot's airspeed indicator.

(2) Autopilot airspeed differential pressure switch.

(3) Overspeed warning switch.

Ensure that plugged tubing does not leak.

q. **Connect a calibrated pressure source to the pilot's and copilot's pitot head, and seal drain holes in pitot head.**

r. **Slowly apply equal pressure to both pitot heads until the machmeter indicates 0.80 (±0.01) mach. Allow the pressure to stabilize for 2 minutes.**
RESULT: At 0.80 (±0.01) mach, the stabilizer trim system should start trimming for nose-up.

s. **Allow the pitot systems to bleed to ambient pressure, turn the mach trim compensator system off, and return the system to normal.**

27.3.3 TROUBLESHOOTING.

27.3.3.1 AILERON TRIM SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INCORRECT AILERON TRIM INDICATION.		
Trim position transmitter improperly adjusted.	Actuate the aileron trim tab to the tab rig angle. The trim indicator should show zero trim in this position.	Adjust the position transmitter as needed.
Defective position transmitter or indicator.	Check operation of the transmitter and indicator.	Replace defective units and adjust as needed.
INCORRECT AILERON TRIM TAB TRAVEL.		
Limit switches in aileron trim actuator assembly improperly adjusted, or tab system improperly adjusted. or tab system improperly rigged.	Check the total aileron trim tab travel.	If the total travel is within limits, adjust the tab. If the total travel is not within limits, adjust the limit switches in the actuator assembly.
AILERON TAB ACTUATOR INOPERATIVE.		
Disengaged or defective aileron trim circuit breaker.	Check for disengaged or defective aileron trim circuit breaker.	Engage or replace circuit breaker as needed.
Trim disconnect circuit energized.	Press the TRIM SYS RESET button on the center console.	If the TRIM SYS RESET button fails to return aileron trim capabilities, check the disconnect relays for operation.
Defective aileron trim actuator, motor, or open circuit.	Check motor operation.	Replace defective motor, or repair circuit as needed.

27.3.3.2 RUDDER TRIM SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INCORRECT RUDDER TRIM INDICATION.		
Trim position transmitter incorrectly adjusted.	Actuate the rudder trim tab to the rig angle. The trim indicator should show zero trim in this position.	Adjust the position transmitter as needed.
Defective position transmitter or indicator.	Check operation of the transmitter and indicator.	Replace defective components and readjust system as needed.
INCORRECT RUDDER TAB TRAVEL.		
Limit switches in rudder trim actuator improperly adjusted.	Check total tab travel with the rudder in the faired position.	If the total tab travel is not within limits, adjust the limit switches. If the total travel is within limits, adjust the tab linkage.
Binding or interference in tab actuating linkage.	Check tab actuating linkage for binding or interference.	Relieve binding or interference as needed.
RUDDER TAB ACTUATOR INOPERATIVE.		
Disengaged or defective rudder trim circuit breaker. Defective rudder trim actuator motor.	Check for disengaged or defective rudder trim circuit breaker. Check operation of the actuator motor.	Engage or replace circuit breaker as needed. Repair or replace actuator assembly as needed.
Energized trim disconnect circuit.	Press TRIM SYS RESET button.	If trim system fails to reset, check the trim disconnect relays and circuits.
Defective rudder trim switch or open trim control circuit.	Check circuit and trim switch.	Repair circuit or replace trim switch as needed.

27.3.3.3 PITCH TRIM SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRIM SYSTEM OPERATES ON EITHER HALF OF "STAB TRIM" SWITCH ASSEMBLY.		
Defective or incorrectly wired trim switches.	Check operation and wiring of trim switches.	Replace defective switches, or repair circuit as needed.
PITCH TRIM ACTUATOR FAILS TO STOP AT PROPER TRAVEL LIMITS.		
Actuator limit switches improperly adjusted.	Check actuator lengths against trim limits as given in timing the pitch trim actuator halves.	Adjust the trim switches as needed.
Defective or improperly adjusted maximum trim limit switch in flap control lever assembly.	Check operation of the switch in the flap control lever assembly.	Adjust or replace switch as needed.

27.3.3.3 PITCH TRIM SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INCORRECT PITCH TRIM INDICATION.		
Defective or improperly adjusted position transmitter.	Check operation and adjustment of the pitch trim transmitter.	Replace components as needed.
PILOTS' TRIM COMMANDS INTERFERE WITH EACH OTHER.		
Defective or improperly wired nose-up or nose-down stabilizer trim relay.	Check operation and wiring of the stabilizer trim relays.	Repair circuits or replaced components as needed.
NORMAL PITCH TRIM INOPERATIVE.		
Normal power stabilizer actuator contactor energized.	Check the position of the HOR STAB TRIM SYS switch.	Place the switch in the "NORMAL" position.
Defective normal pitch trim actuator motor.	Check the operation of the normal pitch trim motor by using the normal trim system switches.	Replace the pitch trim actuator as needed.
Defective normal stabilizer actuator current limiter.	Check the current limiter.	Replace the current limiter as needed.
Defective normal stabilizer actuator contactors.	Check the operation of the normal circuit contactors.	Replace contactors as needed.

27.3.3.4 MACH TRIM COMPENSATOR SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MACH TRIM COMPENSATOR FAILS TO FUNCTION IN MACH TRIM RANGE.		
Disengaged or defective mach trim control circuit breakers.	Check both the AC and the DC control circuit breakers.	Replace or engage the circuit breakers as needed.
Defective mach trim compensator unit.	Perform a continuity check of the airplane wiring.	Replace the trim unit as needed.
Pitot and static ports of compensator unit not receiving proper pressures.	Check the ports of the unit for restrictions.	Obtain proper porting as needed.
Horizontal stabilizer trim selector switch in the emergency position.	Check the normal pitch trim system selector switch position.	Reposition the selector switch as needed.
Autopilot engaged.	Check autopilot control switches.	Disengage the autopilot.
MACH TRIM COMPENSATOR SYSTEM FAILS TO FUNCTION PROPERLY DURING TEST CYCLE.		
Defective mach trim test circuit in mach trim compensator unit.	Check airplane wiring.	Replace compensator unit as needed.
Improperly wired MACH TRIM COMP TEST switch.	Perform a continuity check of the airplane wiring.	Rewire switch as needed.

27.3.3.4 MACH TRIM COMPENSATOR SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MACH TRIM RATE NOT WITHIN PRESCRIBED LIMITS.		
Improperly adjusted mach trim compensator unit.	Check signals from compensator unit for signals duration.	Adjust the unit as needed. (See paragraph 27.3.5.4.)
Insufficient power supply to mach trim compensator system.	Check power supply to compensator unit.	Obtain proper airplane power supply.
Defective normal pitch trim actuator motor.	Check the operation of the pitch trim motor using the normal trim switches.	Replace the trim actuator as needed.

27.3.4 REMOVAL AND REPLACEMENT.

Tools Required

Nomenclature	Description	Use and Application
Empennage Lock	Lockheed P/N 11078-MEU	To prevent the empennage from tilting backwards when the pitch trim actuator is disconnected.

27.3.4.1 AILERON AND RUDDER TRIM ACTUATOR HANDLING. When handling aileron or rudder trim actuators, make sure the actuating rods remain parallel. Spacers may be locally manufactured to keep the rods of the actuators approximately 2.09 inches apart during handling. This dimension is as measured from the insides of the rods, not center to center.

CAUTION

Do not exert pressure on actuator rods during handling. To force the rods away from their normally parallel position may break internal drive components rendering the actuator unserviceable. (An overhaul manual for these actuators is available from the Talley Corp., Newbury Park, Calif.)

27.3.4.2 PITCH TRIM ACTUATOR ASSEMBLY. To remove or replace the pitch trim actuator assembly, refer to figure 27-18.

27.3.4.3 DRIVE MOTOR. This is a procedure for the removal and replacement of the primary motor in the aircraft. A similar procedure can be used for the removal and replacment of the emergency motor.

27.3.4.3.1 REMOVAL.

a. The primary motor of the actuator assembly can be removed and replaced in the aircraft as follows:

(1) Drain oil per Section 12.9.

(2) Disconnect the two flex lines at the sump can.

(3) Remove the aircraft wiring terminal access plate.

(4) Disconnect the main electrical wires from "J" box and remove these clamps.

(5) Remove the five screws on the upper edge of the "J" box.

(6) Remove the second and fourth bolts in the area of the six wiring clamps. These bolts attach the sump box to the "J" box.

(7) Remove the two screws on the bottom of each motor that attach the sump box clips.

(8) Remove the sump box.

b. Remove the two bolts through the chock which is between the motor and the screw jack (the chock cannot be removed until later). Then remove the two primary motor mount bolts common to the "J" box mount bracket flange. This will allow sufficient movement of the filter box to facilitate the rest of the removal procedure.

27.3.4.3.1 (Cont).

c. Disconnect wires A, B, C, D, and E of the primary motor, P/N 1506D183, (the right hand motor when looking forward at the actuator as installed in the air craft) at filter box terminals. Mark terminals A, B, C, D, and E to facilitate installation of replacement motor. It may be necessary to cut off "stake-on" connectors to pull the wires through the rubber grommet in the back of the box.

d. Remove the two bolts through the chock between the two motors common to the band assembly. Remove the chock, and then remove the two remaining bolts in the primary motor mounting flange; remove the motor. The chock between the motor and screw jack will then fall free. Take care to retain any shims installed between the split chocks (two places) for reinstallation.

27.3.4.3.2 REPLACEMENT.

a. Install the replacement motor as follows:

(1) Loosely install the replacement motor using AN4-10 bolts with MS21042-4 self-locking nuts or AN310-4 nuts with AN381-2-10 cotter pins. Use plain washers as required.

(2) Reinstall the chock between the motor and jackscrew (loose).

(3) Torque the motor mount inboard two bolts to 90 (±10) inch-pounds.

(4) Reinstall the chock between the two motors and install the bolts through the band assembly. Make sure all shims are in place.

(5) Insert the wires through the rubber grommet in the back of the filter box. Attach the wires to the same terminals as marked. Stake-on connectors may be required.

CAUTION

Terminate shielded wires in accordance with paragraph 24.3.6 of this manual. Make sure that the shielded wires are terminated properly so that there is no contact with the stake-on connector, if installed. Contact of the shielded wires with the stake-on connector will result in activation of the trim system from either half of the pilot's or copilot's horizontal stabilizer trim switch.

(6) Reposition the filter box, and torque the two out-board motor mount bolts to 90 (±10) inch-pounds. Install the two outboard bolts through the clamp assembly.

(7) Refer to the attached wiring diagram to check out that the wires are attached to their proper terminals.

b. Reinstall the sump, access plate, and flex lines.

c. Service with oil per Section 12.9.

27.3.5 ADJUSTMENT.

Tools Required

Nomenclature	Description	Use and Application
Control surfaces rigging kit.	Lockheed P/N 11019-MEU	To determine the faired position of the rudder.
Rudder and trim travel kit.	Lockheed P/N 11068-TG	To measure rudder tab travel.
Aileron and elevator protractor.	Lockheed P/N 11069-9-MEU	To measure horizontal stabilizer travel.

27.3.5.1 AILERON TRIM SYSTEM ADJUSTMENT PROCEDURES. (Refer to figure 27-14.)

27.3.5.2 RUDDER TRIM SYSTEM ADJUSTMENT PROCEDURES. (Refer to figure 27-15.)

27.3.5.3 TIMING THE PITCH TRIM ACTUATOR HALVES. If the pitch trim actuator halves become separated and thus get out of time, retime them according to these procedures:

a. **Connect 12- to 15-volts DC directly to the motor studs inside the terminal box**, and drive the right actuator half either to the extension mechanical stop or to the retraction mechanical stop.

b. Drive the left half of the actuator to its corresponding mechanical stop by turning the cross-drive shaft by hand.

c. With both jackscrews against their mechanical stops, reassemble the halves.

27.3.5.3 (Cont).

d. The limit switches should open at the following values when properly adjusted:

NORMAL MOTOR LIMIT SWITCHES	JACKSCREW LENGTH (IN INCHES)	DIRECTION OF TRAVEL
9 degrees nose up	20.121 (±0.080)	retracting
4 degrees nose up	24.701 (±0.080)	retracting
1 degree nose down	29.289 (±0.080)	extending

EMERGENCY MOTOR LIMIT SWITCHES	JACKSCREW LENGTH (IN INCHES)	DIRECTION OF TRAVEL
9 degrees nose up	20.121 (±0.080)	retracting
1 degree nose down	29.29 (±0.080)	extending

27.3.5.3 (Cont).

Note

Measure jackscrew length by inserting No. 30 drill shanks into the holes in the head of the mounting bolts. Use 36-inch vernier calipers to measure the distance from the outsides of the shanks; then subtract the diameter of one shank (0.128 inch) from the measurement to allow for true distance from centerline to centerline.

27.3.5.4 MACH TRIM COMPENSATOR SYSTEM.

Note

This procedure is for adjusting the trim rate only. This procedure should also be followed when a pitch trim actuator or primary motor is replaced and/or there is a change in actual pre-load.

a. Gain access to the mach trim compensator unit on the nose electronics equipment rack, and remove the adjustment access plug in the top of the unit.

b. Loosen the locknut holding the adjustment shaft until the shaft can be turned freely.

27.3.5.4 (Cont).

CAUTION

Do not loosen the retaining nut beneath the locknut. To do so may allow the unit to turn and place strain on the wires.

c. Start all engines and use the airplane generators and inverters during this adjustment.

d. Place the mach trim compensator control switch on the center console in the "ON" position, and then place the MACH TRIM COMP TEST switch in the test position.

e. As the test cycle is operating the pitch trim actuator, time the rate of trim change using the instrument panel clock and the horizontal trim indicator on the center console.

f. Use a small common screwdriver to adjust the compensator unit to obtain a trim rate of from 0.10 to 0.13 degrees per second (7.7 - 10.0 seconds per degree of travel).

27.3.5.4 (Cont).

g. After the proper trim rate is obtained, tighten the locknut on the adjustment shaft, making sure not to change the adjustment while doing so.

h. Shut down the engines, reinstall the adjustment plug, remove external power, and return the airplane to normal.

WARNING

Once a test cycle is initiated, by using the MACH TRIM COMP TEST switch, the test cycle must be allowed to complete itself. If AC power to the system is turned off before the test cycle is complete, the remainder of the test cycle will occur the next time the system is turned on.

27.3.6 MINOR REPAIR, INSPECTION, AND SERVICING.

27.3.6.1 EMPENNAGE ATTACHMENT POINT WEAR TOLERANCES. (See figure 27-17.) The amount of allowable wear in empennage scissors joints, actuator fittings, and empennage pivot points is as follows:

a. Side play in any one of the three scissors joints should not exceed 0.0035 inch. Radial play (play between the bolts and their bushings) in any of the three joints should not exceed 0.007 inch.

b. Radial play (play between the bolts and the fittings) should not exceed 0.007 inch in either the empennage or the actuator fittings. Longitudinal play should not exceed 0.019 inch with no preload on the actuator. Longitudinal play is the up-and-down play as the actuator is installed in the airplane.

Note

Longitudinal play is the play between the bolts and the empennage and bulkhead fittings, added to the play between the bolts and the actuator rod ends, but does not include internal play within the actuator itself.

c. Side play in the empennage pivot fittings should not exceed 0.005 inch. Radial play should not exceed 0.005 inch. Radial play is the play between the pins and their bushings.

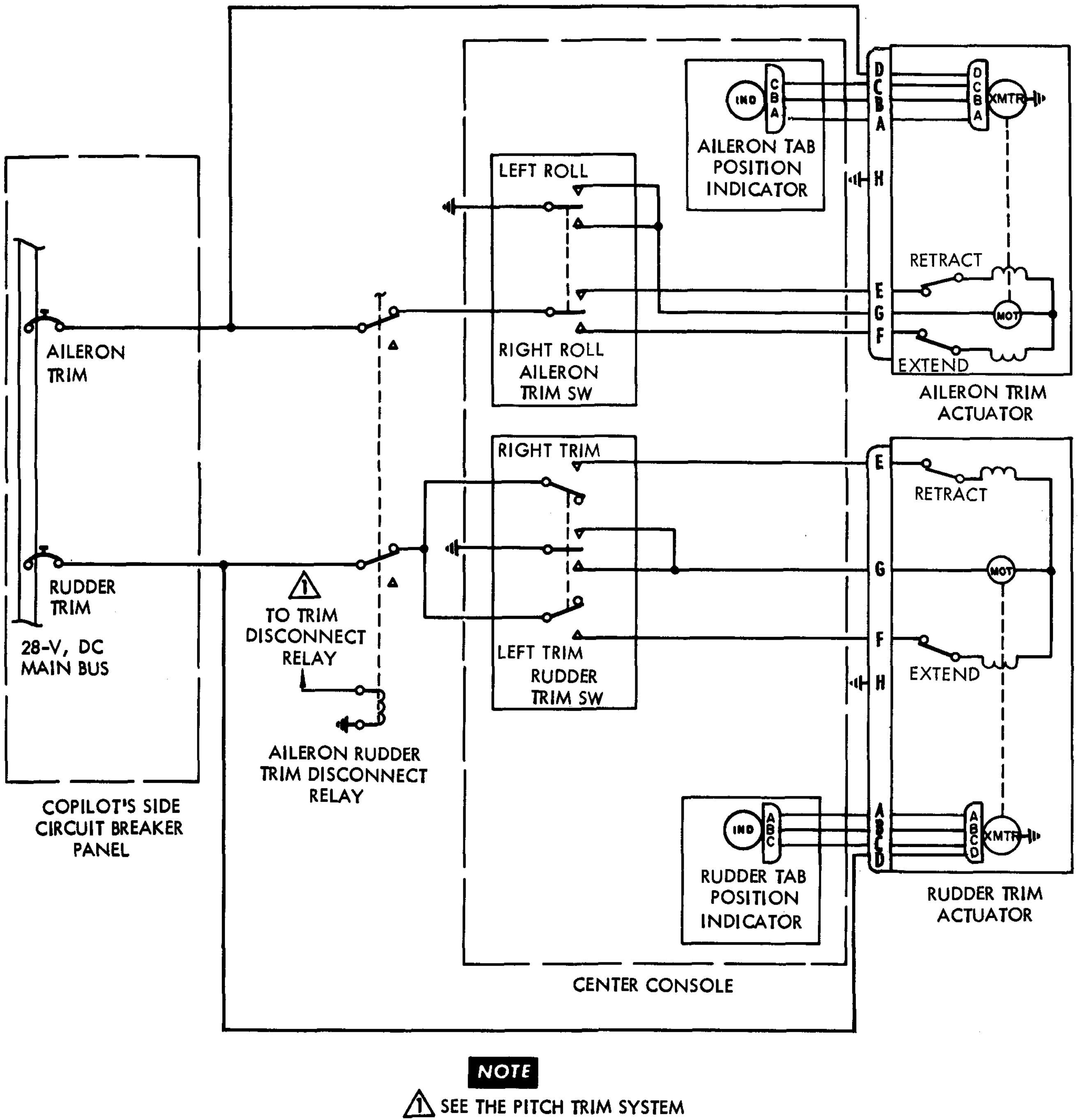
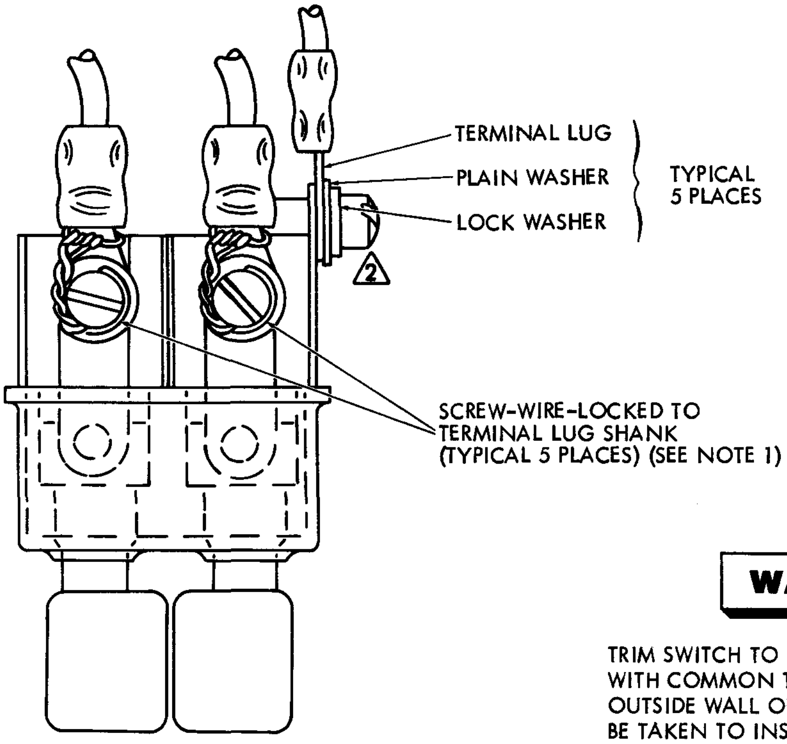


Figure 27-10. Aileron and Rudder Trim Systems Schematic Diagram


TERMINAL LUG
PLAIN WASHER
LOCK WASHER
TYPICAL 5 PLACES
2
SCREW-WIRE-LOCKED TO TERMINAL LUG SHANK (TYPICAL 5 PLACES) (SEE NOTE 1)

WARNING

TRIM SWITCH TO BE INSTALLED IN HOUSING WITH COMMON TERMINAL ADJACENT TO OUTSIDE WALL OF HOUSING - CARE SHALL BE TAKEN TO INSURE THAT FLANGE OF RUBBER BOOT IS SEATED IN RECESS IN BOTH HALVES OF HOUSING PRIOR TO INSTALLATION OF COMPLETE ASSEMBLY TO WHEEL

NOTE

1. USE .020 CRES. LOCKWIRE MS20995C20 OR EQUIV. INSTALLED AS SHOWN. WARNING DO NOT USE LONGER SCREW THAN THAT SUPPLIED OR INTERNAL SWITCH DAMAGE MAY OCCUR.

△2 WIRE LOCKING NOT SHOWN HERE FOR CLARITY

JS-2-2-2728

Figure 27-11. Pitch Trim Switch Assembly (Part No. JR003-3) Terminal Locking Instructions

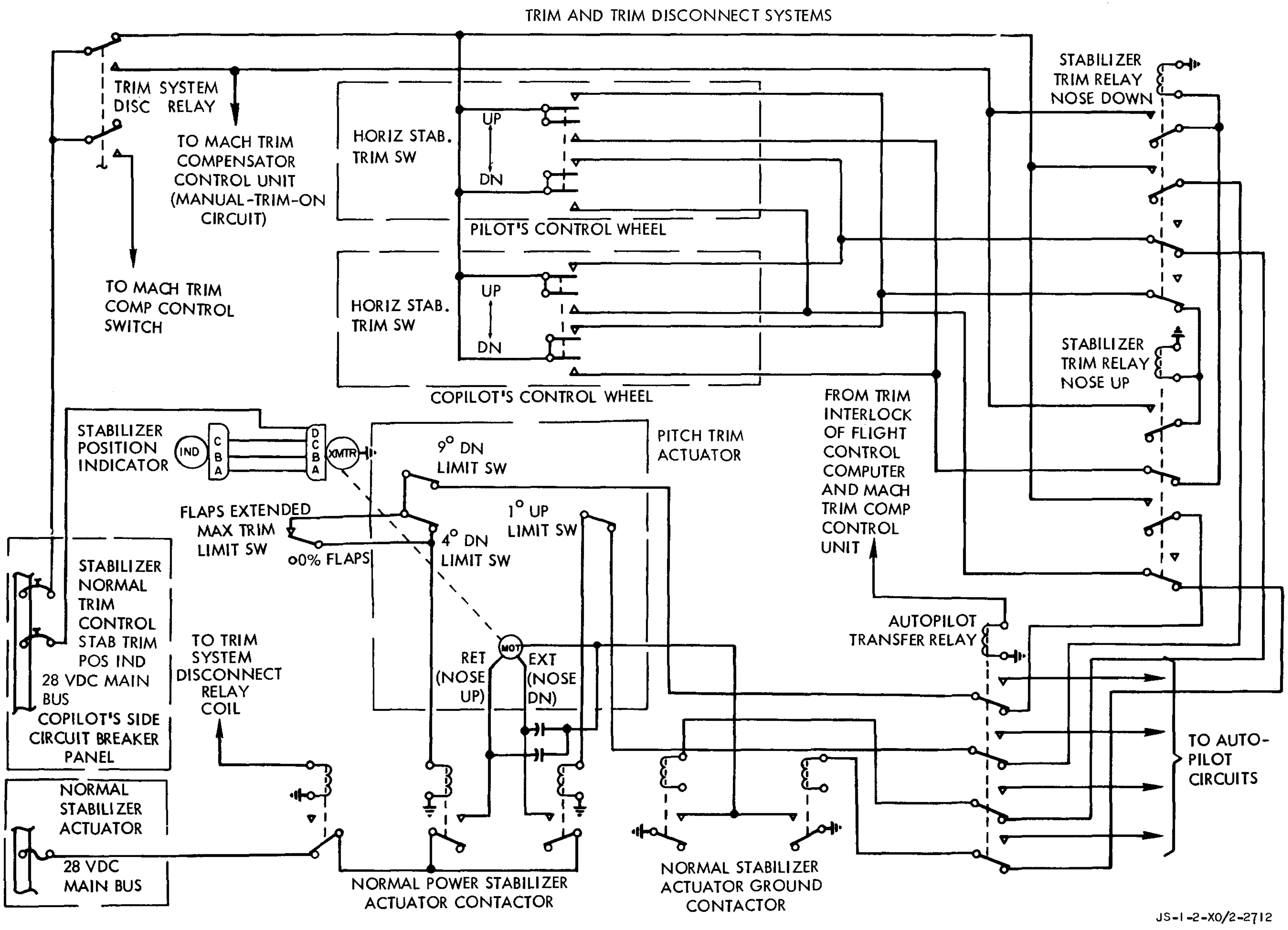

TRIM AND TRIM DISCONNECT SYSTEMS
TRIM SYSTEM DISC RELAY
TO MACH TRIM COMPENSATOR CONTROL UNIT (MANUAL-TRIM-ON CIRCUIT)
TO MACH TRIM COMP CONTROL SWITCH
HORIZ STAB. TRIM SW
UP
DN
PILOT'S CONTROL WHEEL
HORIZ STAB. TRIM SW
UP
DN
COPILOT'S CONTROL WHEEL
STABILIZER TRIM RELAY NOSE DOWN
STABILIZER TRIM RELAY NOSE UP
FROM TRIM INTERLOCK OF FLIGHT CONTROL COMPUTER AND MACH TRIM COMP CONTROL UNIT
STABILIZER POSITION INDICATOR
IND
C
B
A
D
C
B
A
XMTR
PITCH TRIM ACTUATOR
9° DN LIMIT SW
FLAPS EXTENDED MAX TRIM LIMIT SW
0% FLAPS
4° DN LIMIT SW
1° UP LIMIT SW
MOT
RET (NOSE UP)
EXT (NOSE DN)
STABILIZER NORMAL TRIM CONTROL
STAB TRIM POS IND
28 VDC MAIN BUS
COPILOT'S SIDE CIRCUIT BREAKER PANEL
TO TRIM SYSTEM DISCONNECT RELAY COIL
NORMAL STABILIZER ACTUATOR
28 VDC MAIN BUS
NORMAL POWER STABILIZER ACTUATOR CONTACTOR
NORMAL STABILIZER ACTUATOR GROUND CONTACTOR
AUTOPILOT TRANSFER RELAY
TO AUTO-PILOT CIRCUITS
JS-1-2-X0/2-2712

Figure 27-12. Pitch Trim System Schematic Diagram

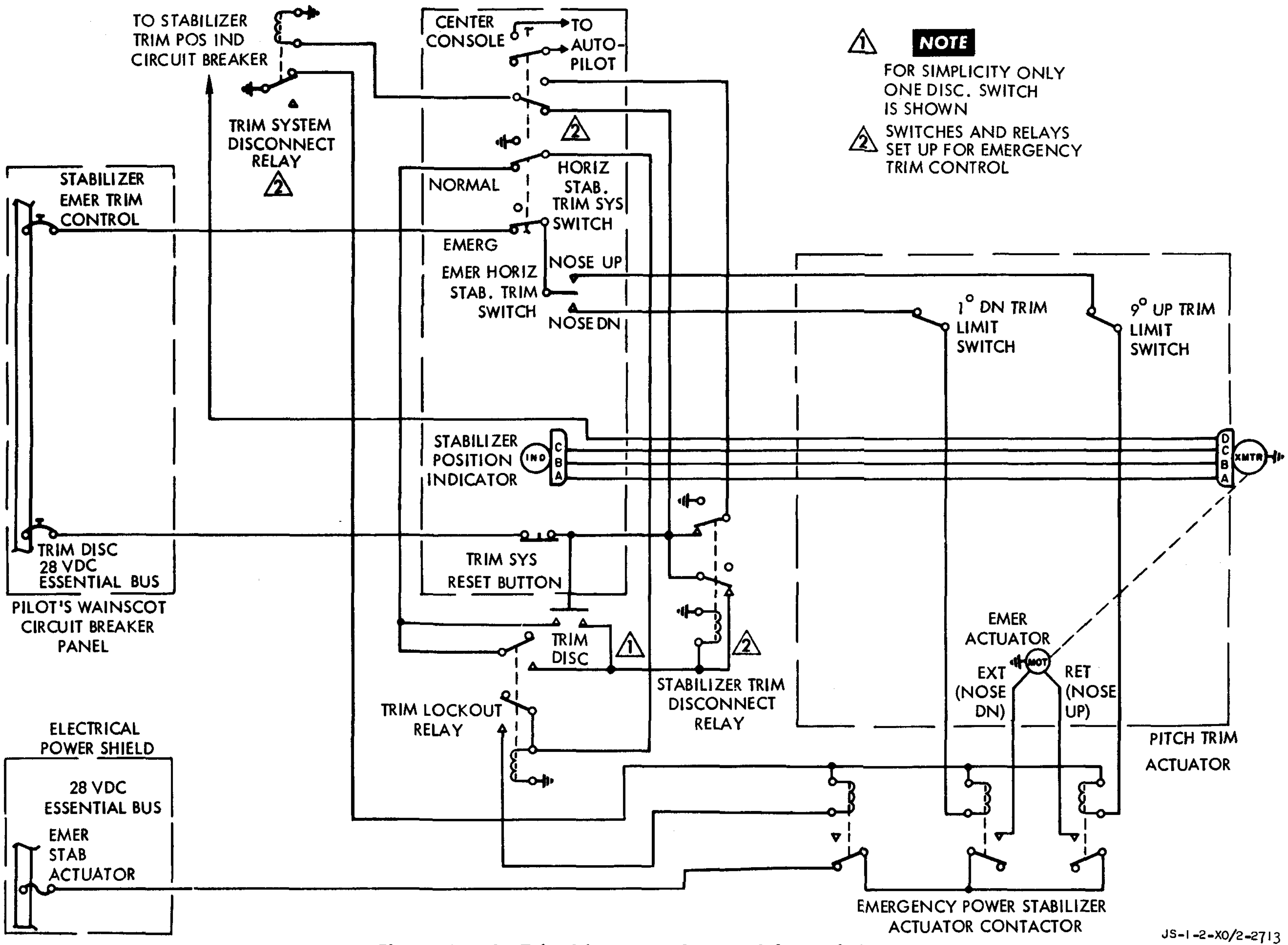
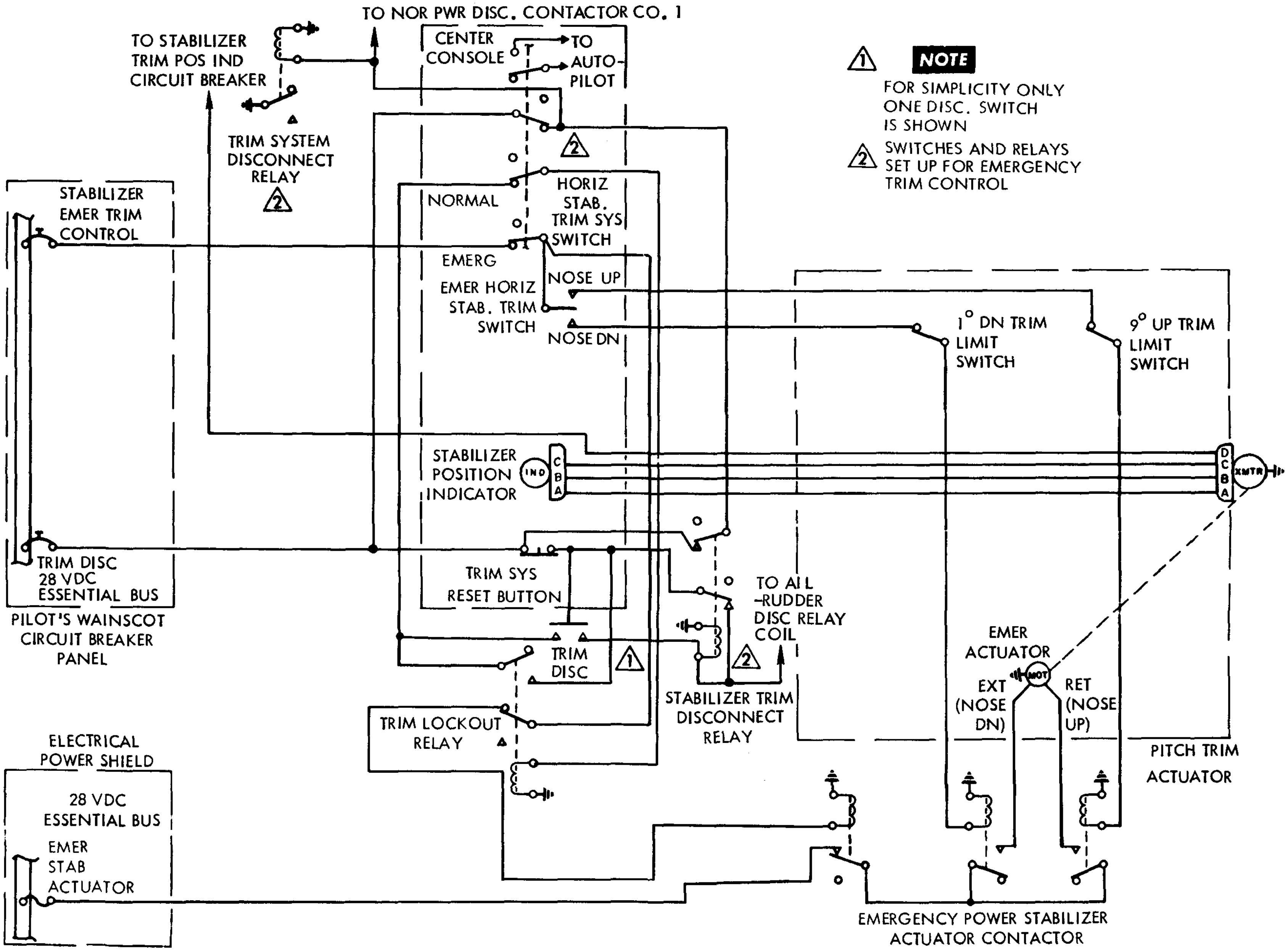


Figure 27-13. Trim Disconnect System Schematic Diagram
(Airplanes 5001 thru 5069)


TO NOR PWR DISC. CONTACTOR CO. 1
TO STABILIZER TRIM POS IND CIRCUIT BREAKER
TRIM SYSTEM DISCONNECT RELAY
2
CENTER CONSOLE
TO AUTO-PILOT
2
NORMAL
HORIZ STAB. TRIM SYS SWITCH
EMERG
EMER HORIZ STAB. TRIM SWITCH
NOSE UP
NOSE DN
STABILIZER EMER TRIM CONTROL
TRIM DISC 28 VDC ESSENTIAL BUS
PILOT'S WAINSCOT CIRCUIT BREAKER PANEL
1° DN TRIM LIMIT SWITCH
9° UP TRIM LIMIT SWITCH
STABILIZER POSITION INDICATOR
IND
C
B
A
D
C
B
A
XMTR
TRIM SYS RESET BUTTON
TRIM DISC
1
TO AIL-RUDDER DISC RELAY COIL
2
STABILIZER TRIM DISCONNECT RELAY
TRIM LOCKOUT RELAY
EMER ACTUATOR
MOT
EXT (NOSE DN)
RET (NOSE UP)
PITCH TRIM ACTUATOR
ELECTRICAL POWER SHIELD
28 VDC ESSENTIAL BUS
EMER STAB ACTUATOR
EMERGENCY POWER STABILIZER ACTUATOR CONTACTOR

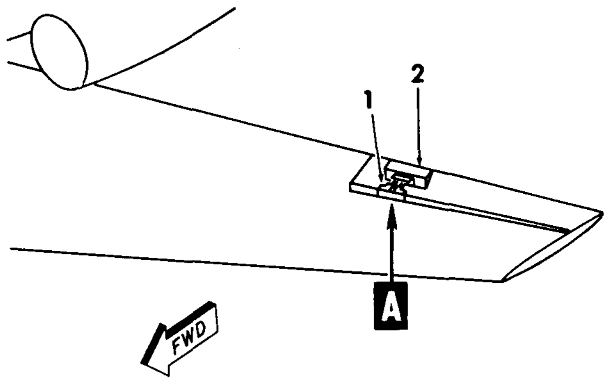
1 **NOTE**

FOR SIMPLICITY ONLY ONE DISC. SWITCH IS SHOWN

2 SWITCHES AND RELAYS SET UP FOR EMERGENCY TRIM CONTROL

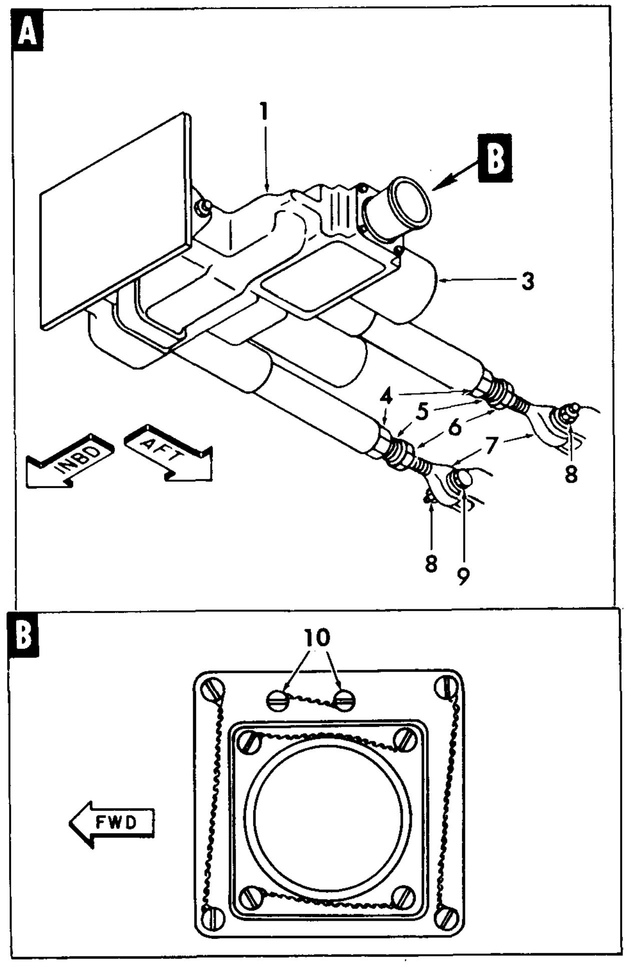
JS-1-2-X0/2-2713A

Figure 27-13A. Trim Disconnect System Schematic Diagram (Airplanes 5070 and Up)



NOTE

ADJUST THE ROD ENDS (7) AS FOLLOWS: LOOSEN NUTS (4) AND (6) AND USE A THIN OPEN-END WRENCH TO TURN THE SWIVELS (5).



STEP 1

LOOSEN THE NUTS (8) AND BOLTS (9) AT THE AFT END OF THE ACTUATOR (1), AND ADJUST THE ROD ENDS (7) TO RELIEVE THE PRELOAD ON THE ACTUATOR. CHECK THAT PRELOAD IS RELIEVED BY MOVING THE BOLTS (9) BACK AND FORTH. BOLTS SHOULD MOVE BACK AND FORTH FREELY.

STEP 2

PLACE THE AILERON TRIM SWITCH ON THE CENTER CONSOLE IN THE RIGHT-WING-UP POSITION UNTIL TAB ACTUATION STOPS. THE ACTUATOR SHOULD BE RETRACTED, AND THE TAB SHOULD BE DISPLACED 15 (± 1) DEGREES DOWN FROM THE NEUTRAL POSITION, OR 1.37 (± 0.09) INCHES AS MEASURED FROM THE INBOARD TRAILING EDGE OF THE TAB TO THE AILERON CONTROL SURFACE.

NOTE

"NEUTRAL POSITION" AS USED IN THESE PROCEDURES IS THE AILERON TAB RIG ANGLE AS RECORDED IN THE AIRCRAFT LOG.

STEP 3

PLACE THE AILERON TRIM SWITCH IN THE LEFT-WING-UP POSITION UNTIL TAB ACTUATION STOPS. THE ACTUATOR SHOULD BE EXTENDED, AND THE TAB SHOULD BE DISPLACED 15 (± 1) DEGREES UP FROM THE NEUTRAL POSITION, OR 1.37 (± 0.09) INCHES AS MEASURED FROM THE INBOARD TRAILING EDGE OF THE TAB TO THE AILERON CONTROL SURFACE. IF THE TAB DOES NOT TRAVEL THE REQUIRED AMOUNT AND THE ACTUATOR IS PROPERLY INSTALLED, ADJUST THE LIMIT SWITCH ADJUSTMENT SCREWS (10) OR ROD END (7) TO OBTAIN THE PROPER TAB TRAVEL.

NOTE

WITH THE TAB IN THE POSITION OF STEP 3, THE DISTANCE BETWEEN THE CENTERLINE OF THE ROD END ATTACH BOLTS (9) AND THE SHOULDERS OF THE SWIVELS (5) SHOULD NOT EXCEED 1.3 INCHES.

STEP 4

ACTUATE THE TRIM TAB (2) TO THE NEUTRAL POSITION AND OBSERVE THE AILERON TRIM INDICATOR. THE INDICATOR SHOULD INDICATE 0 (± 1) DEGREE. IF THE INDICATOR DOES NOT INDICATE THE ZERO TRIM POSITION, REMOVE THE FOUR MOUNTING SCREWS FROM THE POSITION TRANSMITTER COVER (3), LIFT THE COVER ENOUGH TO UNLOCK THE TRANSMITTER FLANGE, AND TURN THE FLANGE TO NULL THE INDICATOR. REPLACE THE COVER (3) AND SAFETY THE MOUNTING SCREWS.

STEP 5

PLACE THE AILERON TRIM SWITCH IN THE RIGHT-WING-UP POSITION UNTIL TAB ACTUATION STOPS. THE POSITION INDICATOR SHOULD INDICATE 15 (± 2) DEGREES RIGHT WING UP, AND THE TAB SURFACE SHOULD BE DOWN.

STEP 6

PLACE THE AILERON TRIM SWITCH IN THE LEFT-WING-UP POSITION UNTIL TAB ACTUATION STOPS. THE POSITION INDICATOR SHOULD INDICATE 15 (± 2) DEGREES LEFT WING UP, AND THE TAB SURFACE SHOULD BE UP

STEP 7

ACTUATE THE TAB TO THE NEUTRAL POSITION, AND CHECK THAT THE BOLTS (9) MOVE FREELY BACK AND FORTH. ADJUST THE ROD ENDS (7), IF NECESSARY TO OBTAIN THIS CONDITION.

NOTE

PERFORM STEP 8 ON AIRPLANES WITH JC1000-10 ACTUATORS ONLY.

STEP 8

PRELOAD THE ACTUATOR AS FOLLOWS: ADJUST EACH ROD END (7) 1/4 TO 5/16 TURN IN OPPOSITE DIRECTIONS OR, ADJUST ONE ROD END 1/2 TO 5/8 TURN IN EITHER DIRECTION. IF POSSIBLE, USE THE PRELOAD ADJUSTMENTS TO OFFSET ANY ADJUSTMENTS MADE IN STEP 7.

NOTE

AT NO TIME SHOULD THE DISTANCE BETWEEN THE CENTERLINE OF THE ROD END ATTACH BOLTS (9) AND THE SHOULDERS OF THE SWIVELS (5) EXCEED 1.45 INCHES.

STEP 9

CHECK THAT ALL CONNECTIONS, BOLTS AND NUTS, AND ADJUSTMENT SCREWS ARE SECURE AND SAFETIED; THEN PERFORM AN OPERATIONAL CHECKOUT OF THE SYSTEM.

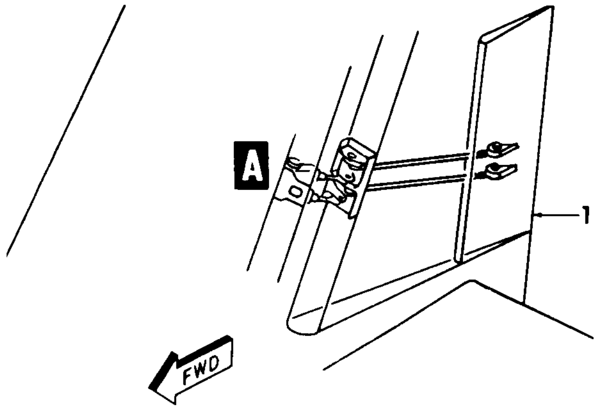
NOTE

AIRPLANE SERIALS 5042 AND 5046 AND UP HAVE A FIXED TRIM TAB ON THE RIGHT AILERON IN ADDITION TO THE OPERABLE TAB ON THE LEFT AILERON. ON THESE AIRCRAFT, THE NEUTRAL POSITION OF THE LEFT TAB IS THE FAIRED POSITION. ALL TRIM RIGGING IS DONE ON THE FIXED RIGHT TAB. THE RIG ANGLE OF THIS TAB IS RECORDED IN THE AIRCRAFT LOG. ADJUSTMENTS TO FIXED TAB ARE MADE AS FOLLOWS:

1. REMOVE THE BOLT AT FWD. END OF THE LINK AND LOWER TRUNNION FITTING OUT OF AILERON FITTING.
2. LOOSEN JAM NUT ON ROD END FITTING AND ROTATE TRUNNION FITTING AS REQUIRED.
3. ROTATE ONE LINK AND ADDITIONAL 1/2 TO 1 TURN AFTER CORRECT POSITION IS ESTABLISHED TO CREATE A SLIGHT PRELOAD.

JS-2-2-2722

Figure 27-14. Aileron Trim Actuator Adjustment

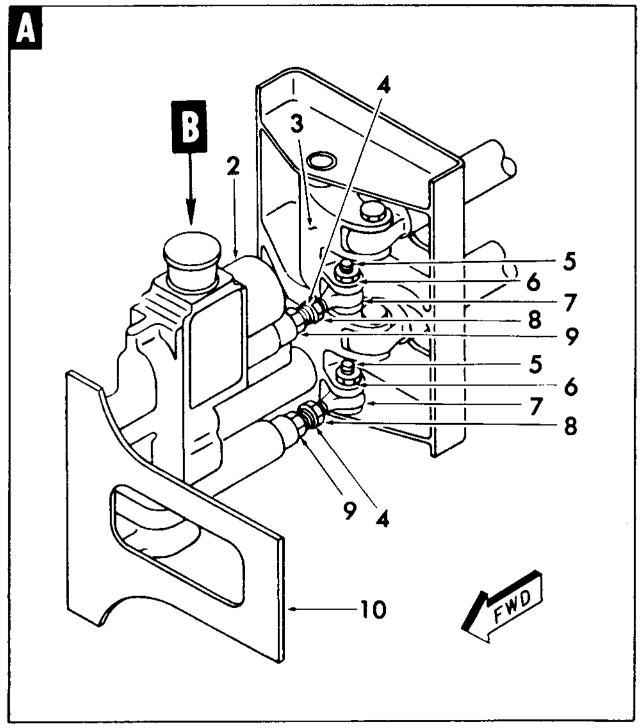


NOTE

TO RIG A RUDDER TRIM SYSTEM WITH THE ACTUATOR ALREADY INSTALLED, OMIT STEPS (1) AND (2).

STEP 1

FULLY RETRACT THE ACTUATOR JACKSCREWS BEFORE MOUNTING THE ACTUATOR TO THE FIN FITTING (10).



STEP 2

POSITION THE ACTUATOR JACKSCREWS AT THE APPROXIMATE MID-STROKE POSITION AND INSTALL THE BOLTS (5) AND NUTS (6) THROUGH THE BELLCRANK (3) AND THE ROD ENDS (7) AS SHOWN IN VIEW "A". BOLTS (5) SHOULD BE FREE TO MOVE UP AND DOWN INDICATING NO PRELOAD.

STEP 3

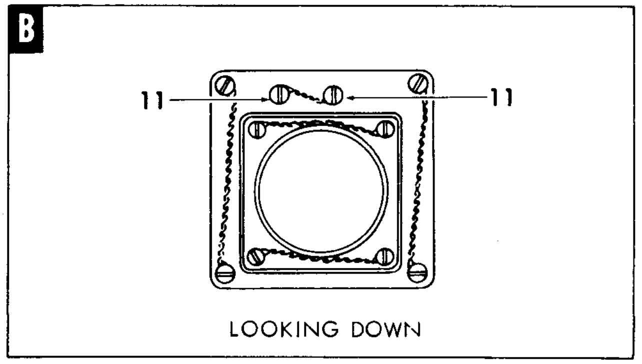
WITH NO PRELOAD ON THE ACTUATOR, PLACE THE RUDDER TRIM KNOB ON THE CENTER CONSOLE IN THE LEFT-TRIM POSITION UNTIL TAB ACTUATION STOPS. WITH THE ACTUATOR IN THIS POSITION, THE TAB (1) SHOULD BE DEFLECTED 1.27 (± 0.07) INCHES (9 3/4) DEGREES) TO THE RIGHT OF THE FAIRED RUDDER. IF NECESSARY, ADJUST THE ROD ENDS (7) TO GIVE THIS DEFLECTION. A MINIMUM OF 0.46 INCHES OF THE THREAD ENGAGEMENT SHALL BE MAINTAINED.

STEP 4

PLACE THE RUDDER TRIM KNOB IN THE RIGHT-TRIM POSITION UNTIL TAB ACTUATION STOPS. WITH THE ACTUATOR IN THIS POSITION, THE TAB (1) SHOULD BE DEFLECTED 1.27 (± 0.07) INCHES (9 3/4 DEGREES) TO THE LEFT OF THE FAIRED RUDDER. IF NECESSARY, ADJUST THE ROD ENDS (7) TO OBTAIN THIS DEFLECTION.

NOTE

MEASURE THE DEFLECTION IN INCHES BY PLACING A STRAIGHT EDGE ALONG RUDDER CHORD AT THE TOP OF THE TRIM TAB AND READING THE OFFSET AT THE TRAILING EDGE. ONE DEGREE EQUALS 0.13 INCHES.



NOTE

IF THE TAB DOES NOT TRAVEL THE REQUIRED AMOUNT, AND THE ACTUATOR IS PROPERLY INSTALLED, ADJUST THE LIMIT SWITCH ADJUSTMENT SCREWS (11) TO OBTAIN THE PROPER TAB TRAVEL.

STEP 5

POSITION THE TAB (1) AT THE RUDDER TAB RIG POSITION AS RECORDED IN THE AIRCRAFT LOG, THIS POSITION MUST NOT EXCEED 0.26 INCHES (2 DEGREES) FROM THE FAIRED POSITION. WITH THE TAB IN THIS POSITION, THE RUDDER TRIM INDICATOR SHOULD INDICATE 0 (± 0.5) DEGREE. TO OBTAIN THIS INDICATION, ADJUST THE POSITION TRANSMITTER AS FOLLOWS: REMOVE THE FOUR MOUNTING SCREWS FROM THE TRANSMITTER COVER (2), LIFT THE COVER TO UNLOCK THE TRANSMITTER FLANGE, AND TURN THE FLANGE AS REQUIRED. REPLACE THE COVER (2) AFTER ADJUSTING THE TRANSMITTER.

NOTE

TOTAL TAB TRAVEL WILL STILL BE 19 1/2 DEGREES (2.54 INCHES REF). BUT DUE TO PRE-SET, MAXIMUM TRAVEL AS INDICATED ON POSITION INDICATOR, WILL NOT BE SYMMETRICAL LEFT AND RIGHT. MAXIMUM TRAVEL CAN VARY FROM 7 3/4 (± 1) DEGREE TO 11 3/4 (± 1) DEGREE ON LEFT OR RIGHT DEPENDING ON AMOUNT AND DIRECTION OF PRESET.

STEP 6

ACTUATE THE TAB TO FAIR WITH THE RUDDER AND CHECK THAT THE BOLTS (5) STILL MOVE UP AND DOWN WITH NO BINDING.

STEP 7

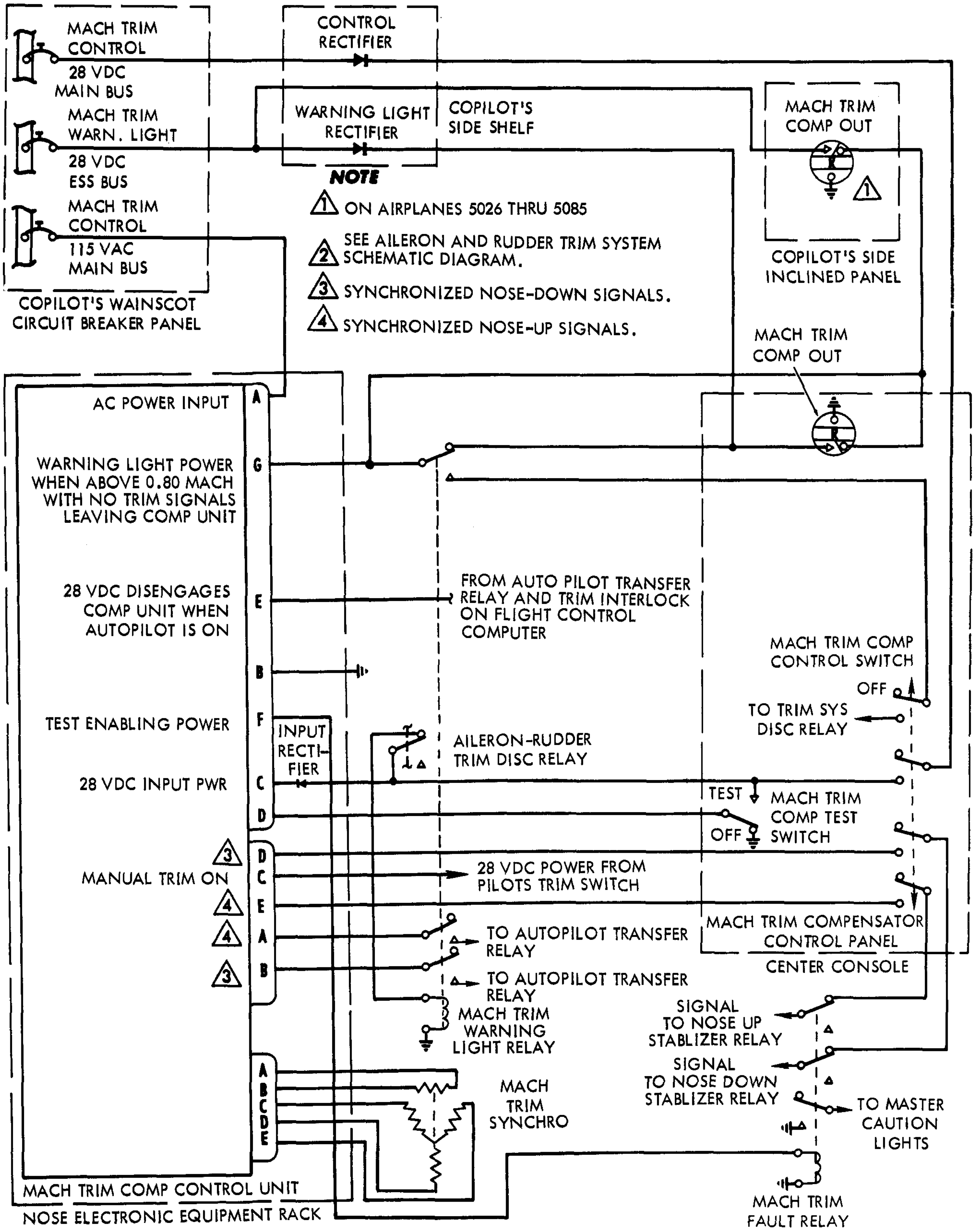
TIGHTEN AND SAFETY ALL NUTS AND BOLTS AS REQUIRED. DO NOT EXCEED 5 INCH-POUNDS TORQUE ON BOLTS (5).

STEP 8

RETURN THE RUDDER TRIM SYSTEM TO NORMAL AND PERFORM AN OPERATIONAL CHECKOUT OF THE SYSTEM.

JS-1-2-X0/2-2715

Figure 27-15. Rudder Trim Actuator Adjustment


MACH TRIM CONTROL
28 VDC MAIN BUS
MACH TRIM WARN. LIGHT
28 VDC ESS BUS
MACH TRIM CONTROL
115 VAC MAIN BUS
COPILOT'S WAINSCOT CIRCUIT BREAKER PANEL
CONTROL RECTIFIER
WARNING LIGHT RECTIFIER
COPILOT'S SIDE SHELF
MACH TRIM COMP OUT
1
COPILOT'S SIDE INCLINED PANEL
MACH TRIM COMP OUT
AC POWER INPUT
A
WARNING LIGHT POWER WHEN ABOVE 0.80 MACH WITH NO TRIM SIGNALS LEAVING COMP UNIT
G
28 VDC DISENGAGES COMP UNIT WHEN AUTOPILOT IS ON
E
FROM AUTO PILOT TRANSFER RELAY AND TRIM INTERLOCK ON FLIGHT CONTROL COMPUTER
B
TEST ENABLING POWER
F
INPUT RECTI-FIER
AILERON-RUDDER TRIM DISC RELAY
28 VDC INPUT PWR
C
D
MACH TRIM COMP CONTROL SWITCH
OFF
TO TRIM SYS DISC RELAY
TEST
OFF
MACH TRIM COMP TEST SWITCH
3
MANUAL TRIM ON
4
4
3
D
C
E
A
B
28 VDC POWER FROM PILOTS TRIM SWITCH
TO AUTOPILOT TRANSFER RELAY
TO AUTOPILOT TRANSFER RELAY
MACH TRIM WARNING LIGHT RELAY
MACH TRIM COMPENSATOR CONTROL PANEL
CENTER CONSOLE
SIGNAL TO NOSE UP STABLIZER RELAY
SIGNAL TO NOSE DOWN STABLIZER RELAY
TO MASTER CAUTION LIGHTS
A
B
C
D
E
MACH TRIM SYNCHRO
MACH TRIM FAULT RELAY
MACH TRIM COMP CONTROL UNIT
NOSE ELECTRONIC EQUIPMENT RACK

NOTE

1 ON AIRPLANES 5026 THRU 5085

2 SEE AILERON AND RUDDER TRIM SYSTEM SCHEMATIC DIAGRAM.

3 SYNCHRONIZED NOSE-DOWN SIGNALS.

4 SYNCHRONIZED NOSE-UP SIGNALS.

Figure 27-16. Mach Trim Compensator Schematic Diagram (Airplanes 5001 thru 5085)

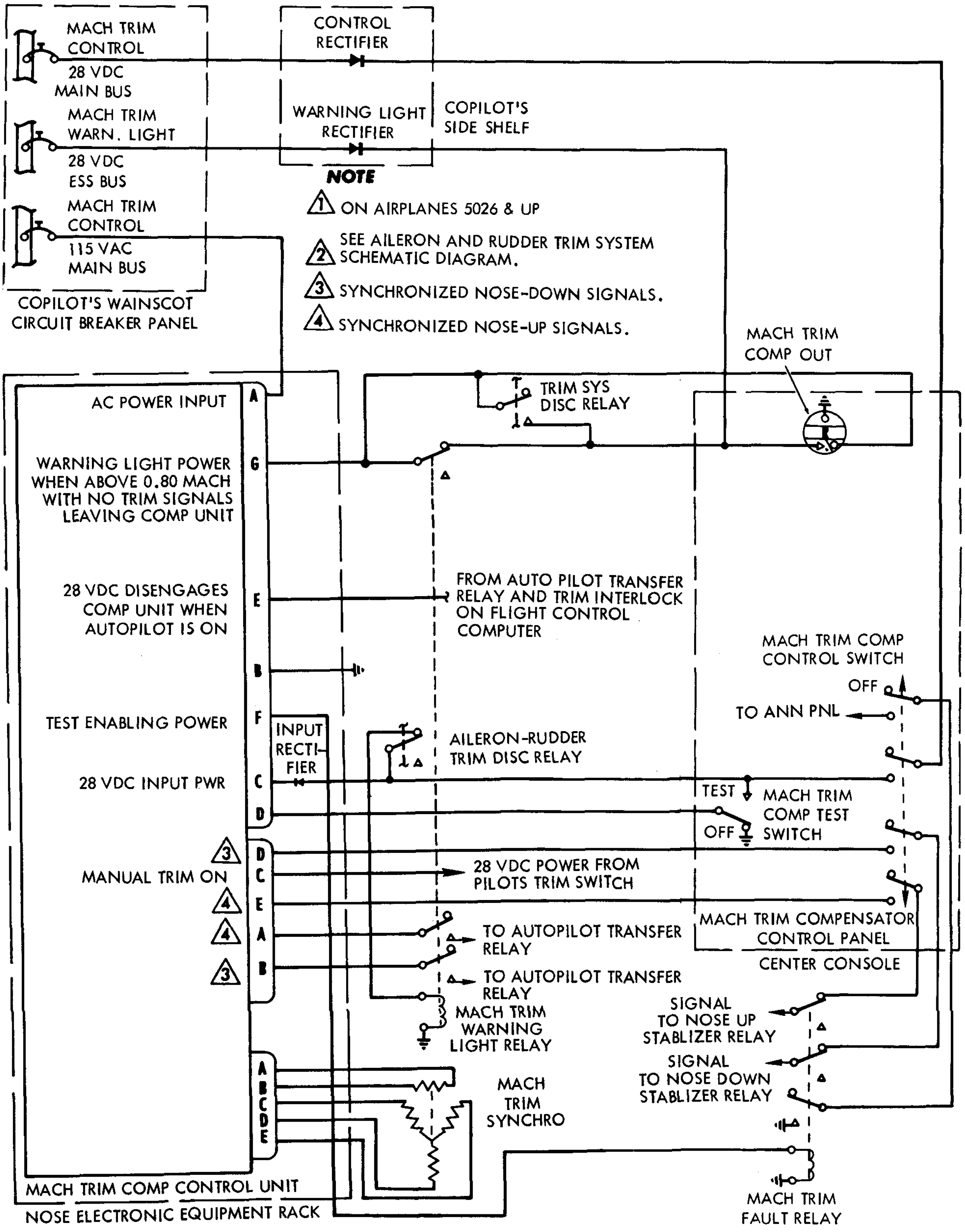
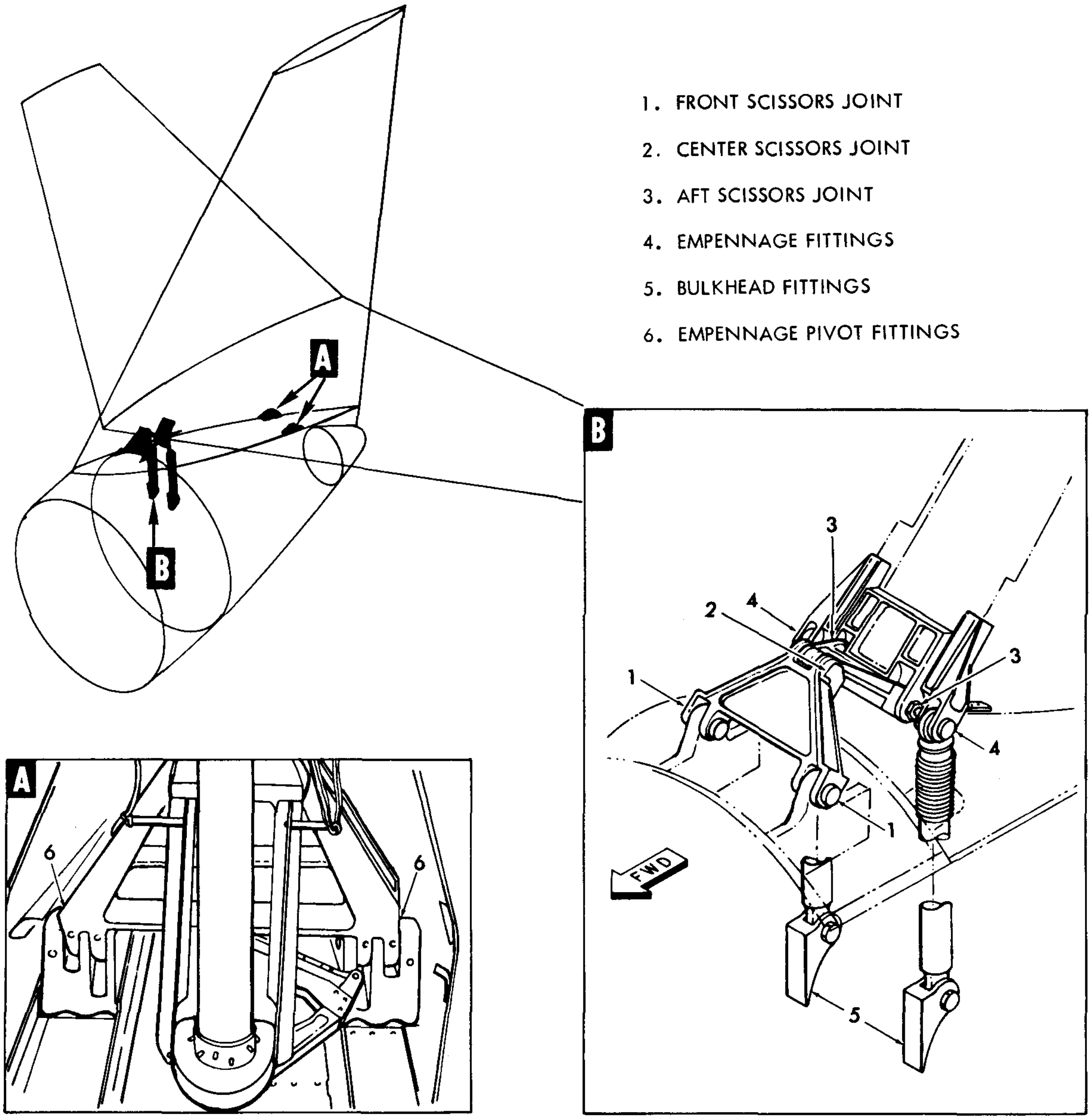

MACH TRIM CONTROL
28 VDC MAIN BUS
MACH TRIM WARN. LIGHT
28 VDC ESS BUS
MACH TRIM CONTROL
115 VAC MAIN BUS
COPILOT'S WAINSCOT CIRCUIT BREAKER PANEL
CONTROL RECTIFIER
WARNING LIGHT RECTIFIER
COPILOT'S SIDE SHELF
NOTE
1 ON AIRPLANES 5026 & UP
2 SEE AILERON AND RUDDER TRIM SYSTEM SCHEMATIC DIAGRAM.
3 SYNCHRONIZED NOSE-DOWN SIGNALS.
4 SYNCHRONIZED NOSE-UP SIGNALS.
MACH TRIM COMP OUT
TRIM SYS DISC RELAY
AC POWER INPUT
A
WARNING LIGHT POWER WHEN ABOVE 0.80 MACH WITH NO TRIM SIGNALS LEAVING COMP UNIT
G
28 VDC DISENGAGES COMP UNIT WHEN AUTOPILOT IS ON
E
FROM AUTO PILOT TRANSFER RELAY AND TRIM INTERLOCK ON FLIGHT CONTROL COMPUTER
B
MACH TRIM COMP CONTROL SWITCH
OFF
TO ANN PNL
TEST ENABLING POWER
F
INPUT RECTI-FIER
AILERON-RUDDER TRIM DISC RELAY
28 VDC INPUT PWR
C
TEST
MACH TRIM COMP TEST SWITCH
D
OFF
MANUAL TRIM ON
3 D
C
28 VDC POWER FROM PILOTS TRIM SWITCH
4 E
4 A
TO AUTOPILOT TRANSFER RELAY
MACH TRIM COMPENSATOR CONTROL PANEL
3 B
TO AUTOPILOT TRANSFER RELAY
CENTER CONSOLE
MACH TRIM WARNING LIGHT RELAY
SIGNAL TO NOSE UP STABLIZER RELAY
SIGNAL TO NOSE DOWN STABLIZER RELAY
A
B
C
D
E
MACH TRIM SYNCHRO
MACH TRIM COMP CONTROL UNIT
NOSE ELECTRONIC EQUIPMENT RACK
MACH TRIM FAULT RELAY

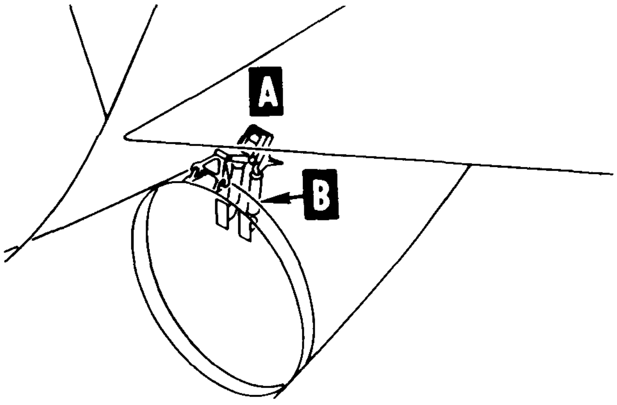
Figure 27-16A. Mach Trim Compensator Schematic Diagram (Airplanes 5086 and Up)

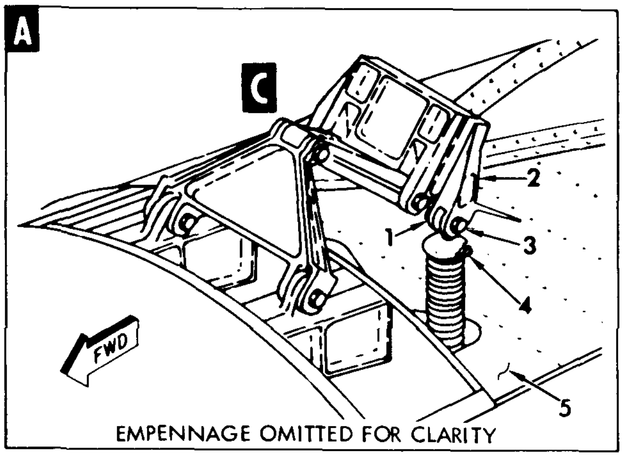


1. FRONT SCISSORS JOINT
2. CENTER SCISSORS JOINT
3. AFT SCISSORS JOINT
4. EMPENNAGE FITTINGS
5. BULKHEAD FITTINGS
6. EMPENNAGE PIVOT FITTINGS

(LOOKING FORWARD FROM REAR OF EMPENNAGE)

Figure 27-17. Pitch Trim Actuator-and-Empennage Attach Points





REMOVAL

STEP 1

POSITION THE EMPENNAGE SO THAT THE ACTUATOR LENGTH AS MEASURED CENTER TO CENTER BETWEEN ROD ENDS IS 25 ± 0.10 INCHES. COCKPIT INDICATOR SHOULD READ 3.6 TO 3.8° NOSE UP.

STEP 2

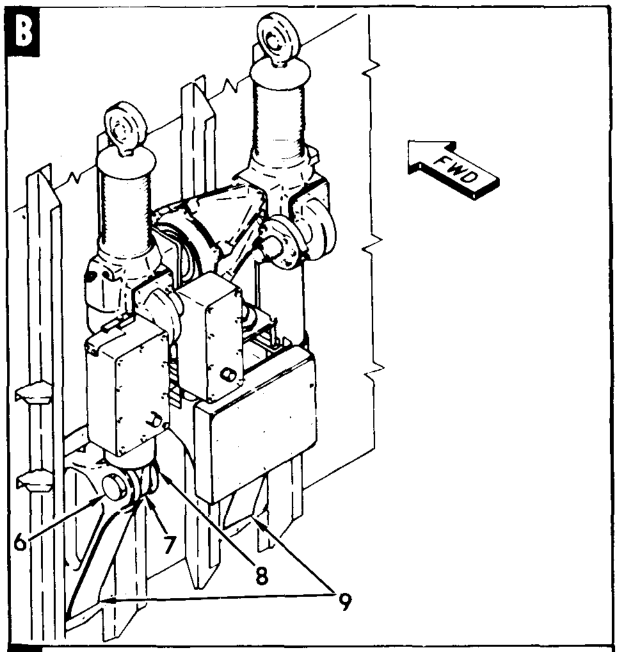
INSTALL A 11078-MEU FIXTURE AS SHOWN IN VIEW "C".

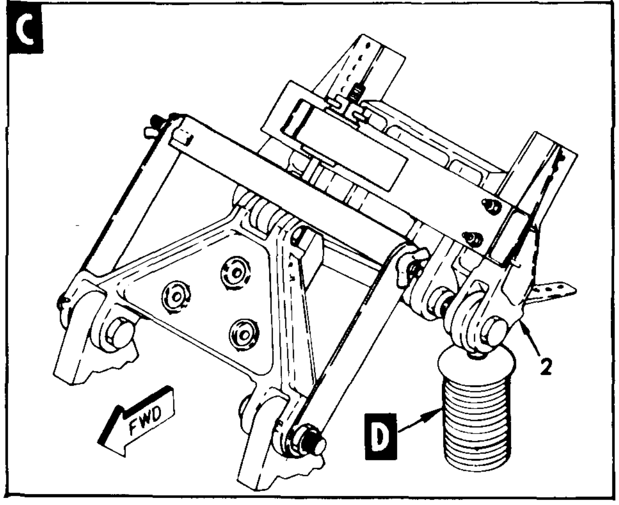
CAUTION

FAILURE TO INSTALL ABOVE FIXTURE WILL ALLOW THE EMPENNAGE TO TILT BACKWARD AND CAUSE STRUCTURAL DAMAGE.

CAUTION

DO NOT SEPARATE THE ACTUATOR HALVES OR UNIT WILL GET OUT OF TIME. IF THIS IS DONE, THE UNIT MUST BE RETIMED PRIOR TO REINSTALLATION. (SEE PARAGRAPH 27.3.5.3).





STEP 3

TAPE UP THE OIL VENT HOLES (11) TO PREVENT SPILLING WHILE HANDLING. THERE ARE TWO OIL VENT HOLES IN EACH UPPER ROD END (10).

CAUTION

THE OIL USED IN THE ACTUATOR IS HARMFUL TO PAINTED SURFACES.

STEP 4

DISCONNECT ALL ELECTRICAL WIRING FROM THE ACTUATOR AND STOW IT OUT OF THE WAY.

STEP 5

REMOVE COTTER PINS, NUTS (8), WASHERS, AND BOLTS (6) AT THE LOWER ROD END BEARINGS.

JS-1-2-X0/2-2718-1

Figure 27-18. Removal and Replacement of the Pitch Trim Actuator Assembly (Sheet 1 of 4)

STEP 6

WHILE SUPPORTING THE ACTUATOR WEIGHT (APPROXIMATELY 100 POUNDS), REMOVE THE COTTER PINS, NUT (1), AND BOLTS (3), THEN LOWER THE ACTUATOR, PASS IT FORWARD THROUGH THE LOWER PART OF THE FUSELAGE BULKHEAD ON WHICH IT IS MOUNTED AND THEN OUT THROUGH THE SPEED BRAKE OPENING.

INSTALLATION

STEP 7

CHECK TO SEE THAT THE ACTUATOR LENGTH IS SET 25 ± 0.10 INCHES (THIS IS CONSIDERED THE OPTIMUM LENGTH FOR INSTALLATION AND REMOVAL. ALL UNITS ARE LEFT AT THIS LENGTH AFTER TESTING AT THE AIRCRAFT MANUFACTURER'S PLANT).

IF NECESSARY TO RUN THE UNIT BEFORE INSTALLATION, THE FOLLOWING PRECAUTIONS SHOULD BE OBSERVED.

1. INSTALL A JUMPER WIRE FROM THE EMERGENCY MOTOR GROUND WIRE TO THE PRIMARY MOTOR GROUND WIRE.
2. DO NOT RUN THE UNIT INTO THE MECHANICAL STOPS WITHOUT SUITABLE OVERLOAD PROTECTION FOR THE MOTOR. FAILURE TO OBSERVE THESE CAUTIONS CAN RESULT IN DAMAGE TO THE MOTOR AND/OR CLUTCH.

STEP 8

TAPE UP THE OIL VENT HOLES (11) ON UNIT TO BE INSTALLED. OBSERVE CAUTION IN STEP 3.

STEP 9

PASS THE ACTUATOR THROUGH THE SPEED BRAKE OPENING, THEN AFT THROUGH THE LOWER SIDE OF THE BULKHEAD ON WHICH IT IS MOUNTED.

STEP 10

PASS THE UPPER ENDS THROUGH THE SHELF (5). HOLD THE ACTUATOR IN PLACE AND INSTALL LOWER ROD END BOLTS (6), WASHERS, AND NUTS (8) FINGER TIGHT. (FINGER TIGHT MEANS THREADS FULLY ENGAGED READY TO RECEIVE COTTER PIN WITH MINIMUM TORQUE.

STEP 11

CONNECT THE LEFT JACKSCREW TO THE EMPENNAGE FITTING (2) BY INSTALLING ONE BOLT (3), WASHER, AND NUT (1) FINGER TIGHT.

CAUTION

CHECK AND ASSURE THE SECURITY OF THE LH JACKSCREW ROD END BEFORE MAKING THE FOLLOWING ADJUSTMENTS IN THE RH ROD END. IF EITHER ROD END IS NOT SECURE, THE ADJUSTMENTS MADE WILL NOT BE MAINTAINED.

STEP 12

ADJUST THE RIGHT JACKSCREW LENGTH AS REQUIRED TO INSTALL THE OTHER BOLT (3), WASHER, AND NUT (1) FINGER TIGHT. IN CASE THIS BOLT CAN BE INSTALLED WITHOUT LENGTH ADJUSTMENT OF THE JACKSCREW, SKIP OVER TO STEP 15.

JACKSCREW LENGTH ADJUSTMENT

STEP 13

COARSE ADJUSTMENT OF THE ROD END IS OBTAINED BY ROTATING THE ROD END (10) ONE-HALF TURN FOR EACH 0.025 INCH OF ADJUSTMENT. THIS ROD END IS FREED BY LOOSENING THE CLAMP (4) AT THE UPPER END OF THE BOOT TO EXPOSE THE UPPER END OF THE JACKSCREW.

CAUTION

DO NOT ALLOW FOREIGN MATTER TO GET INTO THE EXPOSED JACKSCREW.

REMOVE COTTER PIN (17). TURN THE LOCKNUT (12) COUNTERCLOCKWISE (VIEWED FROM TOP) TO FREE THE RADIAL SERRATIONS ON THE WASHER (14) FROM THOSE ON THE TOP FLANGE OF THE NUT (15) SO THAT THE ROD END (10) MAY BE TURNED. IT IS NECESSARY TO REMOVE THE BOLTS (3) FROM BOTH UPPER ROD ENDS AND TILT THE ACTUATOR AWAY FROM THE EMPENNAGE FITTING TO TURN THE ROD END.

NOTE

WHILE TURNING THE ROD END, HOLD LOCK (13) IN POSITION TO ENGAGE SPLINES ON BOTH NUT (15) AND SCREW (16). OTHERWISE THE NUT MAY TURN AND CAUSE CLEARANCE ON BOTTOM SIDE OF NUT (15) FLANGE AND SCREW (16) TO CLOSE UP. SEE CAUTION BELOW.

CAUTION

AFTER THIS REINSTALL BOLTS (3), WASHER, AND NUTS (1) FINGER TIGHT. CONTINUE WITH FINE ROD END ADJUSTMENT IN NEXT STEP.

STEP 14

FOR FINE ADJUSTMENT OF THE ROD END, TURN THE LOCKNUT (12) COUNTERCLOCKWISE UNTIL THE INTERNAL SPLINES OF THE LOCK (13) NO LONGER ENGAGE THE SPLINES ON THE SCREW (16) BUT CONTINUE TO ENGAGE THE SPLINES OF THE NUT (15). IN THIS CONFIGURATION, THE LOCK (13) IS USED AS A TOOL TO TURN THE NUT (15).

JS-1-2-XO/3-2718-2

Figure 27-18. Removal and Replacement of the Pitch Trim Actuator Assembly (Sheet 2 of 4)

CAUTION

CLEARANCE BETWEEN THE BOTTOM SIDE OF THE FLANGE ON THE NUT (15) AND THE TOP OF THE SCREW (16) IS LIMITED. IF THE NUT BOTTOMS ON THE SCREW, TURN THE NUT (15) CLOCKWISE TO INCREASE CLEARANCE. IT MAY BE NECESSARY TO OFFSET THIS ADJUSTMENT BY TURNING THE ROD END (10) ONE-HALF TURN CLOCKWISE.

PRELOAD ADJUSTMENT

STEP 15

REMOVE TAPE FROM OIL VENT HOLES. REMOVE 11078-MEU FIXTURE. CONNECT ALL ELECTRICAL WIRING. INSURE THAT ALL ROD END BOLTS ARE INSTALLED WITH NUTS FINGER TIGHT (AS DEFINED ABOVE). COTTER PINS MAY BE INSTALLED BUT NOT REQUIRED AT THIS POINT. PRELOADING IS ACCOMPLISHED BY LENGTHENING PER STEP 14, THE RIGHT HAND JACKSCREW TO PUT IN A COMPRESSION LOAD OF ABOUT 400 POUNDS AFTER MAKING SURE THAT THERE IS NO EXISTING PRELOAD.

STEP 16

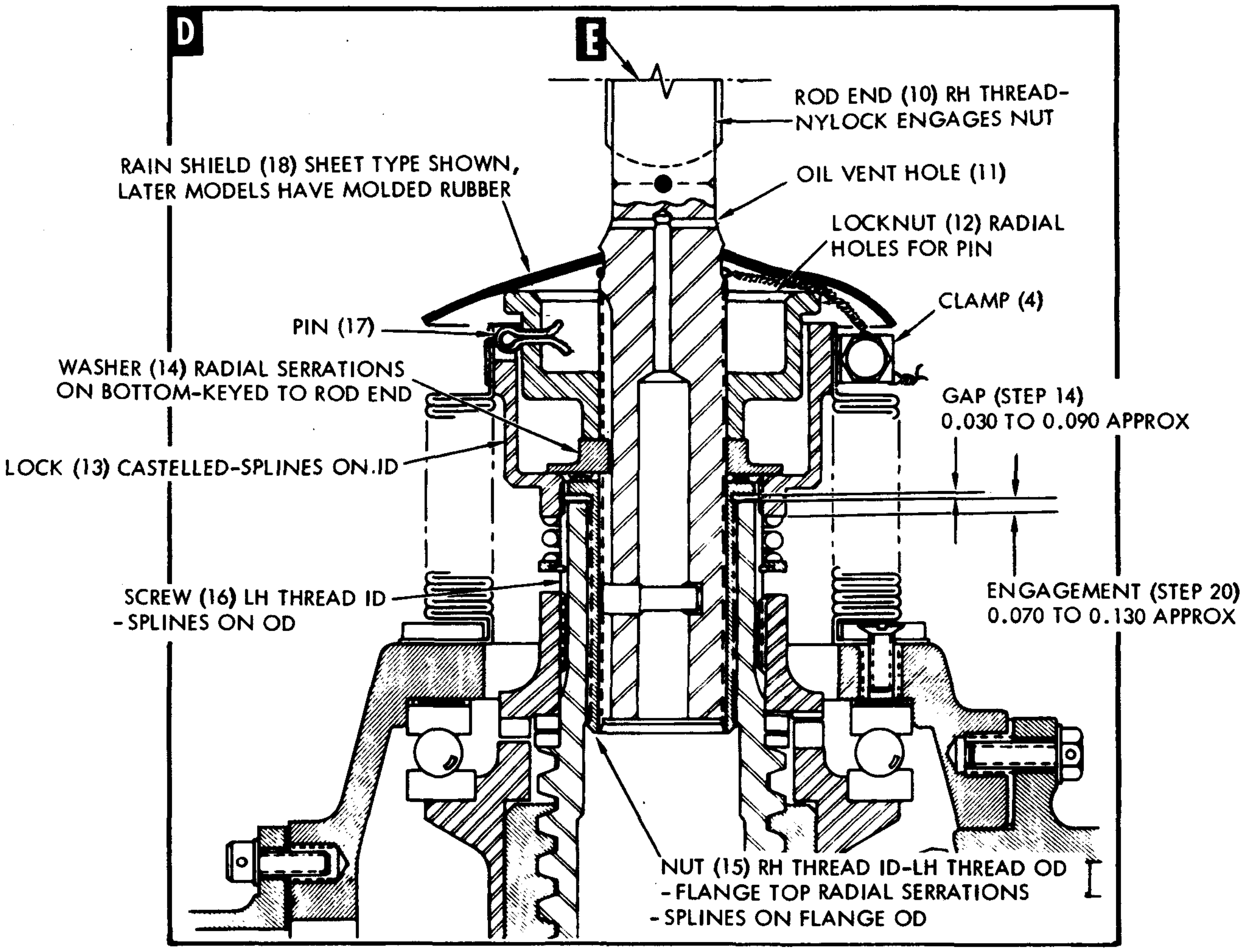
ON THE RIGHT HAND JACKSCREW, LOCATE THE POSITION AT WHICH THE LOCK (13) AND THE NUT (15) MAY BE ROTATED SLIGHTLY IN EITHER DIRECTION BY HAND TO ARRIVE AT POSITION OF ZERO AXIAL LOAD IN THE JACKSCREW. IF THERE IS ANY DOUBT ABOUT THIS POSITION USE A TORQUE WRENCH WITH SPANNER TOOL TO FIND THE POSITION FOR THE NUT (15) WHERE MINIMUM TORQUE IS REQUIRED TO ROTATE THE NUT IN EITHER DIRECTION. IF THERE IS STILL DOUBT, ADJUST NUT SO THAT THE RIGHT HAND ROD END BOLT (3) MAY BE REMOVED AND REINSERTED BY HAND.

STEP 17

OPERATE THE ACTUATOR THROUGH ONE COMPLETE MOVEMENT UP AND DOWN. RECORD CURRENT AND TRAVEL TIME EACH WAY. CURRENT SHOULD BE 10 TO 16 AMPERES AND SHOULD NOT EXCEED 20 AMPERES ANY TIME DURING THE CYCLE.

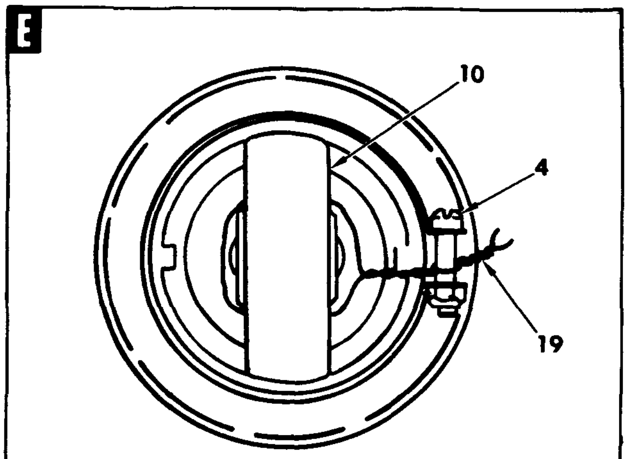
NOTE

☞ DISREGARD THE INITIAL PEAK AMPERAGE DRAW AT THE BEGINNING AND END OF THE CYCLE.


D
E
ROD END (10) RH THREAD-NYLOCK ENGAGES NUT
RAIN SHIELD (18) SHEET TYPE SHOWN, LATER MODELS HAVE MOLDED RUBBER
OIL VENT HOLE (11)
LOCKNUT (12) RADIAL HOLES FOR PIN
CLAMP (4)
PIN (17)
WASHER (14) RADIAL SERRATIONS ON BOTTOM-KEYED TO ROD END
GAP (STEP 14) 0.030 TO 0.090 APPROX
LOCK (13) CASTELLED-SPLINES ON ID
SCREW (16) LH THREAD ID -SPLINES ON OD
ENGAGEMENT (STEP 20) 0.070 TO 0.130 APPROX
NUT (15) RH THREAD ID-LH THREAD OD -FLANGE TOP RADIAL SERRATIONS -SPLINES ON FLANGE OD

JS-1-2-X1/2-2718-3

Figure 27-18. Removal and Replacement of the Pitch Trim Actuator Assembly (Sheet 3 of 4)



TYPICAL FOR BOTH JACKSCREWS-
SHIELD (18) OMITTED IN THIS VIEW

STEP 18

SET THE ACTUATOR AT APPROXIMATELY ZERO DEGREES AND ELIMINATE ALL FREE PLAY ON THE RIGHT JACKSCREW. ALTERNATELY, SHAKE THE RIGHT STABILIZER TIP WITH A RHYTHMIC LIGHT HAND PRESSURE AND, THEN, TURN NUT (15) CLOCKWISE AS MUCH AS POSSIBLE BY HAND. REPEAT THIS FOUR TIMES. CHECK CURRENT REQUIRED TO MAKE SMALL MOVEMENT OF ACTUATOR AND RETURN TO APPROXIMATELY ZERO DEGREES. THIS CURRENT SHOULD NOT BE SIGNIFICANTLY DIFFERENT FROM THAT READ IN STEP 17.

STEP 19

NOW APPLY THE PRELOAD USING A TORQUE WRENCH. ROTATE ADJUSTMENT NUT CLOCKWISE (FROM THE TOP) AT LEAST FOUR CASTELLATION ON THE LOCK (ABOUT 90 DEGREES) WITH A TORQUE OF 120 TO 240 INCH-POUNDS. RECHECK CURRENT, AND TRAVEL TIME. THE CURRENT SHOULD INCREASE AT LEAST 4 AMPERES, BUT NOT TO EXCEED 30 AMPERES, AND THE TRAVEL TIME SHOULD BE NO MORE THAN 4 SECONDS SLOWER THAN THOSE RECORDED IN STEP 17.

NOTE

AFTER FINAL ADJUSTMENT THE AMPERAGE DRAW SHOULD BE A MINIMUM OF 22 BUT NOT GREATER THAN 30 AMPERES AT THE NORMAL RANGE OF TRAVEL FOR THOSE JETSTARS MODIFIED TO THE 731 CONFIGURATION.

STEP 20

WITH THE PRELOAD SET, PRES DOWN ON THE LOCK (13) TO ASSURE ENGAGEMENT WITH THE SPLINES ON BOTH THE NUT (15) AND THE SCREW (16). A SLIGHT ROTATION (LESS THEN 3°) OF NUT (15) MAY BE REQUIRED FOR THIS BUT THIS WILL HAVE A NEGLIGIBLE EFFECT ON THE PRELOAD.

STEP 21

TIGHTEN DOWN THE LOCKNUT (12) WITH A TORQUE OF 20 TO 30 INCH-POUNDS.

STEP 22

AT ONE LOCATION AROUND THE PERIPHERY, A SLOT OF THE LOCK (13) WILL LINE UP WITH A HOLE IN THE LOCKNUT (12). INSTALL THE COTTER PIN (17) IN THIS HOLE AND SECURE. PULL UP BOOT AND INSTALL CLAMP (4). INSTALL LOCKWIRE (19) AS SHOWN IN VIEW E.

NOTE

THE MACH TRIM CYCLE TIMING SHOULD BE CHECKED AND/OR ADJUSTED IN ACCORDANCE WITH PARAGRAPHS 27.3.2.5 AND 27.3.5.4 WHENEVER PITCH TRIM PRELOAD IS ADJUSTED OR A PRIMARY MOTOR REPLACED IN A PITCH TRIM ACTUATOR.

COMPLETION

STEP 23

INSTALL THE RAIN SHIELDS (18) AS SHOWN IN VIEW D. LAP THE SEAM APPROXIMATELY 3/8 INCH USING SPECIFICATION MMM-A-1617 (1), TYPE III, 3M EC 2126, SCOTCHGRIP 2126, OR B.F. GOODRICH A-1231 B, OR EQUIVALENT. ON SOME LATE MODEL ACTUATORS, A MOLDED RUBBER RAIN SHIELD IS PROVIDED. THIS PART IS HELD IN PLACE WITH A SAFETY WIRE AND NO CEMENT IS REQUIRED. AS AN ADDED PRECAUTION, THE CAVITY UNDERNEATH THE BOOTS MAY BE PACKED WITH GREASE TO SEAL THE WATER PATHS. ANY GOOD GRADE OF GREASE WILL BE SATISFACTORY FOR THIS PURPOSE.

STEP 24

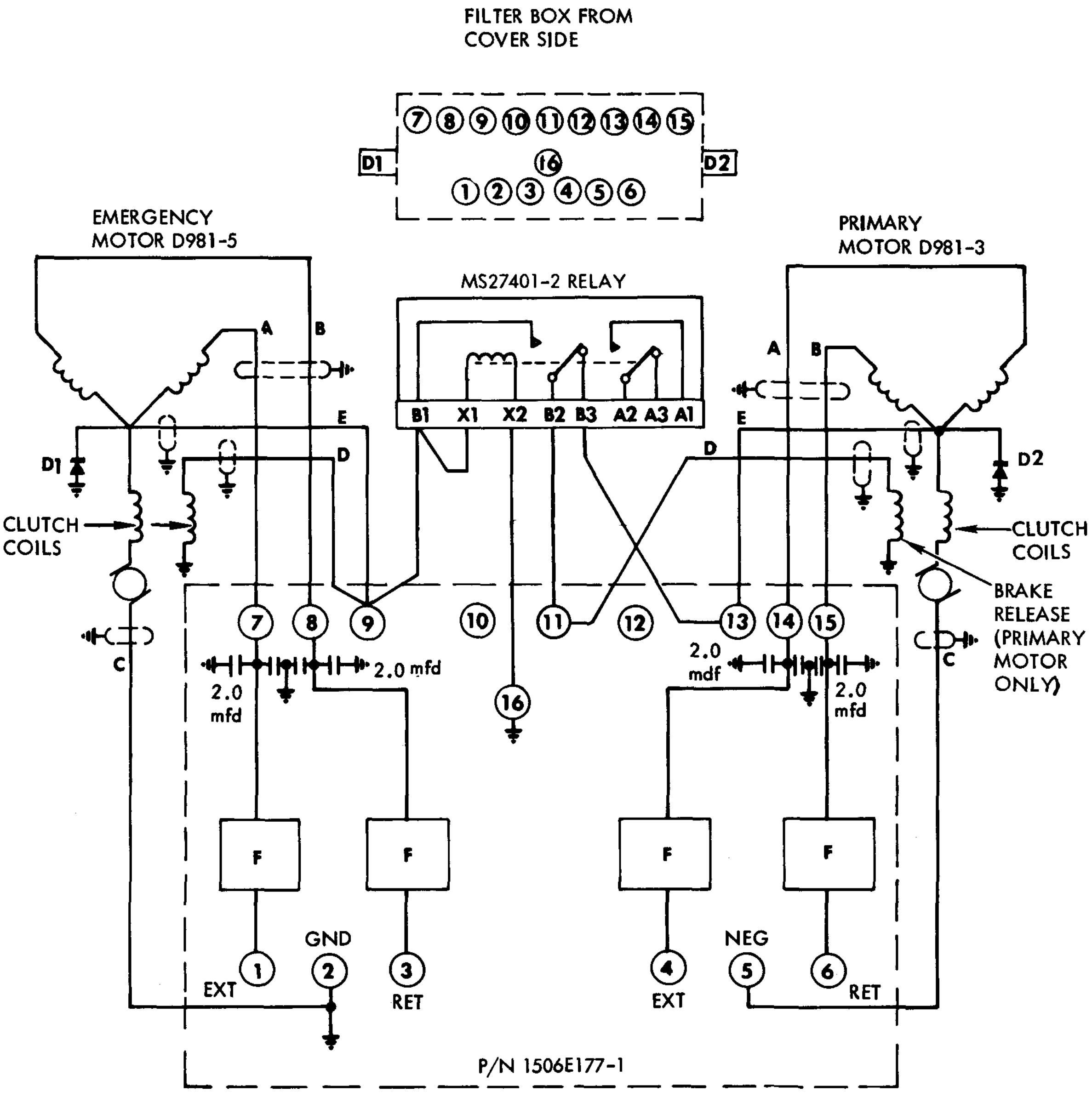
CHECK THAT ALL ROD END BOLT NUTS (1 AND 8) ARE FINGER TIGHT AND SECURED WITH COTTER PINS. CHECK ELECTRICAL CONNECTIONS AND INSTALL ANY REMOVED JUNCTION BOX COVERS. CHECK OIL LEVEL IN ACTUATOR.

STEP 25

RESTORE ALL REMOVED FAIRINGS AND RESTORE SYSTEM TO NORMAL. PERFORM OPERATIONAL CHECKOUT OF THE SYSTEM.

JS-1-2-X1/2-2718-4

Figure 27-18. Removal and Replacement of the Pitch Trim Actuator Assembly (Sheet 4 of 4)



F = FERRITE FILTER P/N T0620A-101D (ALLEN BRADLEY)

JS-1-2-X0/2-2718A

Figure 27-18A. Pitch Trim Actuator Electrical Schematic

SECTION 27.4
LEADING AND TRAILING EDGE WING FLAP SYSTEMS

Contents . . . Page

DESCRIPTION AND OPERATION . . . 27-60
General . . . 27-60
Wing Flap Controls . . . 27-61
Normal Flap Operation . . . 27-61
Emergency Flap Operation . . . 27-62
Position Indicating System . . . 27-62
Asymmetry Circuits . . . 27-62
Control Unit . . . 27-62
Travel Limit Switches . . . 27-63
Disconnect Reset Relay . . . 27-63
Safety Relays . . . 27-63
Wing Flap Selector Relay . . . 27-63
Leading Edge Flap Bypass Valve . . . 27-63
Shuttle Valves . . . 27-63
Wing Flap Selector Valve, System No. 1 . . . 27-63
Flap Drive Gearbox Assemblies . . . 27-64
Asymmetry Control Box . . . 27-64
Asymmetry Potentiometers . . . 27-64
Leading Edge Flap Angle Gearbox . . . 27-64
Leading Edge Flap Pressure Reducer . . . 27-64
Flow Regulators . . . 27-64
Surge Dampers . . . 27-64
Restrictors . . . 27-64
Check Valves . . . 27-64A

OPERATIONAL CHECKOUT . . . 27-64A
Leading and Trailing Edge Flap Systems Checkout Procedure . . . 27-65

TROUBLESHOOTING . . . 27-66

REMOVAL AND INSTALLATION . . . 27-79
Removal of Inboard Leading Edge Flap . . . 27-79
Installation of Inboard Leading Edge Flap . . . 27-79
Removal of Outboard Leading Edge Flap . . . 27-79
Installation of Outboard Leading Edge Flap . . . 27-79
Removal of Trailing Edge Flap Assembly Inboard and Outboard . . . 27-80
Installation of Trailing Edge Flap Assembly Inboard and Outboard . . . 27-80
Removal of External Tank-Flap Fairing . . . 27-80
Installation of External Tank-Flap Fairing . . . 27-80
Removal of Hydraulic Motors . . . 27-80
Installation of Hydraulic Motors . . . 27-81
Removal of Limit Valve . . . 27-81
Installation of Limit Valve . . . 27-81
Removal of Flap Drive Gearbox . . . 27-81
Installation of Flap Drive Gearbox . . . 27-81
Removal of Leading Edge Flap Actuator . . . 27-82
Installation of Leading Edge Flap Actuator . . . 27-82
Removal of Trailing Edge Flap Actuator . . . 27-82
Installation of Trailing Edge Flap Actuator . . . 27-83
Removal of Leading Edge Flap Asymmetry Potentiometer . . . 27-83
Installation of Leading Edge Flap Asymmetry Potentiometer . . . 27-84
Removal of Trailing Edge Flap Asymmetry Potentiometer . . . 27-84
Installation of Trailing Edge Flap Asymmetry Potentiometer . . . 27-84
Removal of Leading Edge Flap Limit Switch Assembly . . . 27-84
Installation of Leading Edge Flap Limit Switch Assembly . . . 27-85

Removal of Trailing Edge Flap Limit Switch Assembly 27-85
Installation of Trailing Edge Flap Limit Switch Assembly 27-85
Removal of Leading and Trailing Edge Torque Tubes 27-85
Installation of Leading and Trailing Edge Flap Drive Torque Tubes 27-86
ADJUSTMENT 27-86
Wing Flap Rigging Instructions 27-86
MAINTENANCE AND SERVICING PROCEDURES 27-86
Leading Edge Wing Flap Downstop Inspections 27-86
Repair of Teflon Rub Strips 27-86
Aerodynamic Smoothing and Sealing of Leading Edge Flap Shrouds 27-86
Leading Edge Flap Actuator Rod End Lubrication 27-87
Flap Actuator Jackscrew Lubrication 27-87
Flap Actuator Disassembly, Inspection, Lubrication, and Reassembly 27-87
Inspection for Excessive Looseness in Trailing Edge Wing Flaps 27-88
Leading Edge and Trailing Edge Flap Followup Switches 27-88A
Inspection For Excessive Looseness In Leading Edge Flaps. 27-88B
Flap Torque Tube and Drive Installation 27-88B
Indexing of Flap Drive Universal Joints 27-88B

FIGURES

27-19. Wing Flap Asymmetry System Schematic Diagram 27-89
27-20. Leading Edge Wing Flap Actuator Assembly 27-90
27-21. Aerodynamic Smoothing and Sealing of the Leading Edge Flap Shrouds 27-91
27-22. Leading and Trailing Edge Wing Flap Systems Schematic Diagram (Airplanes 5001 thru 5045) 27-93
27-23. Leading and Trailing Edge Wing Flap Systems Schematic Diagram (Airplanes 5046 and up) 27-95
27-24. Wing Flap Rigging Instructions 27-97
27-24A. Installation of Leading Edge and Trailing Edge Flap Torque Tubes 27-102A

27.4.1 DESCRIPTION AND OPERATION.

27.4.1.1 GENERAL. (See figures 27-22 and 27-23.) There are four leading edge flap surfaces and four trailing edge flap surfaces on JetStar airplanes. The inboard and outboard leading edge flaps travel different amounts, even though they are both driven by the same flap drive gearbox assembly. The trailing edge flaps, located inboard and outboard of the external fuel tanks, travel the same amount, and are connected together by fairing which fairs the flaps to the external fuel tanks, when the flaps are fully up.

a. Both leading and trailing edge flaps are controlled by a single flap control lever on the center console. During normal flap actuation, the forward half of lever travel controls the leading edge flaps and the aft half of lever travel controls the trailing edge flaps. When using the normal hydraulic system to operate the flaps, positioning the lever from full forward to full aft actuates the leading and trailing edge flaps simultaneously. Flow regulators in the hydraulic lines to the trailing edge flap drive motor insure that the leading edge flaps move faster than the trailing edge flaps, so the leading edge flaps always reach the fully down position before the trailing edge flaps.

b. During emergency or standby flap operation, the aft half of flap control lever travel actuates both the leading and the trailing edge flaps. In this mode of operation, the trailing edge flap control circuits control both leading and trailing edge flap actuation. As the leading edge flaps move faster than the trailing edge flaps, the leading edge flaps reach the fully-down position before the trailing edge flaps do, and instead of the leading edge flaps being stopped electrically as is normally the case, they are stopped by the mechanical action of a hydraulic limit valve in the flap drive gearbox assembly. This hydraulic limit valve shuts off fluid flow to the leading edge flap drive motor to limit leading edge flap travel. When flaps are retracted using this mode of operation, the leading edge flaps are stopped by the mechanical stops.

c. Normal flap operation uses power from the normal hydraulic system with the auxiliary hydraulic pump being automatically energized to supplement normal system power during trailing edge flap extension. Emergency or standby flap operation uses power from the standby hydraulic system. System selection is controlled by the position of a FLAP & BR HYD SYS selector switch on the copilot's side of the main instrument panel.

d. A flap asymmetry system automatically stops flap operation when symmetrically located flaps extend asymmetrically. During normal flap operation, when one of the leading edge flaps extends at a different rate than its symmetrically located

27.4.1.1 (Cont).

counterpart on the opposite wing, the asymmetry system trips a circuit to de-energize the leading edge flap selector valve to stop leading edge flap actuation. This action does not affect trailing edge flap operation. The same is true for the trailing edge flaps: when a trailing edge flap extends asymmetrically, the asymmetry circuit de-energizes the trailing edge flap selector valve to stop trailing edge flap actuation but does not affect leading edge flap operation. On airplanes 5001 through 5045, during emergency or standby flap operation, an asymmetric condition in the leading edge flaps will de-energize the leading edge bypass valve, stopping movement in the leading edge flaps only. An asymmetric condition in the trailing edge flaps will de-energize the

27.4.1.1 (Cont).

standby flap selector valve, stopping all flap movement. During emergency or standby flap operation on airplanes 5046 and up, an asymmetric condition in either the leading or trailing edge flaps will de-energize the standby flap selector valves and stop both the leading and trailing edge flap movement.

e. A flap position indicating system displays the wing flap positions in PERCENT EXTENDED. Two indicators, one for the leading edge flaps and one for the trailing edge flaps, are on the copilot's side of the main instrument panel. These indicators receive flap position signals from flap position transmitters in the followup switch assemblies which are positioned by the flap surfaces.

27.4.1.2 WING FLAP CONTROLS. A two-position ("NORM-1," "STANDBY-2") FLAP & BR HYD SYS selector switch on the copilot's side of the main instrument panel is provided for selection of either the normal hydraulic system or the standby hydraulic system to power the wing flaps. A flap control lever on the center console controls flap actuation. Positions on the flap control quadrant are, from front to rear: leading edge "UP," leading edge "DN" (or trailing edge "UP"), "TAKEOFF/APPROACH," and trailing edge "DN" (or "LANDING"). The auxiliary hydraulic pump is automatically energized to supplement normal system power during trailing-edge flap extension if the FLAP & BR HYD SYS switch is in the "NORM-1" position.

27.4.1.3 NORMAL FLAP OPERATION. During normal flap operation, the FLAP & BR HYD SYS switch must be in the "NORM-1" position. This makes it possible to control either the leading or the trailing edge flaps independently of the others according to the travel range of the flap control lever. During emergency or standby flap operation, both systems are controlled by the trailing edge flap followup control switch.

a. Placing the flap control lever in the leading-edge "DN" position mechanically positions the leading edge flap followup control switches to admit 28-volt, DC power from the main bus to pass through the FLAP & BR HYD SYS switch, through the leading-edge flap followup control switch, through the down-limit contacts of the leading-edge flap followup switch, through the de-energized flap disconnect reset relay No. 1 and the leading edge flap safety relay to energize the down solenoid of the leading edge flap selector valve. Energizing this valve admits normal hydraulic system pressure to drive the flap-drive motor to actuate the leading-edge flaps down. The flap followup switch is mechanically linked to the flap surfaces, and when the flaps reach the fully-down position the down-limit contacts in this switch open to break the control circuit to the selector valve. This returns the selector valve to the de-energized position and blocks hydraulic pressure to the flap-drive motor. If the down-limit switch should fail to stop flap actuation, a hydraulic limit valve is mechanically driven by the flap drive gearbox to block hydraulic pressure from the flap drive motor.

b. Placing the flap control lever in the "TAKEOFF/APPROACH" position permits electrical power to pass through the 40 percent contacts in the trailing edge flap followup switch, through the trailing-edge flap disconnect reset relay No. 2, through the wing flap selector relay, and through the trailing edge flap safety relay to the flap boost disconnect relay and to the down solenoid of the trailing edge flap selector valve. Energizing the flap boost disconnect relay allows essential bus power to energize the auxiliary hydraulic system No. 3 relay to start the automatic action of the auxiliary hydraulic pump. Energizing the down solenoid of the trailing edge flap selector valve directs normal (and auxiliary) hydraulic power to the trailing-edge flap-drive motor to drive the trailing edge flaps down. When the flaps reach approximately 40 percent extension, the 40 percent limit contacts in the flap followup switch open to break the circuit to the auxiliary hydraulic pump and to the trailing-edge flap selector valve, thereby stopping flap actuation.

c. Placing the flap control lever in the "LANDING" position completes a circuit to the trailing edge flap selector valve similar to the one for the "TAKEOFF/APPROACH" actuation, except power passes through different contacts in the flap followup control switch and in the flap followup switch. In this case, if the flaps fail to stop at the fully extended position, a hydraulic limit valve blocks pressure from the flap drive motor in the same manner as the leading-edge flap down-limit is effected.

d. Placing the flap control lever in the trailing-edge "UP" (same as leading-edge "DN") position when flaps are extended, energizes the trailing-edge flap safety relay to allow main bus power to go through the trailing edge flap followup control switch contacts, through the up contacts in the trailing-edge flap followup switch, through flap disconnect reset relay No. 2, through the wing flap selector relay, and through the energized trailing-edge flap safety relay to energize the up solenoid of the trailing-edge flap selector valve. As the flaps reach faired position, the trailing-edge flap followup switch up contacts open to break the control circuit and stop flap actuation.

e. Placing the flap control lever in the leading-edge "UP" position allows main bus power to go through the FLAP & BR HYD SYS switch to energize the leading edge flap safety relay. This

27.4.1.3 (Cont).

admits main bus power through the up contacts of the leading edge flap followup control switch, through the up-limit contacts of the leading edge flap followup switch, through flap disconnect reset relay No. 1 and the energized leading edge flap safety relay to the up solenoid of the leading edge flap selector valve. As flaps streamline with the wing, the up contacts in the leading-edge flap followup switch open to break the control circuit to the selector valve, stopping flap actuation.

27.4.1.4 EMERGENCY FLAP OPERATION. For emergency, or standby, flap operation, the FLAP & BR HYD SYS switch must be in the "STANDBY-2" position. This disables the leading edge flap control circuits and, on airplanes 5001 through 5045, sends power through the leading edge flap disconnect reset relay No. 1 to energize the leading edge flap bypass valve. Energizing this valve admits standby hydraulic system pressure to the leading edge flap selector valve and blocks normal hydraulic system power from both the leading and the trailing edge flap-drive motors. This makes it possible to actuate both the leading and the trailing edge flap systems with standby hydraulic system power by controlling the wing flap selector valve. During standby operation of the flaps, the trailing edge flaps can be controlled to stop at the selected positions, but the leading edge flaps normally stop only when the trailing edge flaps stop. If the trailing edge flap followup switch down-limit contacts fail to stop flap actuation, both flap systems will continue extending until they reach their fully-down positions and their hydraulic limit valves block standby system pressure from the drive motors. Another function of the FLAP & BR HYD SYS switch is to energize the flap selector relay which disables the automatic function of the auxiliary hydraulic pump.

a. Placing the flap control lever in the "TAKEOFF/ APPROACH" or "LANDING" positions admits power from the main bus, through the appropriate trailing edge flap followup control switch contacts, through the corresponding followup travel-limit contacts, through the flap disconnect reset relay No. 2, through the energized wing flap selector relay, and through the trailing edge flap safety relay to the down solenoid of the wing flap selector valve. As the trailing edge flaps reach the selected position, the appropriate limit contact in the trailing edge flap followup switch opens to break the circuit and stop both the leading and trailing edge flap system actuations.

b. Placing the flap control lever in the trailing edge "UP" (same as the leading edge "DN") position completes a circuit through the up contacts in the trailing edge flap followup control switch to energize the trailing edge flap safety relay, and completes a circuit through the up-limit contacts of the trailing edge flap followup switch, through the flap disconnect reset relay No. 2, through the energized wing flap selector relay, and through the energized trailing edge flap safety relay to the up solenoid of the wing flap selector valve to actuate both the leading and the trailing edge flaps up. As the trailing edge flaps reach the fully-up position, the trailing edge flap up-limit contacts in the trailing edge flap followup switch open to stop both leading and trailing edge flap actuation.

27.4.1.5 POSITION INDICATING SYSTEM. Separate leading and trailing edge flap position indicating systems show flap positions in PERCENT EXTENDED. Leading edge and trailing edge flap position indicators are on the copilot's instrument panel. Position transmitters for these indicators are incorporated in the followup switch assemblies.

27.4.1.6 ASYMMETRY CIRCUITS. Both the leading and the trailing edge flap systems have independent asymmetry circuits that shut off power to their flap systems when the flaps start out of symmetry. (Refer to figure 27-19.) If leading edge flaps start out of asymmetry, the leading edge flap system will normally be shut off automatically, but the trailing edge flap system will continue to operate. Similarly, if the trailing edge flaps are de-energized because of asymmetry, the leading edge flap system should continue to operate. During emergency or standby flap operation on airplanes 5001 through 5045, an asymmetry condition in the leading edge flaps will de-energize the leading edge bypass valve, stopping movement in the leading edge flaps only. An asymmetric condition in the trailing edge flaps will de-energize the wing flap selector valve, stopping all flap movement. During emergency or standby flap operation on airplanes 5046 and up, an asymmetric condition in either the leading or trailing edge flaps will de-energize the wing flap selector valve and stop both the leading and trailing edge flap movement. Each asymmetry circuit consists of an asymmetry potentiometer for each flap which feeds a signal into the asymmetry controller. Each asymmetry potentiometer has its wiper linked to the flap surface by a cable. As flap positions normally remain symmetrical, the signals fed into the asymmetry controller are normally balanced. When a flap starts out of symmetry, the relative positions of the wipers are no longer the same, and thus the signals to the asymmetry controller are unbalanced. When the unbalanced signals reach the asymmetry controller, they trip a circuit to send power to energize the respective flap disconnect reset relay and thus break the power supply circuit to the flap control system. As the power supply circuit is broken, the flap control valves return to their de-energized position to stop flap actuation.

27.4.1.7 CONTROL UNIT. A flap control unit on the center console contains a leading edge flap followup control switch and a trailing edge flap followup control switch. Contacts of these switches are actuated by positioning the flap control lever. The forward half of lever travel controls the leading edge flap followup switch contacts, and the aft half

27.4.1.7 (Cont).

of lever travel controls the trailing edge flap followup control switch contacts while maintaining the leading edge flap followup switch down contact in the closed position.

27.4.1.8 TRAVEL LIMIT SWITCHES. One leading edge flap followup switch and one trailing edge flap followup switch electrically limit flap travel during normal flap operation. The leading edge flap followup switch contacts open as the flaps reach the position selected with the flap control lever, to break the leading edge flap actuating circuit. The trailing edge flap followup switch contacts open as the flaps reach the position selected with the flap control lever, to break the trailing edge flap actuating circuit. During emergency or standby flap operation, the trailing edge flap actuating circuits control both leading and trailing edge flap operation, so in this mode of operation, the trailing edge flap followup switch limits trailing edge flap actuation and the leading edge flap actuation is limited by mechanical action of a hydraulic limit valve in the leading edge flap drive gearbox assembly.

27.4.1.9 DISCONNECT RESET RELAY. A No. 1 flap disconnect reset relay for the leading edge flap asymmetry circuit, and a No. 2 flap disconnect reset relay for the trailing edge flap asymmetry circuit de-energize their respective flap selector valves to stop flap actuation when an asymmetrical-flap condition exists. Energizing the No. 1 flap disconnect reset relay breaks power between the wing flap control circuit breaker and the leading edge flap safety relay, thereby interrupting the actuating circuit. Energizing the No. 2 flap disconnect reset relay breaks power between the wing flap control circuit breaker and the wing flap selector relay thereby interrupting the actuating circuit. The No. 1 flap disconnect reset relay is in the left wheel well. The No. 2 flap disconnect reset relay is on the pilot's side shelf.

27.4.1.10 SAFETY RELAYS. A leading edge flap safety relay energizes for leading edge flap retraction and de-energizes for leading edge flap extension to prevent power from being applied simultaneously to both the up and down solenoids of the leading edge flap selector valve. A trailing edge flap safety relay energizes for trailing edge flap retraction and de-energizes for trailing edge flap extension to prevent power from being applied simultaneously to both the up and down solenoids of the trailing edge flap selector valve. The trailing edge flap safety relay also operates with the wing flap selector relay to select emergency flap circuitry. The leading edge flap safety relay is in the left wheel well. The trailing edge flap safety relay is in the right wheel well.

27.4.1.11 WING FLAP SELECTOR RELAY. A wing flap selector relay is energized when the FLAP & BR HYD SYS selector switch is placed in the "STANDBY-2" position. Energizing this relay disconnects the wing flap selector valve, system No. 1, circuits from the valve and connects them to the wing flap selector valve system No. 2, to ready the flap systems for emergency or standby operation. During normal flap operation, the wing flap selector valve, system No. 1, controls trailing edge flap actuation. During emergency or standby flap operation, the wing flap selector valve, system No. 2, controls both leading and trailing edge flap actuation. The wing flap selector relay is in the right wheel well.

27.4.1.12 LEADING EDGE FLAP BYPASS VALVE (AIRPLANES 5001 THRU 5045 ONLY). The leading edge flap bypass valve is a three-way, two-position, single-solenoid valve. This valve is energized by the FLAP & BR HYD SYS selector switch when the switch is in the "STANDBY-2" position. Energizing this valve allows standby hydraulic system fluid from the wing flap selector valve, system No. 2, to bypass the leading edge flap selector valve and thus power the leading edge flap drive motor. This valve is in the left wheel well.

27.4.1.13 WING FLAP SELECTOR VALVE, SYSTEM NO. 2. The wing flap selector valve, system No. 2, is a four-way, three-position, double-solenoid valve. This valve controls both leading and trailing edge flap actuation during emergency or standby flap operation. During normal flap operation, this valve is de-energized to open the flap-down lines from the gearbox assemblies to standby system return, thereby allowing the shuttle valve to shift for using the normal hydraulic system. On airplanes 5046 and up, both flap up and flap down lines are opened to standby return when the valve is de-energized. This selector valve is inside the right, rear wing-to-fuselage fairing.

27.4.1.14 LEADING EDGE FLAP SELECTOR VALVE. The leading edge flap selector valve is a four-way, three-position, double-solenoid valve. This valve controls actuation of the leading edge flaps during normal flap operation. For airplanes 5001 through 5045, when de-energized, this valve opens the down lines from the flap drive gearbox assemblies to normal return to allow the standby system to shift the shuttle valves when the standby system is selected for flap operation. On airplanes 5046 and up, both flap up and flap down lines are opened to normal return when the valve is de-energized. This valve is inside the left front wing-to-fuselage fairing.

27.4.1.15 SHUTTLE VALVES. On airplanes 5001 through 5045, two shuttle valves in the down-lines to the flap gearbox assemblies shift under pressure to allow either the normal or the standby hydraulic system to power the wing flaps. On airplanes 5046 and subsequent, shuttle valves in the up lines and down lines to the flap gearbox assemblies shift under pressure to allow either the normal or the standby hydraulic system to power the wing flaps. The shuttle valves in the up line and the down line to each gearbox are linked mechanically to prevent fluid transfer between the normal hydraulic system and the standby hydraulic system.

27.4.1.16 WING FLAP SELECTOR VALVE, SYSTEM NO. 1. The trailing edge selector valve is a four-way, three-position, double-solenoid valve. This valve controls normal actuation of the trailing edge flaps.

When de-energized, this valve opens the down line of the trailing edge flap gearbox assembly to normal system return. On airplanes 5046 and up, both flap up and flap down lines are opened to normal return when the valve is de-energized. This valve is inside the left, rear wing-to-fuselage fairing. This valve is identical to the leading edge flap selector valve and to the wing flap selector valve, system No. 2. (See 27.4.1.14 and 27.4.1.13.)

27.4.1.17 FLAP DRIVE GEARBOX ASSEMBLIES. There are two flap drive gearbox assemblies: one for the leading edge flaps and one for the trailing edge flaps. Each assembly consists of a hydraulic motor, gearing to drive output shafts, and a hydraulic limit valve. The hydraulic limit valve contains a position indicator which shows when the limit valve is in the shut off position. On installation, the hydraulic system tubes are connected to the hydraulic motor ports and to the limit valve ports such that the fluid flow for flaps down must pass through the limit valve before it reaches the hydraulic motor. The limit valve is mechanically driven by the gearbox output shaft so that, after a preset number of shaft revolutions, the valve closes and shuts off hydraulic fluid flow to the motor. Output shafts of each gearbox assembly are marked to show the direction of rotation for flap extension and retraction. On airplanes 5001 thru 5045 only, during normal flap operation, drainage from the trailing edge flap drive hydraulic motor returns to the standby system reservoir. When this additional fluid from the normal hydraulic system overfills the standby hydraulic system reservoir, the excess fluid passes through the pressurization lines connecting the reservoirs to return to the normal hydraulic system reservoir. Failure of the motor seals in the trailing edge flap drive gearbox assembly can increase this drainage to a point where the fluid tends to back up into the reservoir pressurization lines, and may overflow through the vent system. The leading edge flap drive gearbox assembly is near the left front wing-to-fuselage attachment point inside the fairing.

27.4.1.18 ASYMMETRY CONTROL BOX. A two-channel asymmetry control box for the leading edge flaps and the trailing edge flaps, control the flap disconnect reset relays to stop flap actuation power. The control box normally receives balanced signals from the potentiometers driven by the flaps. When an asymmetrical condition exists, these signals become unbalanced to trip circuits in the control box and energize the corresponding flap disconnect reset relay.

27.4.1.19 ASYMMETRY POTENTIOMETERS. One potentiometer near each flap surface is positioned by a cable attached to the flap surface. Each potentiometer is spring loaded to keep tension on the cable even when the flaps are fully up.

CAUTION

Never allow potentiometer cables to snap back to the spring-loaded position. Allowing the cables to do this damages the potentiometers.

27.4.1.20 LEADING EDGE FLAP ANGLE GEARBOX. This gear box provides a change in direction of torque tubes at the airplane center line to provide for use of a single drive unit for both left-hand and right-hand leading edge flaps.

27.4.1.21 LEADING EDGE FLAP PRESSURE REDUCER. A pressure reducing valve is installed in the leading edge flap system so as to reduce flap up operating pressure regardless of which system is operating the flaps. For airplane serials 5001 through 5145, the pressure is reduced to 1500 (±50) PSI. Serials 5146 and up have a pressure reduction to 1800 (±50) PSI. The reduced pressure results in smaller loads on the mechanical flap up stops during standby system operation and in the event of a failure of the electrical up stop during normal system or standby system operation.

27.4.1.22 FLOW REGULATORS. Flow regulators are used in the flap system to control normal system flow to the flap drive gearbox assemblies. Leading edge hydraulic flow is regulated to 2.5 GPM and trailing edge flow to 5.0 GPM.

a. (Airplanes 5001 through 5045) Two-way flow regulators are located downstream of the leading edge flap selector valve and the trailing edge flap selector valve in the flap down lines.

b. (Airplanes 5046 and up) One-way flow regulators are located upstream of the leading edge flap selector valve and the trailing edge flap selector valve.

27.4.1.23 SURGE DAMPERS. Hydraulic surge dampers are located upstream of the leading edge and trailing edge flap selector valves. The surge dampers protect equipment by reducing the magnitude of pressure ripples in hydraulic pressure lines.

27.4.1.24 RESTRICTORS. Restrictor valve assemblies are located in the standby hydraulic system trailing edge flap plumbing to meter the flow of fluid to the trailing edge flap drive gearbox assembly, thus ensuring proper sequencing of leading edge and trailing edge flaps.

a. (Airplanes 5001 through 5045) A two-way restrictor valve is located in the trailing edge flap down line downstream of the wing flap selector valve, system No. 2, and upstream of the trailing edge shuttle valve.

b. (Airplanes 5046 and up) One-way restrictor valve assemblies are located in the standby hydraulic system flap up and flap down pressure lines downstream of the wing flap selector valve, system No. 2.

27.4.1.25 CHECK VALVES. Several check valve assemblies are installed in wing flap hydraulic plumbing, positioned so as to prevent undesirable hydraulic fluid flow.

a. Two check valves are installed in the plumbing associated with the leading edge flap pressure reducer. One is used to control a bypass line around the pressure reducer for unhampered flap down operation. The second check valve is positioned to prevent inadvertent pressurization of the pressure reducer assembly housing.

b. (Airplanes 5046 and up) Two check valves are installed in the case drain lines from each flap drive hydraulic motor, the leading edge motor, and the trailing edge motor. These check valves protect the motor cases (housings) from overpressurization.

c. (Airplanes 5047 and up, and airplanes having Service Bulletin 329-166 incorporated) Two check valves are located in the return lines downstream of the leading edge flap and trailing edge flap selector valves. The primary purpose of these valves is to prevent repositioning of the flap system shuttle valves by transient surges in the normal hydraulic system return line.

d. (Airplanes 5046 and up having Service Bulletin 329-283 incorporated) There is a check valve in the return line downstream of the wing flap selector valve, system No. 2. The primary purpose of this valve is to prevent repositioning of the flap system shuttle valves by pressures in the standby system return line.

27.4.2 OPERATIONAL CHECKOUT

Test Equipment Required

Nomenclature	Description	Use and Application
Aircraft Hydraulic Test Stand Portable.	Type D-5A or type D-6A. (MIL-H-5606 hydraulic fluid)	To pressurize hydraulic systems.
Generator Set	Type MD-3A or type C-26	To energize flap circuits.
Ground Power Unit	28-volts, DC, 300-amp	

Tools Required

Nomenclature	Description	Use and Application
Aileron and Elevator Protractor.	Lockheed P/N 11069-MEU.	To determine trailing edge flap travel.
Bubble Protractor		To measure leading edge flap travel.

27.4.2.1 LEADING AND TRAILING EDGE FLAP SYSTEMS CHECKOUT PROCEDURE.

CAUTION

On airplanes 5001 thru 5045, do not actuate the flaps down on one system and then up on the other. To do so may deplete one reservoir and overfill the other.

CAUTION

Limit auxiliary hydraulic pump operations to five per hour.

a. Make sure that both the normal and the standby hydraulic systems are serviced properly in accordance with the requirements of Chapter 5.

b. Close the following circuit breakers:

No. 1 Main Inverter
Wing Flap Control, LE and TE
Wing Flap Asymmetry
Hydraulic Pumps
System No. 1 Hydraulic Pressure
System No. 2 Hydraulic Pressure

c. Operate the leading and the trailing edge flaps to the fully-up position. Leading edge flaps are fully up when the flap fittings are no more than 0.03 inch from the wing front beam fittings (refer to the flap rigging instructions, figure 27-24 for locations of these components). Trailing edge flaps are fully up when faired within 0.06 inch of the trailing edge of the wings and fairing.

d. With the flaps up as in step (c), attach bubble protractors to the inboard and outboard leading edge flaps and to the trailing edge flaps. Adjust the protractors to read zero in this position.

e. Connect external hydraulic power to the normal hydraulic system ground test connections. Connect 28-volt, DC power to the external power receptacle of the airplane, and energize the main AC bus.

f. Adjust the hydraulic power source for 3,000 PSI at zero flow and 2,700 PSI at 8 gallons per minute.

g. With the flap and brake HYD SYS selector switch in the "NORM-1" position, place the flap control lever in the leading edge "DN" position. Check the leading edge flap actuation time and flap travel. Observe the flap position indicators.
RESULT: The leading edge flaps should actuate from fully up to fully down in 7.5 (±2) seconds. The inboard leading edge flaps should move down 22(±2) degrees, and the outboard leading edge flaps should move down 27(±2) degrees. The leading edge flap position indicator should show "DN" ±5 percent; the trailing edge flap position indicator should show "UP" ±5 percent.

h. Place the flap control lever in the "TAKEOFF/APPROACH" position, and check the flap travel. Observe the flap position indicators.
RESULT: The trailing edge flaps should move 20(±2) degrees down from the faired position. The trailing edge flap position indicator should show 40(±5) percent. The leading edge flaps should not move.

i. Place the flap control lever in the trailing edge "UP" position until the trailing edge flaps fair to the wing. Place the flap control lever in the "LANDING" position, and time the actuation. Check the flap travel, and observe the trailing edge flap position indicator.
RESULT: The trailing edge flaps should actuate from fully up to fully down in 15 (±2) seconds. Trailing edge flap travel should measure 50 (+0, -3) degrees. The auxiliary hydraulic pump should energize as the trailing edge flaps start down, and it should de-energize as trailing edge flap actuation stops. The trailing edge flap position indicator should show "DN" ±5 percent. The leading edge flaps should not move.

j. Place the flap control lever in the trailing edge "UP" position, and time the flap actuation.
RESULT: The trailing edge flaps should actuate from fully down to fully up in 15 (±2) seconds. The leading edge flaps should not move.

k. Place the flap control lever in the leading edge "UP" position, and time the flap actuation.
RESULT: The leading edge flaps should actuate from fully down to fully up in 7.5 (±2) seconds. The trailing edge flaps should not move.

l. Disconnect the electrical connector from the leading edge flap selector valve, and energize pins B (+) and C(-) with 28-volt, DC power.
RESULT: The leading edge flaps should actuate down normally until they reach their normal travel limit, and then the flaps should actuate slowly until they finally stop with the outboard flaps 28 (±2) degrees down from the faired position.

m. Disconnect the electrical connector from the trailing edge selector valve, and energize pins B (+) and C (-) with 28-volt, DC power.
RESULT: The trailing edge flaps should actuate down normally until they reach their normal travel limit, and then they should actuate slowly until they finally stop 50 (+3, -0) degrees down from the faired position.

n. Reconnect the connectors to both the leading and trailing edge flap selector valves, and return the flap control lever to the leading edge "UP" position. While actuating the flaps, disturb each

27.4.2.1 (Cont).

asymmetry potentiometer cable one at a time, and observe flap operation.

RESULT: When an asymmetry potentiometer cable for the leading edge flaps is disturbed, the leading edge flaps should stop, but the trailing edge flaps should not be affected. When an asymmetry potentiometer cable for the trailing edge flaps is disturbed, the trailing edge flaps should stop, but the leading edge flaps should not be affected. Releasing the potentiometer cable does not reset the flaps on airplanes 5001 thru 5045; instead, the flaps must be reset by pressing the reset button on the controller. On any airplane having the JR 676-51 or JR 3012 controller installed as preferred spare or on airplanes 5046 and up, releasing the potentiometer cable automatically resets the flaps to normal operation.

Note

During periodic check every 300 hours check the safety relays to see that the "UP" circuit grounds are made. To do this, disconnect the connector at the three leading and trailing edge systems selector valves. Use a multimeter to check for continuity to ground on pin A of each plug. Make sure electrical power is off before making this check.

o. Disconnect the leading edge flap potentiometer connector from the flap asymmetry controller, and connect a test harness (the one used in the flap rigging procedures) to the leading edge flap potentiometer connector. (See figure 27-24, detail L.)

p. Actuate the leading edge flaps throughout their full range of travel. Using Triplett Model 630-T or equivalent multimeter set on the 0.3-volt scale, observe the voltage across terminals No. 3 and 4, and then 5 and 6. Terminals 3 and 4 are for the outboard potentiometers; terminals 5 and 6 are for the inboard potentiometers.

RESULT: The voltage across the terminals should be within the limits of +0.15 volt and -0.15 volt. The variation of voltage, as the flaps travel from one extreme position to the other extreme position, must be equally divided about zero volts; that is, the variation in the plus (+) direction must be equal to the variation in the minus (-) direction.

q. Disconnect the asymmetry test harness from the leading edge flap potentiometer connector.

r. Disconnect the trailing edge flap potentiometer connector from the flap asymmetry controller. Connect an asymmetry test harness (the one used in the flap rigging procedures) to the trailing edge flap potentiometer connector. (See figure 27-24, detail L.) Actuate the trailing edge flaps throughout their full range of travel. Using a Triplett Model 630-T or equivalent multimeter, observe the voltage across terminals No. 3 and 4, and then 5 and 6. Terminals 3 and 4 are for the outboard potentiometers; terminals 5 and 6 are for the inboard potentiometers.

RESULT: The voltage across the terminal shall be within the limits of +0.15 volt and -0.15 volt. The variation of voltage, as the flaps travel from one extreme position to the other extreme position, must be equally divided about zero volts; that is, the variation in the plus (+) direction must be equal to the variation in the minus (-) direction.

s. With the flaps in the fully up position, reduce hydraulic pressure to zero, and transfer the external hydraulic source from the normal hydraulic system ground test connections to the standby hydraulic system ground test connections. Adjust the hydraulic power source to deliver no flow at 3,000 PSI and 8 gallons per minute at 2,700 PSI. Position the hydraulic system selector switch in "STANDBY-2" position. Repeat steps (o), (p), and (q) as the flaps are operated. Place the flap control lever in the leading-edge "DN" position.

RESULT: The flaps should not be affected.

t. Place the flap control lever in the "LANDING" position.

RESULT: Both the leading and the trailing edge flaps should actuate. The leading edge flaps should stop before or at the same time as the trailing edge flaps stop. The auxiliary hydraulic pump should not energize.

Note

The trailing edge flaps should actuate from fully up to fully down in 55(± 15) seconds.

u. Return the flaps to the fully up position, reduce hydraulic pressure to zero, return the flap and brake HYD SYS selector switch to the "NORM-1" position, disconnect both hydraulic and electric power sources from the airplane, remove the protractors from the flaps, disconnect asymmetry test harness and return the flap system to normal.

27.4.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM.		
	Note The following tests are arranged in a sequence that will progressively eliminate sections of the system as the cause of a malfunction.	

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM (Cont).		
Asymmetry control tripped.	Obtain access to the asymmetry controller in the L.H. wing root area. Observe the "L.E. tripped" light on. Remove the L.E. connector from the controller. Check for misaligned or defective sensor in accordance with paragraph 27.4.6.	Realign or replace sensors as indicated by test.
	If sensors are aligned correctly as determined in step 1, reconnect the asymmetry controller to the system and observe the warning light during flap operation. If the tripped condition persists, inspect the flaps and drive tubes for indication of failure which will cause asymmetric condition.	Repair mechanical failure.
	If no mechanical asymmetric condition exists, as determined in step 2 and if tripped condition persists the controller is defective.	Replace controller.
Electrical power failure.	Obtain access to the connector on the flap followup control switch on the center pedestal. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Remove the connector from the control switch assembly and check for 28 volts, DC power on pins "M" & "G" of the plug. If power is on pin "G" but not on pin "M" remove the main instrument panel disconnect C-11 and check for power at pins "F" and "D" of the receptacle. If power is on pin "F" but not on pin "D" verify discontinuity of wiring at the plug C-11 through pins F and D.	Replace HYD. system selector switch JR1081-1.
	Obtain access to terminal strip E6 in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down. Check for 28-volt, DC power at terminal "8" of terminal strip E6. If power is not on this point, remove the connector from the flap follow-up control switch and verify continuity from terminal "8" of E6 to pin "L" of the connector.	Replace follow-up control switch, JR1052.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	Obtain access to terminal strip E6 in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down. Check for 28-volt, DC power at terminals 8 and 21 of terminal strip E6. If power is on terminal 8 but not on terminal 21, obtain access to the L.E. follow-up switch and remove the connector from the switch section. Verify 28-volt, DC power at pin "F" and continuity from pin "B" to terminal 21 of E6.	Replace follow-up switch JR1051.
	Obtain access to terminal strip E6 in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down. Check for 28-volt, DC power at terminals 4, 19 and 21 of terminal strip E6. If power is on terminal 21 but not on terminal 4 verify that power is not on terminal 19.	Replace flap disconnect reset relay MS25268-D1.
	Obtain access to the L.E. wing flap safety relay in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down. Check for 28-volt, DC power at terminals A3 and A2 of the safety relay. Verify power to be on one terminal but not on the other.	Replace the safety relay AN3311-2.
	Obtain access to the L.E. wing flap safety relay and the L.E. wing flap control valve in the L.H. wing root area. Close the wing flap control circuit breaker and move the flap handle to flaps "DN." Remove the connector from the control valve. Verify circuit continuity from pin "C" to ground and 28-volt, DC power on pin B.	Replace JH1013 wing flap control valve.
LEADING EDGE FLAPS FAIL TO MOVE UP OR DOWN ON STANDBY SYSTEM.		

Note

The following tests are arranged in a sequence that will progressively eliminate sections of the system as the cause of a malfunction.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE UP OR DOWN ON STANDBY SYSTEM (Cont).		
Asymmetry control tripped (Note: T.E. asymmetry signal will trip L.E. when standby system is selected.)	Obtain access to the asymmetry controller in the L.H. wing root area. Observe the "L.E. tripped" light on. Remove the L.E. connector from the controller. Check for misaligned or defective sensor in accordance with paragraph 27.4.6.	Realign or replace sensors as indicated by test.
	If sensors are aligned correctly as determined above, reconnect the asymmetry controller to the system and observe the warning light during flap operation. If the tripped condition persists inspect the flaps and drive tubes for indication of failure which will cause asymmetric condition.	Repair mechanical failure.
	If no mechanical asymmetric condition exists as determined above, and if tripped condition persists the controller is defective.	Replace controller.
Electrical power failure.	Determine that the wing flap L.E. and T.E. control circuit weaker is closed and that the 28-volt, DC main bus is energized. Remove the main instrument panel connector C-11. Move the hydraulic system select switch to standby. Check that power is on pin F of the plug. On the receptacle side of C-11, check that continuity does not exist between pins F and E.	Replace the hydraulic system select switch JR1081 (or MS25068-23).
	On airplanes 5046 and up, use the procedure specified for the "T.E. flaps fail to move down on standby hydraulic system."	Replace the L.E. flap disconnect relay MS25268-D1.
	On airplanes 5001 thru 5045, obtain access to terminal strip E6 in the L.H. wing fillet area. Determine that the 28-volt, DC main bus is energized and that the wing flap L.E. and T.E. control circuit breaker is closed. Move the hydraulic system selector switch to standby. Show that 28-volt, DC power exists on terminal 22 of E6 but does not exist on terminal 23 or terminal 19 (asymmetry).	Replace the L.E. flap disconnect relay MS25268-D1.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE UP OR DOWN ON STANDBY SYSTEM (Cont).		
Electrical power failure (cont).	On airplanes 5001 thru 5045, use the same set-up as in the preceding step, but check that power is on both terminal 22 and 23 of E6 (and not on 19 of E6). Remove the connector from the L.E. flap bypass valve. Determine that 28-volt, DC power exists on pin A of the connector and a good ground exists on pin B.	Replace the L.E. flap bypass valve - JH1018.
LEADING EDGE FLAPS FAIL TO MOVE UP ON NORMAL SYSTEM.		
	Note The following tests are arranged in a sequence that will progressively eliminate sections of the system as the cause of a malfunction.	
Asymmetry control tripped.	Obtain access to the asymmetry controller in the L.H. wing root area. Observe the "L.E. tripped" light on. Remove the L.E. connector from the controller. Check for misaligned or defective sensor in accordance with paragraph 27.4.6.	Realign or replace sensor as indicated by test.
	If sensors are aligned correctly as determined above, reconnect the asymmetry controller to the system and observe the warning light during flap operation. If the tripped condition persists, inspect the flaps and drive tubes for indication of failure which will cause asymmetric condition.	Repair mechanical failure.
	If no mechanical asymmetric condition exists as determined above, and if tripped condition persists, the controller is defective.	Replace controller.
Electrical power failure.	Obtain access to the connector on the flap follow-up control switch on the center console. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Remove the connector from the control switch assembly and check for 28-volt, DC power on pins "M" and "G" of the plug. If power is on pin "G" but not on pin "M" remove the main instrument panel disconnect C-11 and check for power at pins	Replace hydraulic systems selector switch JR1081-1.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE UP ON NORMAL SYSTEM (Cont).		
Electrical power failure (cont).	"F" and "D" of the receptacle. If power is on pin "F" but not on pin "D", verify discontinuity of wiring at the plug C-11 through pins "F" and "D".	Replace hydraulic systems selector switch JR1081-1.
	Obtain access to terminal strip E6 in the left-hand wing root area. Make sure the wing flap control circuit breaker is engaged and that the 28-volt, DC main bus is energized. Move the flap handle to flaps up. Check for 28-volt, DC power at terminal "5" of terminal strip E6. If power is not on that point, remove the connector from the follow-up control switch and verify continuity from terminal 5 of E6 to pin H of the connector.	Replace follow-up control switch JR1052.
	Obtain access to terminal strip E6 in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps up. Check for 28-volt, DC power at terminals "5" and "20" of terminal strip E6. If power is on terminal "5" but not on terminal "20", obtain access to the L.E. follow-up switch and remove the connector from the switch section. Verify 28-volt, DC power at pin "C" and continuity from pin "A" to terminal "20" of E6.	Replace follow-up switch JR1051
	Obtain access to terminal strip E6 and the L.E. flap safety relay in the L. H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is energized. Move the flap handle to flaps up. Check for 28-volt, DC power at terminals 3, 19, and 20 of terminal strip E6. If power is on terminal 20 but not on terminal 3, verify that power is not on terminal 19.	Replace flap disconnect reset relay MS25268-D1
	Obtain access to the L.E. wing flap safety relay in the L.H. wing root area. Make sure the wing flap control circuit breaker is in and that the 28-volt, DC main bus is ener-	Replace the safety relay AN3311-2.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LEADING EDGE FLAPS FAIL TO MOVE UP ON NORMAL SYSTEM (Cont).		
Electrical power failure (cont).	gized. Move the flap handle to flaps up. Check for 28-volt, DC power at terminals B1, B2 and X1 of the safety relay. Verify power to be on X1 and B1 but not on B2.	Replace the safety relay AN3311-2.
	Obtain access to the L.E. wing flap safety relay and the L.E. wing flap control valve in the L.H. wing root area. Close the wing flap control circuit breaker and move the flap handle to flaps up. Remove the connector from the control valve. Verify circuit continuity from pin "C" to ground and 28-volt, DC power on pin "A."	Replace JH1013 wing flap control valve.
TRAILING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM.		
	Note The following tests are arranged in a sequence that will progressively eliminate sections of the system as the cause of a malfunction.	
Asymmetry control tripped.	Obtain access to the asymmetry controller in the L.H. wing root area. Observe the "T.E. tripped" light on. Remove the T.E. connector from the controller. Check misaligned or defective sensor in accordance with paragraph 27.4.6.	Realign or replace sensor as indicated by test.
	If sensors are aligned correctly as determined above, reconnect the asymmetry controller to the system and observe the warning light during flap operation. If the tripped condition persists, inspect the flaps and drive tubes for indication of failure which will cause asymmetric condition.	Repair mechanical failure.
	If no mechanical asymmetry condition exists, as determined above, and if tripped condition persists, the controller is defective.	Replace controller.
Electrical power failure.	Obtain access to the flap follow-up control switch on the center pedestal. Remove the connector from the control switch and confirm no continuity from pin "G" to pin "F" and/or pin "C" when handle is moved to the landing and the take-off positions respectively.	Replace the flap follow-up control switch, JR1052.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	Obtain access to the flap follow-up control switch on the center pedestal and the flap follow-up switch on the R.H. inboard T.E. flap. Remove the control switch connectors from both switches. Check for open circuit in the aircraft wiring from pin "C" of one connector to pin "C" of the other, and from pin "F" to pin "F" of each connector.	Repair aircraft wiring as indicated by test.
	Verify that the wing flap control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down and determine that 28-volt, DC power is not on terminal 4 of terminal strip H9. Remove the connector on the flap follow-up switch on the R.H. T.E. flap. Verify that the circuit from pin "F" and/or pin "C" to pin "H" on the switch receptacle is open.	Replace the follow-up switch, JR1051.
	Determine that 28-volt, DC power is not on terminal 4 of terminal strip H9 as in preceding step. Remove the switch connector from the flap follow-up switch and verify that the circuit is open between pin "H" of the connector and terminal "4" of H9 terminal strip.	Repair aircraft wiring as indicated by test.
	Remove wires C66C20 and C68A20 from terminal strip H9 terminals 4 and 5. Determine that no 28-volt, DC power is on terminal 3 of H9 with all flaps system power on. Verify that the circuit is open between terminals 4 and 5 of terminal strip H9.	Replace the flap disconnect reset relay in the R.H. wing root.
	Obtain access to terminal strips F3 and H9. Verify that the wing flap control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the flap handle to flaps down. Determine that 28-volt, DC power is on terminal "5" of terminal strip H9 but not on terminal "26" of terminal strip F3. Engage the auxiliary hydraulic pump control circuit breaker and observe that the pump does not start.	Replace the flap safety relay and/or the flap select relay.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE DOWN ON NORMAL HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	Under the same conditions as in the preceding step, observe that 28-volt, DC power is on terminal "26" of F3 and that the electric hydraulic pump is running. Disconnect the plug on the wing flap selector relay for system No. 1 and determine that 28-volt, DC power exists on pin B, and that a ground exists on pin "C".	Replace the wing flap selector valve, JH1013.
TRAILING EDGE FLAPS FAIL TO MOVE UP ON NORMAL HYDRAULIC SYSTEM.		
Asymmetry control tripped.	Obtain access to the asymmetry controller in the L.H. wing root area. Observe the "T.E. tripped" light on. Remove the T.E. connector from the controller. Check for misaligned or defective sensor in accordance with paragraph 27.4.6.	Realign or replace sensors as indicated by test.
	If sensors are aligned correctly as determined in preceding step, reconnect the asymmetry controller to the system and observe the warning light during flap operation. If the tripped condition persists, inspect the flaps and drive tubes for indication of failure which will cause asymmetric condition.	Repair mechanical failure.
	If no mechanical asymmetric condition exists, as determined in preceding step and if tripped condition persists, the controller is defective.	Replace controller.
Electrical power failure.	Verify that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Remove the connector from the flap follow-up control switch on the center pedestal and determine that power is on pin "G" of the connector. Set the flap handle in the up position and verify that continuity does not exist between pins "G" and "A" of the receptacle.	Replace the flap follow-up control switch, JR1052.
	Obtain access to terminal strip G3 in the R.H. wing fillet area. Verify that the wing flap L.E. and T.E.	Replace the T.E. flap safety relay, MS25268-D1.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE UP ON NORMAL HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the flap handle to the flaps up position and determine that 28-volt, DC power is present on terminal "21" of terminal strip G3. Move the flap handle alternately down to up and observe that the T.E. safety relay does not click (operate).	Replace the T.E. flap safety relay, MS25268-D1.
	Obtain access to the flap follow-up switch on the R.H. T.E. flap. Remove the connector from the switch section. Determine that the wing flap L.E and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Ascertain that 28-volt, DC power exists on pin "A" of the follow-up switch connector and that continuity does not exist between pins "A" and "G" of the switch receptacle.	Replace the flap follow-up control switch, JR1051.
	Obtain access to terminal strip H9 in the R.H. wheel well. Determine that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the flap handle to the flaps up position. Ascertain that 28-volt, DC power exists on terminal "6" but not on terminal "7" of terminal strip H9 and that no power exists on terminal "3" of H9 (asymmetry signal).	Replace the flap disconnect reset relay in the R.H. wing fillet area.
	Determine that the wing flap L.E. T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Operate the flap handle from flaps down to up several times and note that the T.E. flap safety relay clicks (operates). Return the flap handle to the flaps down position. Disconnect the ground wire marked B3 on the safety relay. Remove the connector from the wing flap selector valve (System No. 1) and determine continuity exists from the ground wire to pin "A" of the selector valve plug. Ascertain that a good ground exists on pin "C" of	Replace the wing flap selector valve, JH1013.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE UP ON NORMAL HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	the selector valve plug. Replace the ground wire B3 on the safety relay and move the flap handle to flaps up. Determine that 28-volt, DC power exists on pin A of the selector valve connector.	Replace the wing flap selector valve, JH1013.
	Use the same setup and procedures as in the preceding step. As the final check, determine the 28-volt, DC power does not exist on pin "A" of the selector valve connector and does exist on terminal "7" of terminal strip H9.	Replace the wing flap selector relay or T.E. flap safety relay (as determined by continuity check of interconnecting wiring).
TRAILING EDGE FLAPS FAIL TO MOVE DOWN ON STANDBY HYDRAULIC SYSTEM.		
	Note This test assumes that flaps operate normally on the normal hydraulic system.	
Electrical power failure.	Verify that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Operate the hydraulic system selector switch alternately from No. 1 and No. 2 system and observe that the wing flap selector relay does not click (operate). Obtain access to the main instrument panel disconnect C-11 and disconnect the connector. Determine that continuity does not exist between pins "F" and "E" of the receptacle when the switch is in the standby position.	Replace the hydraulic system selector switch, JR1081.
	Use the same procedure as in the preceding step. Determine that continuity does exist between pins "F" and "E" of the receptacle but that the impedance between pin "E" of the plug and ground is not 300-400 ohms. (Open circuit or short circuit.)	Replace the wing flap select relay, MS25268-D1.
	On airplanes 5046 and up, determine that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the system selector switch to standby and the flap handle to flaps down. Note that the wing flap selector relay clicks (operates), that no power exists on terminal "24" of	Replace the wing flap select relay or the safety relay.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE DOWN ON STANDBY HYDRAULIC SYSTEM (Cont).		
Electrical power failure (cont).	terminal strip G3 and that power does exist on terminal "5" of terminal strip H9. Show that power is not on terminal "3" of H9 and not on terminal "19" of E6 (L.E. and T.E. asymmetry signals).	Replace the wing flap select relay or the safety relay.
	On airplanes 5046 and up, determine that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Disconnect the electrical connector on the wing flap selector valve - system 2. Move the flap handle to flaps down (system selector switch on standby) and ascertain that 28-volt, DC power is on terminal 24 of terminal strip G3 and is not on terminal 7 of terminal strip E6. Also show that power is not on terminal 19 of terminal strip E6.	Replace the L.E. flap disconnect reset relay in the L.H. wing fillet area.
	On airplanes 5046 and up, use the same procedure as in the preceding step, and in addition remove the connector from the wing flap selector valve - system 2. Determine that 28-volt, DC power is on pin "B" and that a good ground exists on pin "C".	Replace the wing flap selector valve - system 2.
	On airplanes 5001 thru 5045, use the following procedure to detect faulty relays: Determine that the wing flap L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the system selector switch to standby and the flap handle to T.E. flaps down. Note that the wing flap selector relay clicks (operates). Disconnect the electrical connector from the wing flap selector valve system 2. Determine that 28-volt, DC power is not on pin "B" of the connector.	Replace the wing flap selector relay or the T.E. flap safety relay.
	On airplanes 5001 thru 5045 use the following procedures to detect a faulty valve. Use the same set-up as in the preceding paragraph. Determine that 28V DC power is on pin "B" of the connector and that a good ground exists on pin "C".	Replace the wing flap selector valve - system 2.

27.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
TRAILING EDGE FLAPS FAIL TO MOVE UP ON STANDBY HYDRAULIC SYSTEM.		
	Note This test assumes that flaps operate normally on the normal hydraulic system and move down normally on the standby system.	
Electrical power failure.	Verify that the wing L.E. and T.E. control circuit breaker is closed and that the 28-volt, DC main bus is energized. Move the flap handle to flaps up. Determine that 28-volt, DC power exists on terminal 7 of terminal strip E6 and does not exist on terminal 3 of terminal strip H9 (asymmetry signal). On airplanes 5001 thru 5045, disconnect the electrical connector from the wing flap selector valve and show that no power exists on pin "A". Determine that the safety relay is operating (audible click) when the flap handle is moved from down to up. On airplanes 5046 and up, show that no 28-volt, DC power exists on terminal 25 of G3 and that the safety relay is operating (audible click).	Replace the wing flap select relay or the T.E. flap safety relay.
	On airplanes 5046 and up, use the same setup as in the preceding step. Determine that 28-volt, DC power does exist on terminal 25 of terminal strip G3 but does not exist on terminals 23 or 19 (asymmetry) of terminal strip E6.	Replace the L.E. flap disconnect reset relay.
	Use the set-up as in the first step of this test. Remove the electrical connector from the wing flap selector valve - system 2 and show that 28-volt, DC power exists on pin "A" of the connector and that a good ground exists on pin "C." On airplanes 5046 and up, determine that 28-volt, DC power does exist on terminal 23 of terminal strip G3 and does not exist on terminal 19 of E6 (asymmetry). Remove the electrical connector from the wing flap selector valve - system 2. Show that 28-volt, DC power exists on pin "A" of the connector and that a good ground exists on pin "C."	Replace the wing flap selector valve - system 2.

27.4.4 REMOVAL AND INSTALLATION.

27.4.4.1 REMOVAL OF THE INBOARD LEADING EDGE FLAP.

Note

On airplanes 5046 and up, the outboard flap must be removed before the inboard flap can be removed.

a. Lower the leading edge flaps to the fully-down position, and remove the flap shrouds from over the inboard flap actuators.

b. Disconnect the flap actuator rod ends from the flap fittings on the inboard flap.

c. Disconnect the de-icer line, the followup switch linkage, the flap asymmetry switch cable, and all bonding straps from the inboard flap. (See paragraph 27.4.4.19 for disconnecting asymmetry cable.)

d. Remove the stop bolts from the wing front beam fittings, and lower the flap.

e. Remove the lower fairing from the external fuel tank to provide clearance for pulling the inboard flap hinge pin.

f. Remove the hinge half segment at leading edge flap station 204; while supporting the inboard flap, remove the hinge pin by pulling it inboard, and remove flap from the airplane.

27.4.4.2 INSTALLATION OF THE INBOARD LEADING EDGE FLAP.

a. Place a small amount of Lubriplate No. 130A in the holes of the hinge nodes so that hinge pin insertion will carry the lubricant into the hinges. Reverse the removal procedures; after hinge pin insertion, lubricate the hinges with Specification MIL-L-7870 lubricant while moving the flap throughout its full range of travel. Wipe off excess lubricant with clean cloths.

b. Adjust the torque of the rod-end attachment bolts to 0 to 30 inch-pounds. Grease fittings on actuator rod ends should face upwards.

c. Rig the leading edge flap system as required in accordance with paragraph 27.4.5.1.

d. Perform an operational checkout of the leading edge flap system, and check operation of the de-icer boots.

e. Secure all connections and replace the flap shrouds.

f. Aerodynamically seal the flap shrouds according to the instructions in this section.

27.4.4.3 REMOVAL OF THE OUTBOARD LEADING EDGE FLAP.

a. Lower the leading edge flaps to the fully-down position, and remove the flap shrouds from over the outboard-flap actuators.

b. Disconnect the de-icer lines, the asymmetry switch cable, and all bonding straps from the outboard leading edge flap.

c. Disconnect the actuator rod ends from the flap fittings of the outboard flap.

d. Remove the lower tank-to-wing fairing so that the hinge pin of the inboard flap can be pulled inboard.

e. Remove the hinge half segment at leading edge flap station 204.

f. Slide the inboard flap hinge pin inboard until it clears the outboard flap hinges.

g. Remove the screws from the hinge segment at leading edge flap stations 313 and 375 by sliding it inboard.

h. Pull the hinge pin between leading edge flap stations 258 and 313 by sliding it outboard, and remove flap. On airplanes 5046 and up, lower the flap slightly, to clear hinge; move outboard to disengage alignment pin from inboard flap, and remove flap.

27.4.4.4 INSTALLATION OF THE OUTBOARD LEADING EDGE FLAP.

a. Place a small amount of Lubriplate No. 130A in the holes of the hinge nodes so that inserting the hinge pin will carry lubricant into the hinges. Reverse the removal procedures; after hinge pin insertion, lubricate the hinges with Specification MIL-L-7870 lubricant while moving the flap throughout the full range of travel. Wipe off excess lubricant with clean cloths.

b. Adjust the torque of the rod end attachment bolts to 0 to 30 inch-pounds. The grease fittings in the rod ends should face upwards.

c. Rig the leading edge flap system as required in accordance with paragraph 27.4.5.1.

d. Perform an operational checkout of the leading edge flap system, and check the operation of the de-icer boots.

e. Secure all connections and replace the flap shrouds.

f. Seal the flap shrouds aerodynamically according to the instructions in this section.

27.4.4.5 REMOVAL OF TRAILING EDGE FLAP ASSEMBLY INBOARD AND OUTBOARD.

a. Remove fairings from flap lower surface at the flap supports.

b. Remove flap-to-external tank fairing in accordance with paragraph 27.4.4.7.

c. Disconnect the limit switch actuator rod (if right-hand inboard flap is to be removed), asymmetry potentiometer cable and all bonding jumpers.

d. Support the flap and remove the two bolts at each of the two flap supports. Remove the flap from the airplane. Retain all hardware fairings etc., if serviceable for use on the replacement unit.

27.4.4.6 INSTALLATION OF TRAILING EDGE FLAP ASSEMBLY INBOARD AND OUTBOARD.

a. Lubricate all bearings and bolts prior to installation with Specification MIL-G-23827 grease.

b. Support the flap assembly, align holes in support fittings and install bolts.

c. If right-hand inboard flap, attach limit switch actuator rod. Attach asymmetry potentiometer cable and bonding jumpers.

d. Rig flaps as necessary. If left-hand inboard, L.H. outboard or R.H. outboard flap has been replaced and no other operations performed that would affect rigging adjustment of the affected flap to fair with the wing in the up position, balance the asymmetry potentiometers in accordance with figure 27-24, steps 11 and 23 (the only rigging required). If R.H. inboard flap has been replaced, it will be necessary to check complete rigging of trailing edge flaps system in accordance with figure 27-24.

e. Install flap-to-external tank fairing and fillets in accordance with paragraph 27.4.4.8.

f. Install fairings on flap lower surface at supports.

27.4.4.7 REMOVAL OF EXTERNAL TANK - FLAP FAIRING.

a. Extend flaps to landing position.

b. Defuel the affected external tank in accordance with Section 28.4.

c. Remove the fuel drain tube (JWT50-4) which extends through the fairing. Note location of seals for reinstallation.

d. Remove the fairing-to-flap fillets from both sides of fairing.

e. Remove screws from fairing support angles and remove fairing.

f. Retain all hardware and parts removed for access, if serviceable for installation with the replacement part.

27.4.4.8 INSTALLATION OF EXTERNAL TANK - FLAP FAIRING.

a. Retract trailing edge flaps to "UP" position.

Note

Flaps must be rigged in accordance with figure 27-24, and must be in flap up position when fairing is fitted.

b. Position the fairing and attach angles on flap assembly as required to fair with external tank trim ends, and locate and drill holes in attachment angles to match holes in flap surface and in fairing assembly. Before drilling any holes, check flap and fairing for clearance by extending flaps to landing position. Trim edges where necessary to fit external tank and to clear tank structure as flaps are extended.

c. Install the JWT 50-4 fuel drain tube.

d. Install attaching screws in fairing attachment angles.

e. Retract flap to trailing edge up, and trim and fit fillets.

f. Drill fillet attachment holes to match holes in flap and fairing assembly.

g. Install fillet attaching screws.

27.4.4.9 REMOVAL OF HYDRAULIC MOTORS.

a. Remove access panel in wing lower fillet area (L.H. side at wing leading edge if leading edge flap drive motor is to be removed, and R.H. side at wing trailing edge if trailing edge flap motor is to be removed). (See Chapter 52.)

b. Disconnect hydraulic tubes from the motor pressure, return, case drain and shaft seal drain ports. Cap or plug all tubes and ports in motor.

c. Remove four screws from motor flange and remove motor. Use caution to prevent damage to gearbox breather tube which is attached to one of the screws.

d. After motor splines are disengaged, apply a light coat of Specification MIL-G-23827 grease on the gear box input spline. Cover input spline with tape to prevent entry of foreign matter.

e. Install plastic cup or equivalent over motor output splines.

27.4.4.10 INSTALLATION OF HYDRAULIC MOTORS.

a. Position the motor to be installed. Align the pressure, return case drain, and shaft seal drain ports with the corresponding tubes.

b. Engage the motor output spline with the gearbox input spline, and push the flange of the motor to the gearbox. Assure that the raised boss on the motor is seated in the recess on the gearbox before installing the screws. Do not use excessive force.

c. Install four screws in the motor attaching flange. Assure that the gearbox breather tube support bracket is attached with the appropriate screw. Safetywire the screws.

d. Remove protective caps from the tubes and motor ports, and connect tubes to motor in accordance with the instructions for installing tubes contained in Chapter 29.

e. Service the hydraulic system in accordance with Chapter 12.

f. Perform an operational checkout of flap system.

Note

No rerigging is required if motor only is replaced.

27.4.4.11 REMOVAL OF LIMIT VALVE.

a. Remove the access panel in wing lower fillet area (L.H. side at wing leading edge if leading edge flap limit stop is to be removed, and R.H. side at the wing trailing edge if trailing edge flap limit stop is to be removed). (See Chapter 52.)

b. Disconnect hydraulic tubes from the valve pressure, return and drain ports. Cap or plug all tubes and ports in the valve.

c. Remove safety wire, remove four screws from valve attaching flange, and remove the valve.

27.4.4.12 INSTALLATION OF LIMIT VALVE.

a. Position the valve with the attaching holes aligned with the holes in gearbox.

b. Rotate the valve drive lever as required for the seat to align with the pin in the valve drive lever on the gearbox.

c. Push the valve against the mounting surface on the gearbox, and check for clearance between the valve drive levers on the gearbox and the valve, and for engagement of the drive lever pin.

d. Install four screws and safetywire.

e. Rig flaps (L.E. or T.E., as applicable) in accordance with rigging instructions. (Refer to paragraph 27.4.5.1.)

f. Perform an operational checkout of the applicable system (L.E. or T.E.) in accordance with paragraph 27.4.2.

27.4.4.13 REMOVAL OF FLAP DRIVE GEARBOX. To remove the gearbox, it is necessary to remove the hydraulic motor and hydraulic limit valve assembled to the gearbox.

a. Remove the access panel as required for access to the gearbox (refer to chapter 52).

b. Disconnect the hydraulic tubing from the hydraulic motor pressure, return, case drain, and shaft seal drain ports, and cap or plug all tubes and ports in the motor.

c. Disconnect the hydraulic tubing from the hydraulic limit valve pressure, return and drain ports; cap or plug the tubes and ports.

d. Disconnect the torque tube universal joint from the gearbox output shaft on the motor side only. (The other shaft can be disengaged as the gearbox is removed from the airplane.)

e. While supporting the gearbox, remove the four bolts attaching the gearbox to structure.

f. Lower the gearbox, motor and limit valve assembly from the airplane. Make sure the first movement is in the direction that will disengage the remaining universal joint splines from the other output shaft. Note the positions of motor and limit valve for use on replacement unit.

g. Remove the hydraulic motor and limit valve from the gearbox in accordance with paragraph 27.4.4.9 and 27.4.4.11, and retain for reinstallation on the replacement gearbox assembly. Reinstall the motor and limit valve attaching screws in the gearbox. Remove the spacers from the replacement unit (between the gearbox housing and breather tube support clip), install on the removed unit, and tighten the clip support screw.

27.4.4.14 INSTALLATION OF FLAP DRIVE GEARBOX. The replacement gearbox is equipped with attaching screws for the hydraulic motor and limit valve. Remove the screws from the gearbox or from the attached bag, as applicable.

a. Install the hydraulic motor on the gearbox in accordance with paragraph 27.4.4.10.

b. Install the limit valve on the gearbox in accordance with paragraph 27.4.4.12.

c. Position the assembled components with the gearbox attaching holes aligned with the holes in

27.4.4.14 (Cont).

the structure. Make sure the output shaft splines are engaged with the splines on the torque tube universal joint. Refer to paragraph 27.4.7.

d. Install the four bolts attaching the gearbox to structure and safetywire.

e. Connect the torque tube universal joint to the gearbox output shaft. Refer to paragraph 27.4.7.

f. Connect the hydraulic tubes to the motor and limit valve ports in accordance with instructions for installing tubes contained in Chapter 29.

g. Service the hydraulic system in accordance with Chapter 12.

h. Rig the applicable (L.E. or T.E.) flap system in accordance with paragraph 27.4.5.1.

i. Perform an operational checkout of the applicable flap system (L.E. or T.E.) in accordance with paragraph 27.4.2.

27.4.4.15 REMOVAL OF LEADING EDGE FLAP ACTUATOR.

a. Extend the leading edge flaps to the full down position and remove the access plates and shrouds as required to gain access to the applicable flap actuator (refer to Chapter 52).

b. Disconnect the actuator from the flap by removing the bolt from the actuator rod end.

c. Disconnect the drive torque tube from both sides of the actuator by removing screws and sliding the universal joint onto the tube as required to disengage the splines.

d. Remove the four bolts from each of the two actuator support fittings on the wing front beam, and remove the fittings. Retain bolts and fittings for reuse if serviceable.

e. Remove the actuator.

Note

If the remaining actuator on the affected flap is not actuated or removed and no other torque tubes are disconnected, it may be possible to remove and replace a single actuator without disturbing the rigging.

27.4.4.16 INSTALLATION OF LEADING EDGE FLAP ACTUATOR. Lubricate the rod end bearing and apply a light coat of Specification MIL-G-23827 grease to the splines and trunnions prior to installation.

a. Position the actuator with the rod end installed in the fitting in the flap and trunnions supported by fittings. Install the four attaching bolts in each of the two trunnion support fittings. These are adjusted by moving inboard or outboard as required to provide 0.0 to 0.03 inch clearance between the shoulder on the actuator trunnion and the fork of the support fitting. The actuator should be free to rotate in the trunnion support fittings after the fitting attaching bolts are installed and tightened.

b. Rotate the actuator input splined shaft (with torque tubes still disconnected) as required to align the rod end with the hole in the flap fitting, and install the bolt. Coat the bolt lightly with Specification MIL-G-23827 grease prior to installation.

c. Engage the torque tube universal joints with the actuator input splines, and install the screws common to the universal joint and tube. Refer to paragraph 27.4.7.

Note

Find the position in which the screw holes align and the splines engage without forcing rotation.

d. If flap position was maintained by the other actuator during replacement, and no other operations were performed than those described above, no re-rigging will be required.

e. Perform an operational checkout of the leading edge flap system in accordance with paragraph 27.4.2.

27.4.4.17 REMOVAL OF TRAILING EDGE FLAP ACTUATOR.

Note

The appropriate flap assembly must be removed from the airplane in order to remove an actuator.

a. Extend the flap to landing position and remove the external tank-to-flap fairing in accordance with paragraph 27.4.4.7.

b. Remove the appropriate flap in accordance with paragraph 27.4.4.5.

c. Retract the flaps to full up position, then disconnect the torque tube universal joints from the input shafts of the applicable actuator, and disconnect the closure door support link from the forward flap support fitting.

d. While supporting the actuator support assembly, remove the four bolts attaching the assembly to the two trailing edge ribs.

e. Remove the support assembly from the airplane.

f. Remove the bolt from the actuator rod end. It may be necessary to extend or retract the actuator

27.4.4.17 (Cont).

to align the bolt with the hole in the support fitting to permit its removal.

g. Remove the twelve bolts, six on each side, from the actuator trunnion support fittings, and remove the fittings and actuator. Retain all attaching hardware and fittings, if serviceable, for use on the replacement unit.

27.4.4.18 INSTALLATION OF TRAILING EDGE FLAP ACTUATOR. Lubricate the actuator rod end bearing and apply a light coat of Specification MIL-G-23827 grease to the splines and trunnion prior to installation.

a. Position the actuator between the trunnion support fittings, and engage the trunnion fittings with the main support fitting. Align the six attaching holes in each fitting.

b. Install the twelve, (six each side) trunnion attaching screws. After the screws are tightened, check and make sure the actuator will rotate freely about the trunnions.

c. Rotate the input splined shaft of the actuator as required to align the rod end bearing with the clearance hole in the main support fitting, and install the rod end bolt and cotter pin.

d. Position the actuator and support assembly between the two trailing edge ribs, align the attaching holes, and install the four attaching bolts.

e. With the torque shafts not connected to the actuator, rotate the actuator splined input shaft as required to obtain "V" dimension in accordance with figure 27-24, detail G. Do not disturb this position until after step (f) is completed.

f. Connect the torque tube universal joints to the input splined shafts of the actuator. Do not rotate the torque shafts or the actuator input shaft except the minimum amount required to align the splines. Refer to paragraph 27.4.7.

g. Extend the flaps to the "LANDING" position, and install the flap in accordance with paragraph 27.4.4.6.

h. Retract the flaps, and check the flap for alignment with the wing trailing edge at the end where the actuator was replaced. If required, disconnect the two universal joints from the actuator input spline shaft and rotate the actuator shafts as required to align the flap with the fixed trailing edge surfaces within .06 inch. Replace the external tank-to-flap fairing in accordance with paragraph 27.4.4.8.

i. If the replaced actuator was on the L.H. inboard, L.H. outboard or R.H. outboard flaps, proceed to step k.

j. If the replaced actuator was on the R.H. inboard flap which drives the flap follow-up switch assembly, the trailing edge flap drive system will require re-rigging in accordance with figure 27-24.

k. Perform an operational checkout, of the trailing edge flap system in accordance with paragraph 27.4.2.

27.4.4.19 REMOVAL OF LEADING EDGE FLAP ASYMMETRY POTENTIOMETER.

CAUTION

Do not attempt to cut wires without removing power from the system. Open WING FLAPS "ASYM 26 VAC," "L.H. & T.E. CONT" and "L.E. POS" circuit breakers or disconnect electrical connectors from the asymmetry system control box and attach safety tags.

a. Gain access to the potentiometer location by removing the appropriate access panels in accordance with chapter 52.

b. On airplanes 5046 and up or on airplanes 5001 thru 5045 having Service Bulletin 329-140 accomplished, loosen jam nut (68) (refer to figure 27-24, detail M). Rotate the pulley by using a spanner wrench in the holes provided, until the cable is slack, then rotate the barrel fitting (69) until it is disengaged from the trunnion fitting. Release the pulley gradually until it is in contact with its travel limit stop. On airplanes 5001 thru 5045 on which Service Bulletin 329-140 has not been accomplished, rotate the pulley by using a spanner wrench in the holes provided until the cable is slack, then remove the four attaching screws from the two (2) brackets; refer to figure 27-24, details j and k. Remove the brackets from the cable end fitting. Retain the brackets and screws if serviceable for use on the replacement unit. Release the pulley gradually until it is in contact with its travel limit stop.

c. Note routing of wires through the tubes and clearance holes in fittings, and label wires if desired for use in routing and splicing wires of replacement part, then cut the three wires as close to the existing splices as possible.

d. Remove the three screws from the potentiometer and mounting bracket, and remove the potentiometers. Retain hardware, if serviceable, for use on replacement unit.

27.4.4.20 INSTALLATION OF LEADING EDGE FLAP ASYMMETRY POTENTIOMETER.

CAUTION

Each potentiometer assembly is equipped with an internal spring which holds tension on the cable at all times when installed. Do not permit the spring to snap the pulley into the travel limit stop, as internal damage to the assembly will result.

a. Position the potentiometer assembly in its mounting bracket and install the three screws.

b. Route the wires in accordance with the routing of those on the replaced unit, and splice to the system wires, using the appropriate size insulated splice. Remove temporary identification tags if used.

c. Rotate the pulley as required to keep the cable slack by using a spanner wrench in the holes provided on face of pulley. On airplanes 5046 and up, and on airplanes 5001 thru 5045 if Service Bulletin 329-140 is accomplished, install barrel fitting (69) in the trunnion fitting (see figure 27-24, detail M). On airplanes 5001 thru 5045 on which Service Bulletin 329-140 is not incorporated, position the cable end attaching brackets and install the screws.

d. Balance the potentiometer with its corresponding opposite hand unit in accordance with figure 27-24, step 20.

27.4.4.21 REMOVAL OF TRAILING EDGE FLAP ASYMMETRY POTENTIOMETER.

CAUTION

Do not attempt to cut wires without removing power from the system. Open WING FLAPS "ASYM 26 VAC," "L.E. & T.E. CONT" and "L.E. POS" circuit breakers or disconnect both electrical connectors from the asymmetry system control box, and attach safety tags.

a. Gain access to the potentiometer location by removing the appropriate access panel in accordance with chapter 52.

b. Rotate the pulley by using a spanner wrench in the holes provided, until the cable is slack, lift the cable out of the pulley groove, remove the cable ball end from the seat in the pulley, and release the pulley gradually. Do not allow it to snap into the travel limit stop.

c. Remove the cable assembly from the flap. Note the arrangement of bushings, washers, etc., and retain for use on replacement unit if serviceable.

d. Note routing of the three electrical wires, and attach temporary identification tags, if desired for use in routing and splicing wires on the replacement unit, then cut wires as close as possible to the existing splices.

e. Remove the three screws from the potentiometer mounting flange, and remove the potentiometer from the airplane. Retain hardware, if serviceable, for use on replacement.

27.4.4.22 INSTALLATION OF TRAILING EDGE FLAP ASYMMETRY POTENTIOMETER.

CAUTION

Each potentiometer assembly is equipped with an internal spring which holds tension on the cable at all times when installed. Do not permit the spring to snap the pulley into the travel limit stop, as internal damage to the assembly will result.

a. Position the potentiometer assembly in its mounting and install the three screws.

b. Route the wires in accordance with the routing of those on the replaced unit, and splice to the system wires, using the appropriate size insulated splice. Remove temporary identification tags if desired.

c. Install the replacement cable assembly on the flap. Assure that spacers and washers are installed in accordance with the replaced unit.

d. Rotate the pulley as required to install the cable ball in its seat, and install the cable in the grooves as the pulley is allowed to gradually rotate back until the cable is taut.

e. Balance the potentiometer with its corresponding opposite hand unit in accordance with figure 27-24, step 20.

27.4.4.23 REMOVAL OF LEADING EDGE FLAP LIMIT SWITCH ASSEMBLY.

Note

Open flap system circuit breakers.

a. Operate the flaps to the up position, and do not move them until after the unit is removed and the replacement unit is installed, to minimize re-rigging.

27.4.4.23 (Cont).

b. Gain access to the limit switch assembly by removing the upper surface trailing edge access panel and shroud from the L.H. inboard flap assembly in accordance with chapter 52.

c. Disconnect the electrical connector from the limit switch assembly.

d. Disconnect the drive lever from the switch input splined shaft. Do not disconnect the opposite end of the input lever from the rod end.

e. Remove four screws from the mounting bracket, and remove the unit. Retain hardware for use on replacement unit if serviceable.

27.4.4.24 INSTALLATION OF LEADING EDGE FLAP LIMIT SWITCH ASSEMBLY.

Note

Open flap system circuit breakers.

a. Position the limit switch assembly on the mounting bracket with the index mark on the housing directly above the input shaft, align the attaching holes, and install the four screws. (See figure 27-24, detail A.)

b. With the flaps in the up position, find the input shaft position at which the up limit switch is open, in accordance with figure 27-24, step 11, then engage the input lever with the splines.

c. Connect the electrical connector.

d. Perform a check of rigging in accordance with figure 27-24, steps 11, 12 and 13.

e. Perform an operational checkout of the leading edge flap system in accordance with paragraph 27.4.2.

27.4.4.25 REMOVAL OF TRAILING EDGE FLAP LIMIT SWITCH ASSEMBLY.

Note

Open flap system circuit breakers.

a. Operate the trailing edge flaps to the full up position, and do not move them until after the limit switch assembly is removed and the replacement unit is installed, to minimize re-rigging.

b. Disconnect electrical connector from the limit switch assembly.

c. Gain access to the unit by removing the applicable access panels in accordance with chapter 52.

d. Disconnect the drive lever from the switch input splined shaft. Do not disconnect the opposite end of the input lever from the rod end.

e. Remove four screws, and remove the unit from the airplane. Retain hardware if serviceable for use on replacement unit.

27.4.4.26 INSTALLATION OF TRAILING EDGE FLAP LIMIT SWITCH ASSEMBLY.

Note

Open flap system circuit breakers.

a. Position the limit switch assembly with the index mark on the housing directly below the input shaft, align the attaching holes with holes in structure, and install the four screws. (See figure 27-24, detail F.)

b. With the trailing edge flaps in the up position, find the input shaft position at which the up limit switch is open, in accordance with figure 27-24, step 13, then engage the input lever with the splines.

c. Connect the electrical connector.

d. Perform a check of rigging in accordance with figure 27-24, steps 14 and 15.

e. Perform an operational checkout of the trailing edge flap system in accordance with paragraph 27.4.2.

27.4.4.27 REMOVAL OF LEADING AND TRAILING EDGE TORQUE TUBES. The torque tubes are connected to end fittings with screws. Generally, the torque tubes can be removed by removing the screws and sliding the fittings into the tube enough to disengage from splines or other fittings. To remove the leading edge flap drive tube assembly which extends outboard from wing station 41 through the leading edge fuel tank area, it is necessary to withdraw the complete assembly from the tunnel at the outboard end of the leading edge tank area. (Wing station 114.76, approximately.) It is not necessary to defuel the wing internal fuel to remove this assembly. Remove as follows:

a. Remove the external tank assembly in accordance with Section 28.2.

b. Extend the leading edge flaps, and remove shrouds in the inboard flap area (airplanes 5001 thru 5045).

c. Remove screws from the torque tube universal joint at wing station 140 approximately, and disengage the universal coupling from the splined fitting. On airplanes 5046 and up, remove screws from the sleeve at wing station 120 approximately.

27.4.4.27 (Cont).

d. Remove four screws from the flange on the outboard leading edge tank bulkhead, and withdraw the tube assembly from the tunnel. Moderate force may be required to withdraw the assembly, due to friction of the rubber ring on tunnel wall.

Note

The splined coupling at the inboard end of the assembly may become disengaged and fall from the gear box output shaft. Retain it and all hardware if serviceable for use on the replacement assembly.

27.4.4.28 INSTALLATION OF LEADING AND TRAILING EDGE FLAP DRIVE TORQUE TUBES.

Note

When a torque tube assembly is replaced, if caution is used to prevent relative rotation of the remaining torque tubes while the replaced assembly is disconnected, no re-rigging will be required.

a. Generally, a torque tube assembly can be installed by reversing the removal procedure. Refer to paragraph 27.4.7.

b. When installing the assembly through the leading edge tank area, the universal coupling at the outboard side of the flap drive gearbox must be positioned and held for alignment while the tube assembly is pushed into it. Moderate force may be required to overcome friction of the rubber centering ring on the tunnel walls as the assembly goes in.

27.4.4.29 BOLT TORQUE REQUIREMENTS. Bolts used with castellated nuts and cotter pins at the flap actuator locations and in the TE flap mechanism (reference P/N JW90 assemblies) should be tightened to a torque value of at least 10 inch-pounds to a position where the cotter pin can be installed. Torque should not exceed 50 inch-pounds. Other torque values should be in accordance with values referenced in chapter 51, in this manual, unless otherwise specified.

27.4.5 ADJUSTMENT.

(See paragraph 27.4.2 for test equipment and tools required.)

27.4.5.1 WING FLAP RIGGING INSTRUCTIONS.

(See figure 27-24.)

27.4.6 MAINTENANCE AND SERVICING PROCEDURES.

(Refer to chapter 12 for additional servicing information.)

27.4.6.1 LEADING EDGE WING FLAP DOWNSTOP INSPECTION. To determine that the downstops of the leading edge flaps are functioning properly, perform this inspection:

a. Remove the inspection plate over the two flap actuators on the left and right inboard leading edge flaps to provide access to the downstops.

b. Connect external 28-volt, DC power to the airplane. Connect external hydraulic power to the normal (No. 1) and standby (No. 2) hydraulic system ground test connections. Set hydraulic pressure at 3,000 PSI with no flow and 2,700 PSI at 10 gallons per minute. Energize the AC and DC buses.

c. Place the FLAP & BRAKE HYD SYS selector switch in the standby (No. 2) position, and place the flap control lever in the landing position.

d. When leading edge flap movement stops, check the gap between the stop block and the stop bolt (refer to figure 27-24 for location of these components). The gap should be 0.04 to 0.06 inch.

e. Check position of the hydraulic limit valve (shut off) indicator. The indicator should be in the "OFF" position.

f. Place the flap selector lever in the flap "UP" position. When flap movement stops, place the FLAP & BR HYD SYS selector switch in the NORM-1 position.

g. Place the flap selector lever in the leading-edge-down position. Check the gap between the stop blocks and stop bolts. The gap should be .10 to .25 inch.

h. Re-rig the flaps in accordance with rigging instructions in figure 27-24, if the stops are not correctly rigged in accordance with (d) and (g) above.

27.4.6.2 REPAIR OF TEFLON RUB STRIPS.

(See Chapter 51.)

27.4.6.3 AERODYNAMIC SMOOTHING AND SEALING OF LEADING EDGE FLAP SHROUDS. During normal airplane use and maintenance, the sealer around the leading edge flap shrouds may become damaged or removed. Replace damaged or missing sealer according to these procedures. (For airplanes 5001 thru 5009, 5011 thru 5016, 5018 thru 5021, 5023, 5025, 5029, and 5033.) (See figure 27-21.)

a. Lower the leading edge flaps to the full down position, and then remove the sealing compound using a sharp plastic tool.

CAUTION

Do not use knives or chemical sealant stripper.

b. Remove all grease, oil, dirt, and finger prints from the sealing area (see figure 27-21), using clean Specification TT-N-95 aliphatic naphtha and clean cloths. Do not allow the naphtha to evaporate. Wipe the area dry to prevent the residue that appears when naphtha is allowed to evaporate.

27.4.6.3 (Cont).

Note

With the leading edge flaps in the fully-up position, none of the teflon rubbing strips should be visible. Sealing compound will not stick to teflon.

c. Apply parting agent, MIL-S-8660 or equivalent, to the upper aft edge of the leading edge flaps.

d. Mask the upper aft edge of the leading edge flap and the forward edge of the flap shroud. See figure 27-21 for the desired placement of the tape.

e. Apply a bead of compound in front of the flap shroud step, and fully retract the leading edge flaps.

Note

Use STM40-006, Type I, Class A-1/2 or A-1 compound. Class A-1 compound has a work life of 1 hour; Class A-1/2 compound has a work life of 1/2 hour. The curing time of Class A-1 is approximately 24 hours; the curing time for Class A-1/2 is approximately 8 hours. Curing can be accelerated by using heat not exceeding 140°F.

f. Apply polyethylene tape over the compound, pressing the compound firmly into the gap with a spatula to form a smooth, flush surface.

g. After curing, remove the polyethylene tape; then remove the masking tape.

h. If needed, smooth the area using No. 400 sandpaper.

CAUTION

If sanding is necessary, mask the adjacent skin to prevent scratching the aluminum.

27.4.6.4 LEADING EDGE FLAP ACTUATOR ROD END LUBRICATION. Lubricate the leading edge flap actuator rod ends during leading edge flap rigging procedures, and when the leading edge flaps are lowered enough to provide access to the rod ends. To lubricate the rod ends, it is necessary to lower the flaps and disconnect the jackscrews from the leading edge flaps. Lubricate the rod ends according to the following instructions:

a. Remove the access and inspection panels and flap shrouds to gain access to the leading edge flap actuators.

b. While supporting the weight of the flaps, remove the actuator rod end attaching bolts, and then carefully lower the flaps.

CAUTION

Disconnect all attachments to the leading edge flaps as necessary, enough to gain access to the rod ends.

c. Inspect the rod ends and rod end attaching bolts for excessive wear. Replace excessively worn components.

d. Grease the rod ends using a grease gun filled with Specification MIL-G-23827 grease.

e. Lightly grease the rod end attaching bolts with Specification MIL-G-23827 grease to prevent corrosion.

f. Raise the leading edge flaps, and then reconnect the actuator rod ends to the leading edge flaps. The grease fittings on the rod ends should face upwards.

g. Reconnect the attachments disconnected in step (b), and then rig the leading edge flap system. Reinstall the shrouds and access panels. It will not be necessary to rerig the flap system if the actuator rod end is not rotated while the flap is disconnected.

27.4.6.5 FLAP ACTUATOR JACKSCREW LUBRICATION.
(See figure 27-20.) (Refer to para 5.5.13 for time intervals between inspections.)

a. With the leading edge flaps fully-down, and with the access and inspection panels over the actuators removed, cut the safety wire holding the forward boot retainers, and slide the boots back to expose the jackscrew threads.

b. If there is grease on the jackscrew, and if the grease is not dry, fill the exposed threads of the jackscrew with Specification MIL-G-21164 grease, return the boots and boot retainers to their proper positions, and secure the boot retainers with 0.020 to 0.032 stainless steel lockwire.

c. Jackscrews with either no grease or with dry grease on their threads must be removed from the airplane, disassembled, inspected, and regreased or replaced.

27.4.6.6 FLAP ACTUATOR DISASSEMBLY, INSPECTION, LUBRICATION, AND REASSEMBLY.
(See figure 27-20.) (Refer to para 5.5.15 for time intervals between inspections.)

a. With the actuator removed from the airplane, cut the safety wire holding the boot retainers,

27.4.6.6 (Cont).

and remove the boot retainers and jackscrew from the actuator body.

b. Remove the "Spiralox" retaining ring from one side of the actuator, and rotate the worm shaft counterclockwise to pull it out of the actuator body.

Note

It may be necessary to hold the worm gear to remove the worm shaft.

The bearing, bearing retainer, shims, and seals from one side of the assembly should come out with the worm shaft.

CAUTION

Note the disassembly sequence and amount and position of shims. These parts must be assembled in the same relative positions.

c. Clean the worm shaft, and visually inspect the gears and jackscrew threads. A slight bronze tint on the sides of the worm gear is normal. If scoring or galling is evident, replace the actuator assembly. Check that the end play of the jackscrew does not exceed 0.009 inch when fully engaged with the worm gear nut in the body of the actuator. If end play exceeds this amount, replace the actuator.

d. With the worm shaft and jackscrew disassembled, remove as much old grease as possible from the worm cavity and parts. Then, by access through the worm cavity, fill the teeth of the worm gear with Specification MIL-G-21164 grease.

e. Replace outer static seal and worm shaft seal on bearing retainer. Fill worm threads with Specification MIL-G-21164 grease and reassemble.

CAUTION

Be sure the outer race markings of the bearing face out and that the original amount of shimming is between the bearing and the bearing retainer.

f. Reinstall the worm shaft assembly into the body of the actuator assembly, and lock it in place using the "Spiralox" retaining ring.

g. With the jackscrew not installed, use a drill motor to drive the input shaft for approximately 30 seconds.

h. Wipe off excessive grease and screw the jackscrew into the actuator body. If the jackscrew becomes hard to turn before it bottoms, remove the jackscrew. Make a paper tube to fit inside the jackscrew bore, insert this tube into the actuator body, and extract excess grease. Repeat this procedure until the jackscrew can be bottomed without excessive force.

i. With the jackscrew and the worm shaft assembled in the actuator body, and with the jackscrew boot and boot retainers properly positioned, lock the retainers in place using 0.020 to 0.032-inch stainless steel wire.

27.4.6.7 INSPECTION FOR EXCESSIVE LOOSENESS IN TRAILING EDGE WING FLAPS.

Note

Any bearings with radial play or axial play more than 0.015 inch should be replaced.

a. If total flap motion, measured at the trailing edge with the flaps up, exceeds 0.25 inch when the trailing edge is moved by hand, further inspection of the individual linkage bearings or flap rerigging is required.

Note

Regardless of measurements, if the motion in the landing position exceeds the motion in the up position by a factor of 3 or greater, inspect the flap actuators in accordance with Paragraph 27.4.6.6.

b. If total flap motion, measured at the trailing edge with the flaps in the landing position, exceeds 0.62 inch when the trailing edge is moved by hand, further inspection of the individual linkage bearings or flap rerigging is required.

27.4.6.8 LEADING EDGE AND TRAILING EDGE FLAP FOLLOWUP SWITCHES. The following tables are provided for ascertaining that the internal switches are serviceable and contacts are in the proper position corresponding to degree of shaft rotation. In the event one or more of the internal switch contacts do not make or break at the specific shaft position, this can be remedied by loosening the mounting screws and adjusting the switch position as required.

Leading Edge (JR1051-5)

Shaft Position Measured from Full-Up Position	Direction of Shaft Rotation	Continuity Between Pins	Open Between Pins
0 Degrees (full up)	CW	H-K, T-R, F-B	C-A
0.5 to 4.5*	CW	C-A, H-K, T-R, F-B	
21.5 to 25.5*	CW	C-A, F-B	H-K, T-R
33.00 + 0, −1 (full down)	CW	C-A	F-B, H-K, T-R
32.5 to 28.5*	CCW	C-A, F-B	H-K, T-R
21.75 ± .25 Degrees	CCW	H-K, T-R, C-A, F-B	
0 + 1, −0	CCW	H-K, T-R, F-B	C-A

Leading Edge (JR1051-7, -9, -11, -13) (See Note for -11)

0 Degrees (full up)	CW	F-B (C-D)	C-A (B-A)
0.5 Degrees to 4.5*	CW	C-A, F-B, (B-A, C-D)	
45.0 + 0, −1 (full down)	CW	C-A (B-A)	F-B (C-D)
44.5 to 40.5*	CCW	C-A, F-B (B-A, C-D)	
0 Degrees + 1, −0	CCW	F-B (C-D)	C-A (B-A)

Trailing Edge (JR1051-6)

0 Degrees (full up)	CW	F-H, C-H	A-G, C-G
0.5 to 4.5*	CW	A-G, F-H, C-H	C-G
52.25 ± .25 Degrees	CW	A-G, F-H	C-G, C-H
53.5 to 57.5*	CW	A-G, F-H, C-G	C-H
114 + 0, −1 (full down)	CW	A-G, C-G	C-H, F-H
113.5 to 109.5*	CCW	A-G, C-G, F-H	C-H
53.75 ± .25 Degrees	CCW	A-G, F-H	C-H, C-G
52.5 to 48.5*	CCW	A-G, F-H, C-H	C-G
0 + 1, −0 (full up)	CCW	F-H, C-H	C-G, A-G

Trailing Edge (JR1051-8, -10, -12 and -14 (See Note for -10)

0 Degrees (full up)	CW	F-H, C-H (E-D, C-D)	A-G, C-G (A-B, C-B)
0.5 to 4.5*	CW	A-G, F-H, C-H (A-B, E-D, C-D)	C-G (C-B)
44.00 ± .25 Degrees	CW	A-G, F-H (A-B, E-D)	C-G, C-H (C-B, C-D)
45.25 to 49.25*	CW	A-G, F-H, C-G (A-B, E-D, C-B)	C-H (C-D)
90 + 0, −1 (full down)	CW	A-G, C-G (A-B, C-B)	C-H, F-H (C-D, E-D)
89.5 to 85.5*	CCW	A-G, C-G, F-H (A-B, C-B, E-D)	C-H (C-D)

Shaft Position Measured from Full-Up Position	Direction of Shaft Rotation	Continuity Between Pins	Open Between Pins
45.5 ± .25 Degrees	CCW	A-G, F-H (A-B, E-D)	C-H, C-G (C-D, C-B)
44.25 to 40.25*	CCW	A-G, F-H, C-H (A-B, E-D, C-D)	C-G (C-B)
0 + 1, −0 (full up)	CCW	F-H, C-H (E-D, C-D)	C-G, A-G (C-B, A-B)

*The contacts of the switch being actuated shall close within this span.

Note: On the -10 and -11 switch the circuit shall be closed or open between pins enclosed by parenthesis.

27.4.6.9 INSPECTION FOR EXCESSIVE LOOSENESS IN LEADING EDGE FLAPS. If total flap motion, measured at the leading edge with the flaps at any position, exceeds 0.30 inch when the leading edge is moved by hand, further inspection of the actuator and hinge or flap rerigging is required.

Note

Inspect the flap actuators in accordance with Paragraph 27.4.6.6.

Note

The hole in the hinge halves should not exceed 0.140 inch diameter. The hinge pins should not be less than 0.122 inch diameter.

27.4.7 FLAP TORQUE TUBE AND DRIVE INSTALLATION. Key spline engagement dimensions, normally achieved by adding spacers, can be lost when replacing torque tube assemblies or disassembling the drive installation. If, for any reason, minimum dimensions are not maintained, separation of the components of the drive system can result. Since the installation of both the leading and trailing edge flap torque tube drive involves a large number of different assembly points and varying dimensions for end play, a complete description is considered desirable. Figure 27-24A provides this description and all the information necessary to correctly remove, re-install, and dimensionally check the drive installation.

27.4.7.1 INDEXING OF FLAP DRIVE UNIVERSAL JOINTS. There are location dots or witness holes on the rod side of most JetStar flap drive universal joints. The rod side is opposite the spline side of the universal joint. Some flap drive universal joints have witness holes only on the spline side. In all cases the location dots or witness holes are used as indexing points to properly align the universal joint to the tube assembly and to orient the tube assembly on installation.

CAUTION

If the witness holes or location dots on the flap drive universal joints are inadvertently misaligned the result can be excessive tube whipping and vibration during flap operation. Excessive noise and premature failure of the universal joints and flap drive bearings may also occur. This is especially true at locations where the universals are installed at a noticeable angle.

a. When a new universal joint is installed, the location dot or witness hole should be positioned so that it is aligned with (in the same plane as) the corresponding indicator on the universal joint at the opposite end of the tube assembly.

b. When a flap drive torque tube assembly is installed on the airplane the witness hole on the spline side of each universal joint should align with (in the same plane as) the witness hole on the spline side of the universal joint installed on the opposite end of the spline.

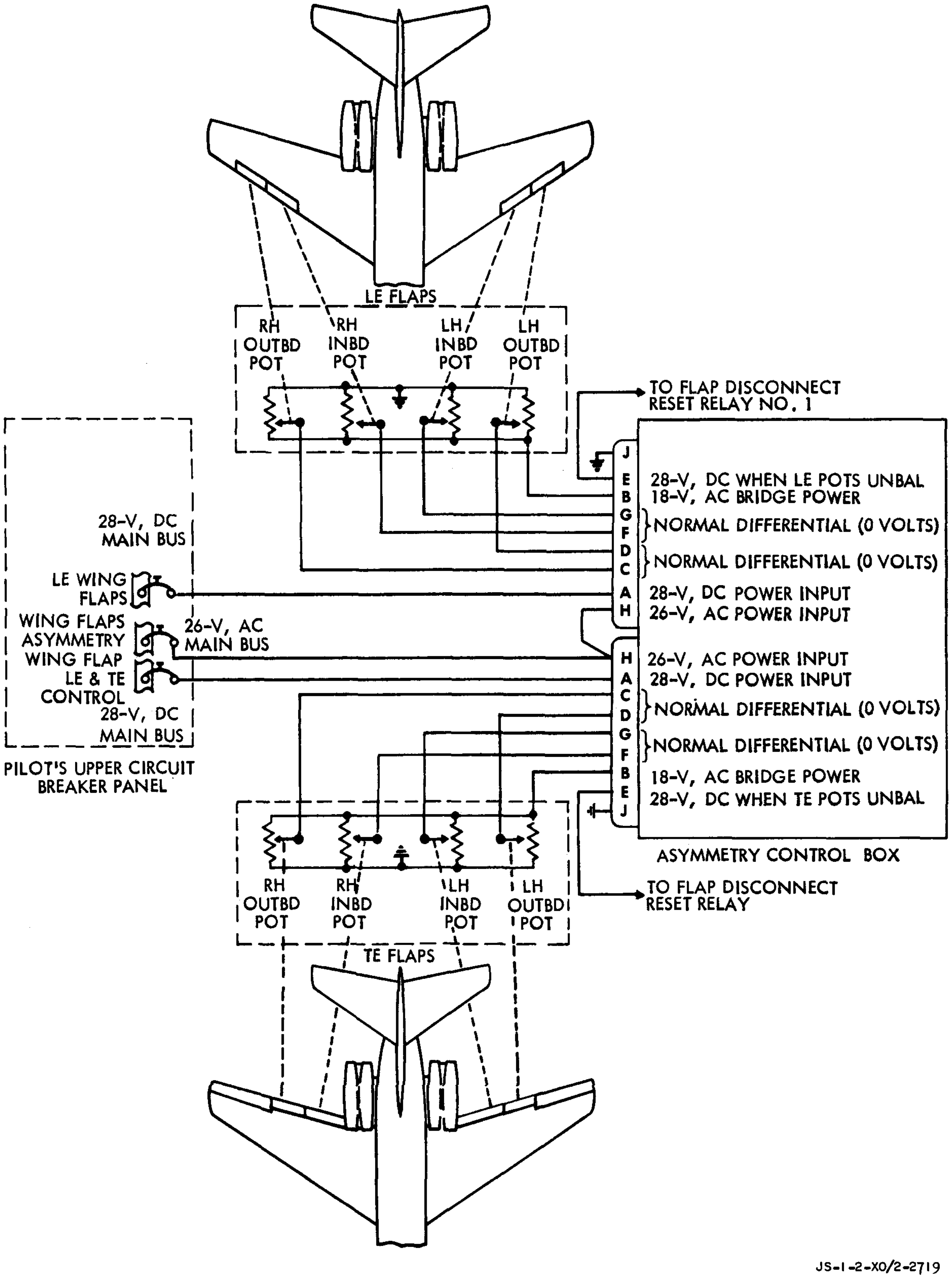
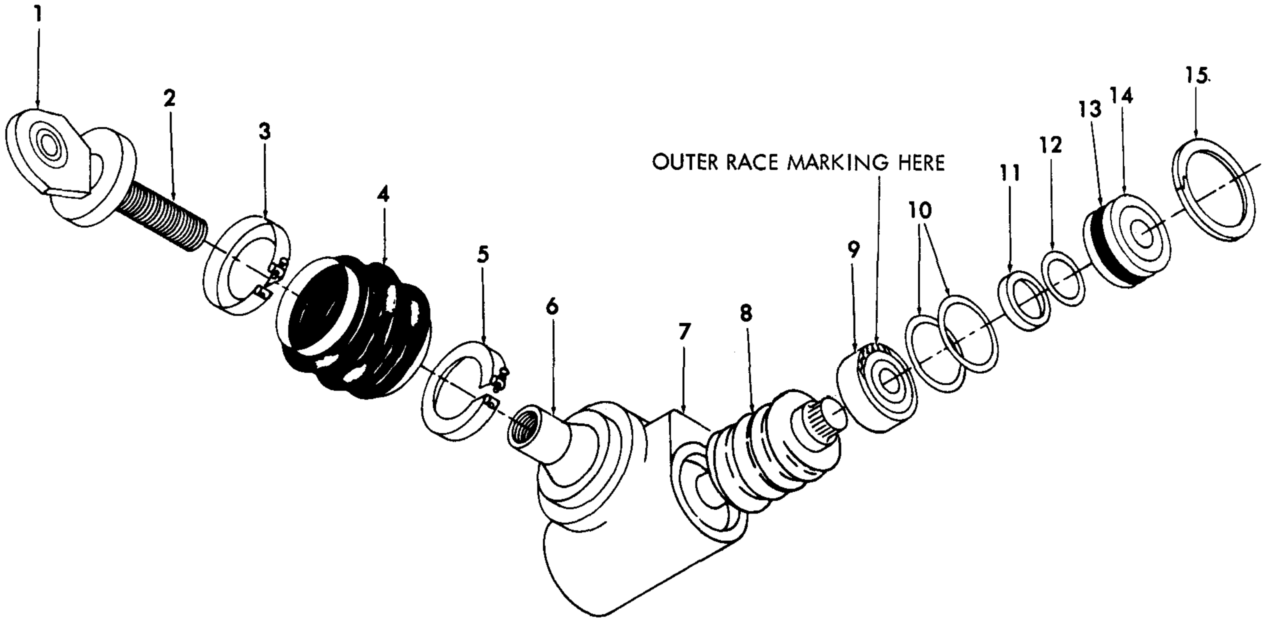
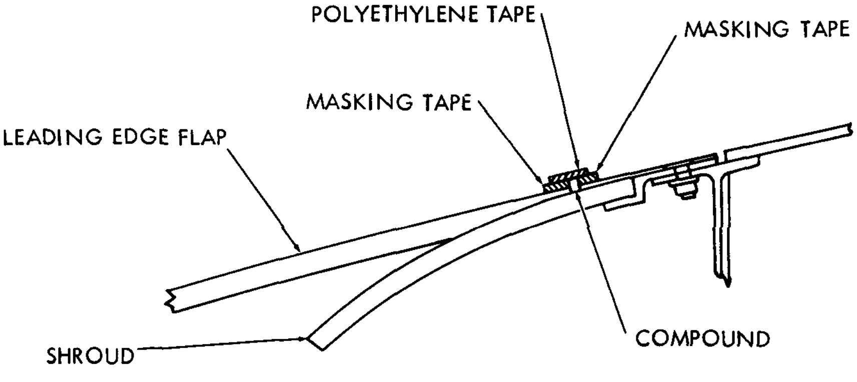


Figure 27-19. Wing Flap Asymmetry System Schematic Diagram



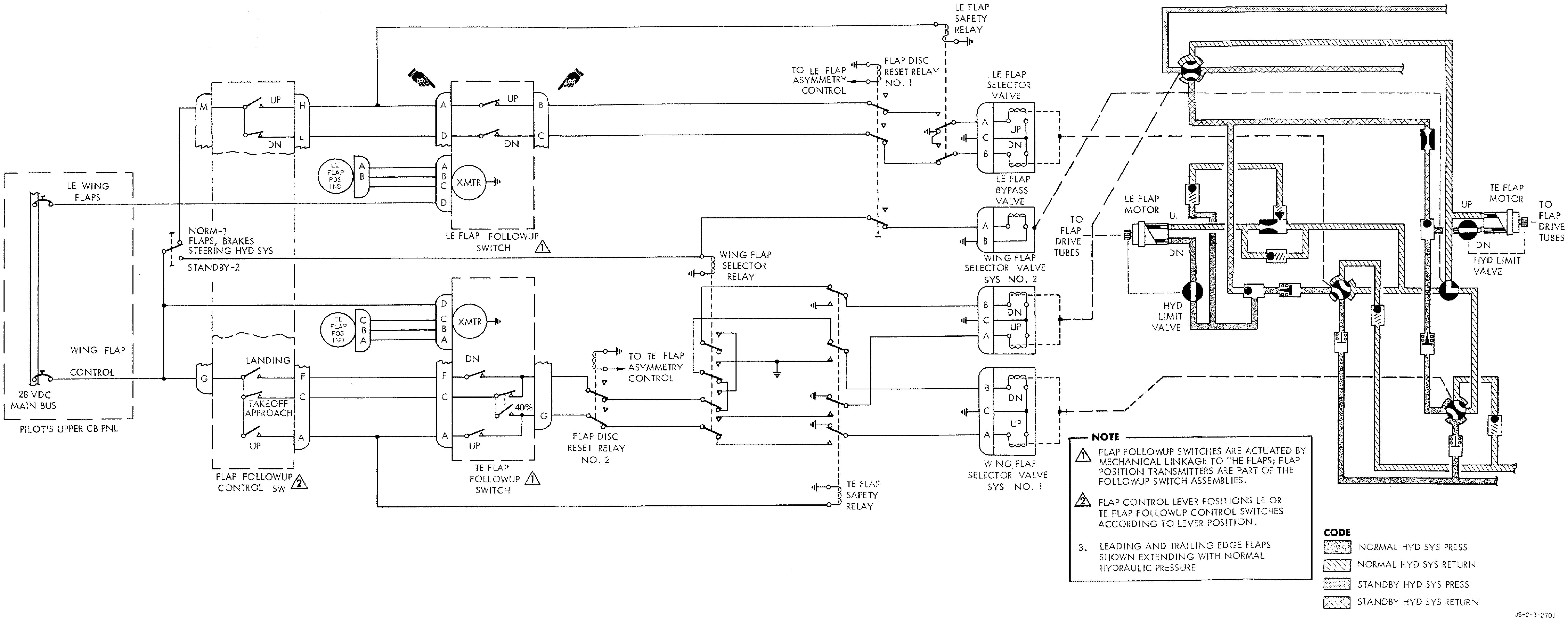
1. ACTUATOR ROD END
2. JACKSCREW
3. FORWARD BOOT RETAINER
4. JACKSCREW BOOT
5. AFT BOOT RETAINER
6. WORM GEAR (JACKSCREW NUT)
7. ACTUATOR BODY
8. WORM SHAFT
9. BEARING
10. SHIMS
11. SPACER (IF PRESENT)
12. SHAFT SEAL
13. STATIC SEAL
14. BEARING RETAINER
15. "SPIROLOX" RETAINER

Figure 27-20. Wing Flap Actuator Assembly



JS-2-2-2711

Figure 27-21. Aerodynamic Smoothing and Sealing of the Leading Edge Flap Shrouds



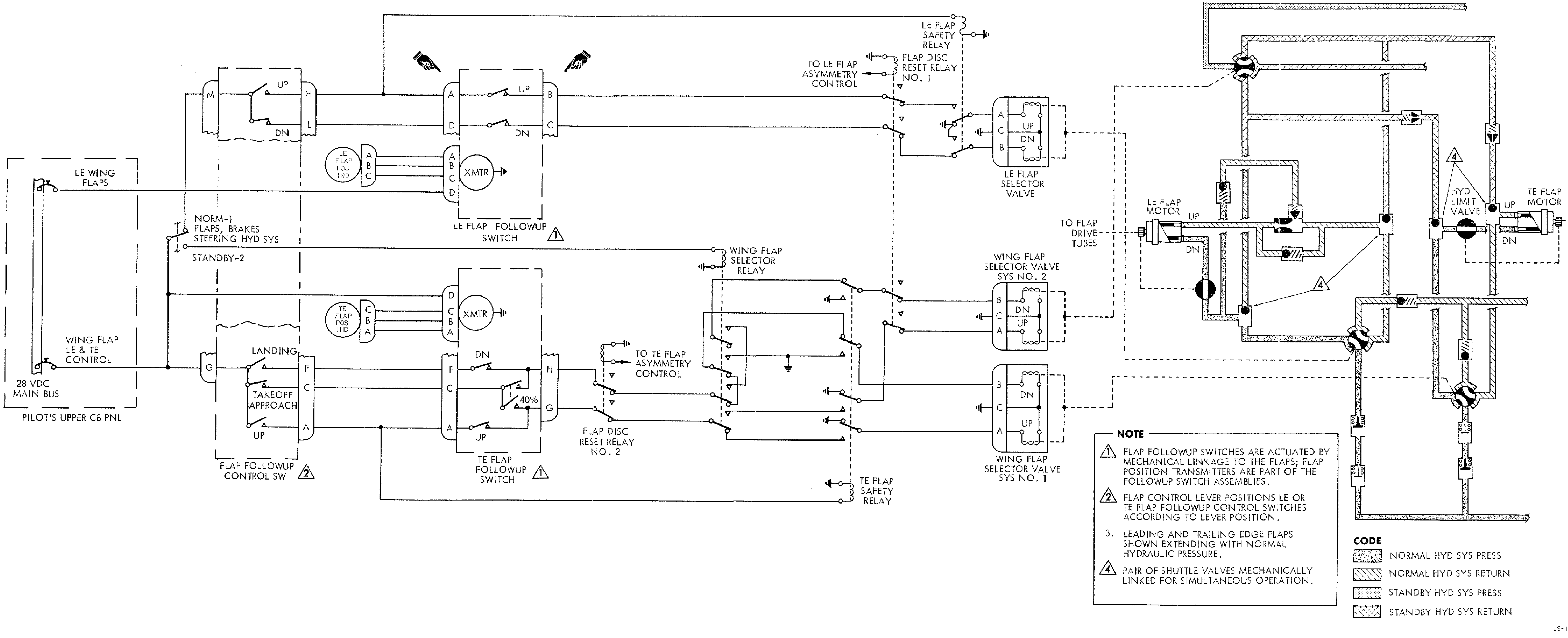
NOTE

1. FLAP FOLLOWUP SWITCHES ARE ACTUATED BY MECHANICAL LINKAGE TO THE FLAPS; FLAP POSITION TRANSMITTERS ARE PART OF THE FOLLOWUP SWITCH ASSEMBLIES.
2. FLAP CONTROL LEVER POSITIONS LE OR TE FLAP FOLLOWUP CONTROL SWITCHES ACCORDING TO LEVER POSITION.
3. LEADING AND TRAILING EDGE FLAPS SHOWN EXTENDING WITH NORMAL HYDRAULIC PRESSURE

CODE

- ▨ NORMAL HYD SYS PRESS
- ▨ NORMAL HYD SYS RETURN
- ▨ STANDBY HYD SYS PRESS
- ▨ STANDBY HYD SYS RETURN

Figure 27-22. Leading and Trailing Edge Wing Flap Systems Schematic Diagram (Airplanes 5001 thru 5045)


LE FLAP SAFETY RELAY
FLAP DISC RESET RELAY NO. 1
TO LE FLAP ASYMMETRY CONTROL
UP
DN
M
H
L
A
D
B
C
LE FLAP POS IND
XMTR
LE FLAP FOLLOWUP SWITCH 1
LE WING FLAPS
NORM-1 FLAPS, BRAKES STEERING HYD SYS
STANDBY-2
WING FLAP SELECTOR RELAY
LE FLAP SELECTOR VALVE
WING FLAP SELECTOR VALVE SYS NO. 2
TO FLAP DRIVE TUBES
LE FLAP MOTOR
HYD LIMIT VALVE
TE FLAP MOTOR
TO FLAP DRIVE TUBES
4
TE FLAP POS IND
WING FLAP LE & TE CONTROL
28 VDC MAIN BUS
PILOT'S UPPER CB PNL
G
LANDING
TAKEOFF APPROACH
F
40%
FLAP FOLLOWUP CONTROL SW 2
TE FLAP FOLLOWUP SWITCH 1
TO TE FLAP ASYMMETRY CONTROL
FLAP DISC RESET RELAY NO. 2
TE FLAP SAFETY RELAY
WING FLAP SELECTOR VALVE SYS NO. 1

NOTE

1. FLAP FOLLOWUP SWITCHES ARE ACTUATED BY MECHANICAL LINKAGE TO THE FLAPS; FLAP POSITION TRANSMITTERS ARE PART OF THE FOLLOWUP SWITCH ASSEMBLIES.
2. FLAP CONTROL LEVER POSITIONS LE OR TE FLAP FOLLOWUP CONTROL SWITCHES ACCORDING TO LEVER POSITION.
3. LEADING AND TRAILING EDGE FLAPS SHOWN EXTENDING WITH NORMAL HYDRAULIC PRESSURE.
4. PAIR OF SHUTTLE VALVES MECHANICALLY LINKED FOR SIMULTANEOUS OPERATION.

CODE

▒	NORMAL HYD SYS PRESS
▨	NORMAL HYD SYS RETURN
░	STANDBY HYD SYS PRESS
▩	STANDBY HYD SYS RETURN

JS-1-2-X0/3-2723

Figure 27-23. Leading and Trailing Edge Wing Flap Systems Schematic Diagram (Airplanes 5046 and Up)

STEP 17

ENERGIZE PINS "B (+)" AND "C (-)" OF THE VALVE WITH 28- VOLT DC POWER. THE FLAPS SHOULD MOVE DOWN PAST THE NORMAL TRAVEL LIMIT, AND THE HYDRAULIC LIMIT VALVE (46) SHOULD INDICATE "OFF" AND STOP FLAP ACTUATION. (SEE DETAIL C.) OBSERVE THAT THE TORQUE TUBES (10) AND (12) ROTATE NOT LESS THAN 5, OR MORE THAN 15 REVOLUTIONS BEFORE GEARBOX OPERATIONS STOPS.

STEP 18

ADJUST THE AFT STOP BOLTS (62) AT THE INBOARD FLAP SUPPORTS ONLY, (6), (8), (13), AND (16) UNTIL THE CLEARANCES BETWEEN THE THREADED ENDS OF THE BOLTS AND THE FLAP HINGE FITTINGS (64) ARE 0.040 TO 0.060 INCH. (SEE DETAIL G.).

STEP 19

RECONNECT THE AIRPLANE ELECTRICAL CONNECTOR TO THE SELECTOR VALVE.

STEP 20

RETURN THE FLAPS TO THE FULLY UP POSITION, AND ADJUST THE FORWARD STOP BOLTS (60) TO OBTAIN 0.040 TO 0.060 INCH CLEARANCE BETWEEN THE THREADED ENDS OF THE BOLTS AND THE FLAP HINGE FITTINGS (64) AT INBOARD AND OUTBOARD FLAP SUPPORTS (2), (4), (6), (8), (13), (16), (18), AND (21). (SEE DETAIL G.).

STEP 21

CLOSE THE "ASYM 26 VAC" CIRCUIT BREAKER, POSITION THE FLAPS IN THE "TAKEOFF/APPROACH" POSITION, AND CONNECT THE ASYMMETRY POTENTIOMETER CABLES (1), (9), (15), AND (19), IF DISCONNECTED. (SEE DETAIL H.).

NOTE

TO PREVENT THE POSSIBILITY OF THE CUTOUT RELAY ENERGIZING, DISCONNECT THE TRAILING EDGE CONNECTOR FROM THE ASYMMETRY CONTROL BOX BEFORE PROCEEDING WITH THE NEXT STEP.

STEP 22

RETURN THE FLAPS TO THE FULLY UP POSITION, AND ADJUST THE ASYMMETRY POTENTIOMETER CABLES (1), (9), (15), AND (19) TO ALIGN THE INDEX MARKS OF THE POTENTIOMETER HOUSINGS (66) WITH THE INDEX MARKS ON THE PULLEYS (67). (SEE DETAIL H.).

STEP 23

MAKE FINAL ADJUSTMENTS TO THE ASYMMETRY POTENTIOMETERS AS FOLLOWS.

A. ATTACH A TEST HARNESS TO THE TRAILING EDGE FLAP CONNECTOR OF THE ASYMMETRY CONTROL BOX. (SEE DETAIL E.) USE A TEST HARNESS SIMILAR TO ONE USED IN THE LEADING EDGE FLAP RIGGING PROCEDURES.

CAUTION

DO NOT CONNECT BOTH CONNECTORS TO THE HARNESS AT THE SAME TIME.

B. CONNECT A TRIPLETT MODEL 630-T (OR EQUIVALENT MULTIMETER TO TERMINALS 3 AND 4 OF THE TEST HARNESS (FOR OUTBOARD POTENTIOMETERS), AND SELECT THE 0.3-VOLT SCALE.

C. OPERATE THE FLAPS FROM ONE EXTREME POSITION TO THE OTHER EXTREME POSITION. OBSERVE THE TOTAL VOLTAGE VARIATION DURING FLAP TRAVEL. THE VOLTAGE VARIATION SHOULD NOT EXCEED +0.15 VOLT.

D. ADJUST THE POTENTIOMETER CABLE ATTACHMENTS UNTIL THE VOLTAGE VARIATION IS MINIMUM AND EQUALLY DIVIDED ABOUT ZERO VOLTS.

NOTE

BY PULLING THE CABLE (1) AND (19), IT CAN BE DETERMINED IN WHICH DIRECTION THE CABLES SHOULD BE ADJUSTED. IN MOST CASES THE VOLTAGE VARIATION CAN BE REDUCED TO 0.05 VOLT. IF A RAPID VOLTAGE VARIATION IS DETECTED AT THE END OF THE UP TRAVEL, IT INDICATES THE POSSIBILITY OF A SLACK CABLE AND/OR A MECHANICAL STOP WHICH IS IMPROPERLY ADJUSTED, AND RESULTS IN STOPPING ONE POTENTIOMETER TRAVEL BEFORE THE OTHER. IF THE VOLTMETER VARIES WITH AN ERRATIC MOVEMENT AT ANY INTERMEDIATE POINT DURING FLAP TRAVEL, THIS IS AN INDICATION OF A MALFUNCTIONING POTENTIOMETER. IN THIS CASE THE POTENTIOMETERS SHOULD BE MONITORED INDIVIDUALLY TO DETERMINE WHICH POTENTIOMETER IS MALFUNCTIONING TO MONITOR THE POTENTIOMETER INDIVIDUALLY, CONNECT A TRIPLETT MODEL 630-T MULTIMETER (SELECT RANGE FOR 28-VOLT DC) BETWEEN THE POTENTIOMETER WIPER AND GROUND. OPERATE THE FLAPS THROUGH THEIR FULL RANGE OF TRAVEL, AND OBSERVE THE METER FOR ERRATIC VOLTAGE VARIATION.

E. ADJUST THE POTENTIOMETER CABLES BY LOOSENING THE JAM NUT (7) AND TURNING THE ADJUSTMENT NUT (70) ON THE CABLE ASSEMBLY. (SEE DETAIL H.)

F. REPEAT (B), (C), (D), AND (E) FOR CABLES (9) AND (15), USING TERMINALS 5 AND 6 OF THE TEST HARNESS.

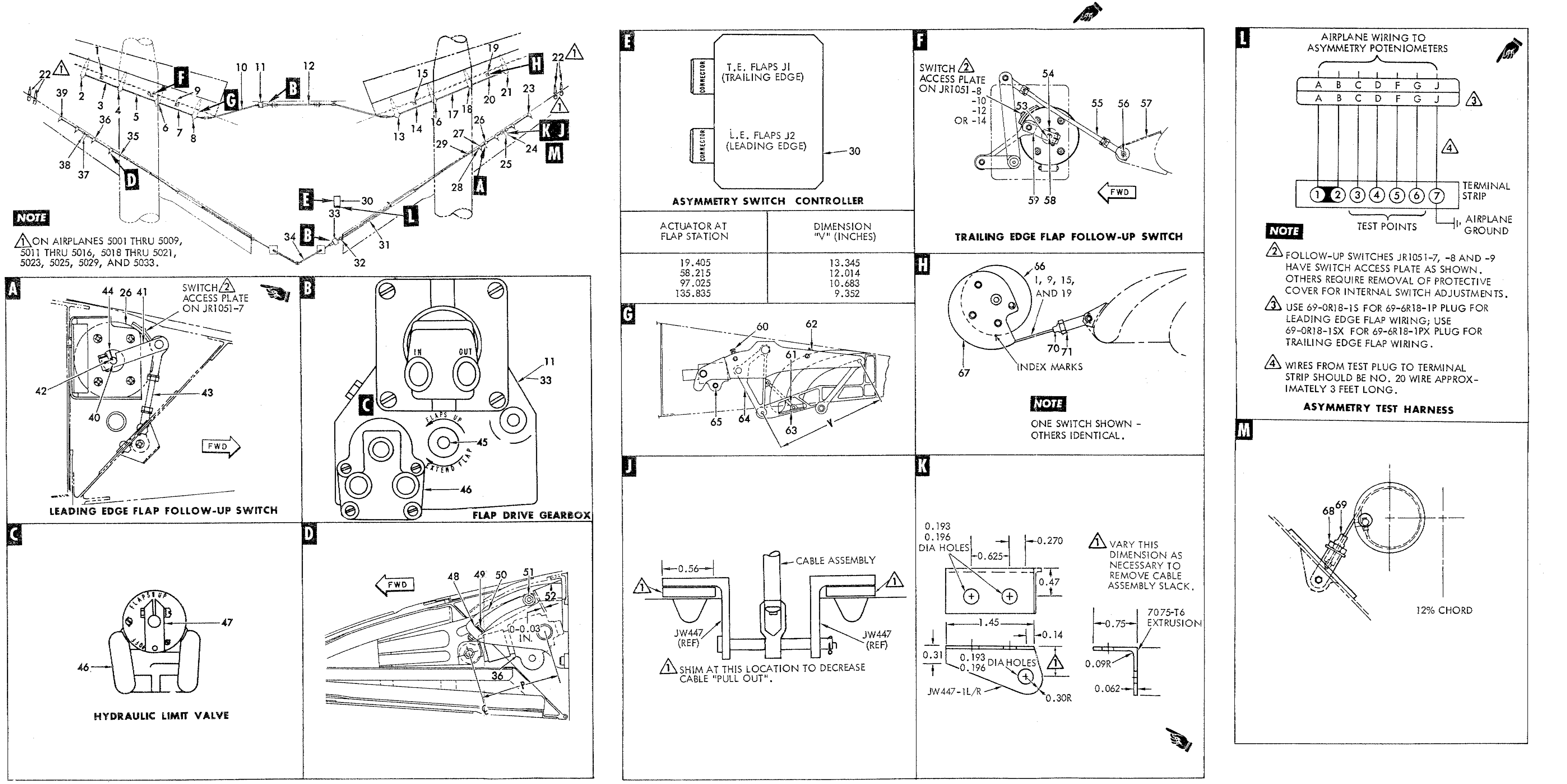
G. DISCONNECT THE TEST HARNESS, AND RECONNECT THE CONNECT TO THE CONTROL BOX.

STEP 24

RECONNECT THE CLOSURE DOOR LINKS (61), AND ADJUST THEM UNTIL THE SEALS AT THE AFT EDGES OF THE DOORS (63) CONTACT THE LOWER SURFACES OF THE FLAPS WHEN THE FLAPS ARE FULLY UP. (SEE DETAIL G.)

STEP 25

OPERATE THE TRAILING EDGE FLAPS FROM FULLY UP TO FULLY DOWN TO CHECK OPERATION. IF THE LEADING EDGE FLAPS WERE DISCONNECTED DURING THE RIGGING PROCEDURE, RECONNECT AND PERFORM AN OPERATIONAL CHECKOUT OF THE FLAPS.



NOTE

1 ON AIRPLANES 5001 THRU 5009, 5011 THRU 5016, 5018 THRU 5021, 5023, 5025, 5029, AND 5033.

A LEADING EDGE FLAP FOLLOW-UP SWITCH

B FLAP DRIVE GEARBOX

C HYDRAULIC LIMIT VALVE

E ASYMMETRY SWITCH CONTROLLER

ACTUATOR AT FLAP STATION	DIMENSION "V" (INCHES)
19.405	13.345
58.215	12.014
97.025	10.683
135.835	9.352

F TRAILING EDGE FLAP FOLLOW-UP SWITCH

H **NOTE**

ONE SWITCH SHOWN - OTHERS IDENTICAL.

J 1 SHIM AT THIS LOCATION TO DECREASE CABLE "PULL OUT".

K 1 VARY THIS DIMENSION AS NECESSARY TO REMOVE CABLE ASSEMBLY SLACK.

L **NOTE**

2 FOLLOW-UP SWITCHES JR1051-7, -8 AND -9 HAVE SWITCH ACCESS PLATE AS SHOWN. OTHERS REQUIRE REMOVAL OF PROTECTIVE COVER FOR INTERNAL SWITCH ADJUSTMENTS.

3 USE 69-0R18-1S FOR 69-6R18-1P PLUG FOR LEADING EDGE FLAP WIRING; USE 69-0R18-1SX FOR 69-6R18-1PX PLUG FOR TRAILING EDGE FLAP WIRING.

4 WIRES FROM TEST PLUG TO TERMINAL STRIP SHOULD BE NO. 20 WIRE APPROXIMATELY 3 FEET LONG.

ASYMMETRY TEST HARNESS

Figure 27-24. Wing Flap Rigging Instructions (Sheet 5 of 5)

WING FLAP RIGGING INSTRUCTIONS - LEADING EDGE

CAUTION

(1) IF THE TRAILING EDGE FLAPS ARE NOT RIGGED THE SYSTEM SHOULD BE DEACTIVATED BY DISCONNECTING ELECTRICAL CONNECTOR AT THE T.E. SELECTOR VALVE.

(2) USE EXTREME CAUTION WHEN OPERATING LEADING EDGE FLAPS PRIOR TO COMPLETION OF RIGGING TO PREVENT ALIGNMENT PIN ON OUTBOARD FLAP FROM INTERFERING WITH THE SLOTTED FITTING ON THE INBOARD FLAP.

(3) IF AIR MOTORS OR OTHER AUXILIARY POWER SOURCES ARE USED TO DRIVE THE FLAP MECHANISMS, THE OUTPUT TORQUE MUST BE LIMITED TO 20 INCH-POUNDS WHEN DIRECTLY CONNECTED TO TORQUE TUBES OR ACTUATORS, AND TO 3 INCH-POUNDS WHEN DIRECTLY CONNECTED TO MANUAL INPUT SHAFTS OF GEARBOXES.

(4) DO NOT ATTEMPT TO ADJUST THE ASYMMETRY CONTROL BOX.

(5) DO NOT APPLY ELECTRICAL POWER TO ASYMMETRY POTENTIOMETERS WHICH ARE AT OPPOSITE EXTREMES OF ADJUSTMENT.

(6) DO NOT ALLOW ASYMMETRY POTENTIOMETER CABLES TO SNAP BACK TO THE SPRING-LOADED POSITION.

STEP 1

WITH ALL ELECTRIC POWER OFF, MOVE THE FLAP CONTROL LEVER TO THE LEADING EDGE "UP" POSITION.

STEP 2

DISCONNECT THE TORQUE SHAFTS (31) AND (34) FROM THE OUTPUT SHAFTS OF THE LEADING EDGE FLAP DRIVE GEARBOX (33). (SEE SHEET 5)

NOTE

THE UNIVERSAL JOINT (28) MAY BE DISCONNECTED INSTEAD OF REMOVING THE COUPLING (32).

STEP 3

DISCONNECT THE SWITCH LINK (41) FROM THE INPUT SHAFT (40) OF THE FOLLOWUP SWITCH ASSEMBLY (26). (SEE DETAIL A).

STEP 4

OPEN THE ASYMMETRY SYSTEM "ASYM" "26 VAC" CIRCUIT BREAKER, OR DISCONNECT THE ASYMMETRY POTENTIOMETER CABLES (24), (25), (37), AND (38) FROM THE LEADING EDGE FLAPS. (SEE SHEET 5)

STEP 5

WITH EXTERNAL HYDRAULIC POWER CONNECTED TO THE NORMAL HYDRAULIC SYSTEM GROUND TEST CONNECTIONS AND WITH EXTERNAL DC POWER CONNECTED TO THE AIRPLANE, PLACE THE FLAP CONTROL LEVER IN THE LEADING EDGE "DN" POSITION, AND OPERATE THE GEARBOX (33) UNTIL OPERATION STOPS AND THE HYDRAULIC LIMIT VALVE INDICATOR (47) INDICATES "OFF". (SEE DETAILS B AND C.)

STEP 6

DISCONNECT THE DC POWER FROM THE AIRPLANE AND RETURN THE FLAP CONTROL LEVER TO THE LEADING EDGE "UP" POSITION.

CAUTION

ELECTRICAL POWER MUST BE DISCONNECTED BEFORE THE CONTROL LEVER IS RETURNED TO THE "UP" POSITION, OR UNCONTROLLED GEARBOX OPERATION WILL RESULT.

STEP 7

ROTATE THE OUTPUT SHAFT (45) OF THE GEARBOX (33) 185 TURNS COUNTERCLOCKWISE ("FLAPS UP" DIRECTION, SEE DETAIL B). MAINTAIN THIS POSITION UNTIL STEPS (8) THROUGH (10) ARE COMPLETED.

STEP 8

CHECK THAT THE "P" DIMENSIONS OF THE TWO INBOARD ACTUATORS (27) AND (36) ARE 4.80 INCHES EACH AND THAT THE "P" DIMENSIONS OF THE OUTBOARD ACTUATORS (23) AND (39) ARE 4.36 INCHES. (SEE DETAIL D.) IF NECESSARY, ADJUST THE ACTUATOR LENGTHS TO OBTAIN THESE DIMENSIONS.

STEP 9

MANUALLY ROTATE THE TORQUE SHAFTS (29) AND (35) UNTIL THE FLAP FITTINGS (50) AND THE WING FRONT BEAM FITTINGS (52) BARELY CONTACT. (SEE DETAIL D.)

STEP 10

CHECK THAT THE FLAP AND WING FRONT BEAM FITTINGS BARELY CONTACT EVENLY ACROSS THE ENTIRE SURFACE (SEVEN PLACES IN EACH LEFT AND RIGHT-HAND WING).

A. IF THE FLAP AND WING FRONT BEAM FITTINGS DO NOT CONTACT, DISCONNECT THE TORQUE TUBES FROM THE ACTUATOR NEAREST THE POINT WHERE THE GAP EXISTS, AND ROTATE THE INPUT SHAFT AS REQUIRED. ONE FULL REVOLUTION WILL CHANGE THE GAP 0.011 INCH. A MAXIMUM OF FOUR TURNS IN EITHER DIRECTION IS PERMISSIBLE.

B. AFTER THE GAPS CLOSE, RECONNECT ALL TORQUE SHAFTS TO THE ACTUATORS WHILE MAINTAINING THE FLAP AND WING FRONT BEAM FITTINGS IN CONTACT. RECONNECT THE TORQUE SHAFTS (31) AND (34) TO THE OUTPUT SHAFTS OF THE GEARBOX (33). CHECK THAT THE EXTERNAL STOPS (22) ARE PROPERLY INSTALLED AND SERVICEABLE ON AIRPLANES 5001 THROUGH 5009, 5011 THROUGH 5016, 5018 THROUGH 5021, 5023, 5025, 5029, AND 5033.

C. MANUALLY ROTATE THE TORQUE TUBES UNTIL ALL FLAP FITTINGS CLEAR THE FRONT BEAM FITTINGS. CHECK THE GAPS AT ALL 14 PLACES IN THE LEFT AND RIGHT WINGS WITH A FEELER GAGE. THE DIFFERENCE IN GAP FROM ANY ONE LOCATION TO ANY ADJACENT LOCATION SHALL NOT EXCEED 0.010 INCH. MAKE NECESSARY ADJUSTMENTS, IF REQUIRED, AS INSTRUCTED IN PARAGRAPH B. ABOVE.

Figure 27-24. Wing Flap Rigging Instructions (Sheet 1 of 5)

STEP 11

ADJUST THE FOLLOWUP SWITCH (SEE DETAIL A) AS FOLLOWS:
A. MANUALLY ROTATE THE INPUT SHAFT (40) UNTIL THE INDEX MARK (42) IS DISPLACED 10 DEGREES CLOCKWISE FROM THE INDEX MARK (44).
B. CONNECT A TRIPLETT MODEL 630.T (OR EQUIVALENT) MULTIMETER TO THE "A" AND "B" PINS OF THE ELECTRICAL CONNECTOR, AND SET THE MULTIMETER TO INDICATE CONTINUITY.
C. MANUALLY ROTATE THE INPUT SHAFT (40) COUNTERCLOCKWISE UNTIL THE MULTIMETER INDICATES AN OPEN CIRCUIT.
D. WITH THE FLAPS FULLY UP AND WITH ALL FLAP FITTINGS (50) (SEE DETAIL D) BARELY CONTACTING THE FRONT BEAM FITTINGS (52), CONNECT THE LINK (41) TO THE FOLLOWUP SHAFT (40) OF THE SWITCH (26). DO NOT DISTURB THE POSITION OF THE SHAFT (40) WHILE MAKING THIS CONNECTION. IF THE LINK WILL NOT LINE UP WITH THE SHAFT, ADJUST THE LENGTH OF THE LINK (43). (SEE DETAIL A)
E. FOR FINAL PRECISE ADJUSTMENTS (TO MAINTAIN CORRECT GAPS ON MECHANICAL STOPS) IT MIGHT BE NECESSARY TO SLIGHTLY VARY THE VERTICAL POSITION OF THE LIMIT SWITCH. RAISING THE SWITCH CLOSES THE GAP AND LOWERING THE SWITCH OPENS THE GAP. ON AIRCRAFT WITH JR1051-7 SWITCHES, AN ACCESS PLATE IF PROVIDED FOR FINE ADJUSTMENT OF THE LIMIT SWITCH (SEE DETAIL A).

CAUTION

BEFORE PROCEEDING WITH FOLLOWING STEPS, ASSURE THAT ALL ACTUATOR ROD ENDS AND ALL SECTIONS OF TORQUE TUBES ARE PROPERLY CONNECTED. SINCE THE ASYMMETRY SYSTEM IS INOPERATIVE DURING THE FOLLOWING STEPS, AN IMPROPERLY CONNECTED ROD END OR TORQUE TUBE CAN RESULT IN SEVERE DAMAGE TO FLAP SECTIONS AND ADJACENT STRUCTURE. MONITOR FLAP SECTIONS CLOSELY DURING OPERATION TO ASCERTAIN THAT NO "OUT OF SYMMETRY" CONDITION DEVELOPS.

STEP 12

APPLY EXTERNAL HYDRAULIC AND ELECTRIC POWER TO THE AIRPLANE AND MOVE THE FLAP CONTROL LEVER FROM THE LEADING EDGE "UP" POSITION TO THE LEADING EDGE "DN" POSITION. THE INBOARD LEADING EDGE FLAPS SHOULD MOVE DOWN 22 (±2) DEGREES, AND THE OUTBOARD FLAPS SHOULD MOVE DOWN 27 (±2) DEGREES IN THE FULLY DOWN POSITION. THE POSITION INDICATOR SHOULD SHOWN "DN" (±5) PERCENT. ON JR1051-7 POSITIONING SWITCHES, THE LIMITS OF FLAP TRAVEL MAY BE ADJUSTED BY OPENING THE ACCESS PLATE, (SEE DETAIL A). THE POSITION INDICATOR MAY BE ADJUSTED BY ROTATING THE POSITION TRANSMITTER IN ITS MOUNT.

STEP 13

RETURN THE FLAPS TO THE "UP" POSITION AND CHECK THE PROXIMITY OF THE FLAP FITTINGS TO THE WING FRONT BEAM FITTINGS. REPEAT FIVE TIMES. THE GAP BETWEEN THE FLAP FITTINGS AND WING FRONT BEAM FITTINGS, IF ONE EXISTS, SHOULD NOT EXCEED 0.030 INCH EACH TIME. THERE SHOULD BE NO EVIDENCE OF HARD CONTACT AT THE FITTING SURFACES AS THE FLAP REACHES THE "UP" POSITION. IF HARD CONTACT IS EVIDENT, READJUST THE LIMIT SWITCH AS INSTRUCTED IN STEP (11).

CAUTION

IF THE FLAPS ARE RIGGED WHILE THE AIRPLANE IS ON JACKS, THE GAPS MUST BE RECHECKED AND CORRECTED AS NECESSARY PER STEPS (9), (10), AND (11) AFTER THE AIRPLANE IS LOWERED FROM THE JACKS.

STEP 14

WITH THE LEADING EDGE FLAPS UP, DISCONNECT THE *ELECTRICAL CONNECTOR FROM THE NO. 1 WING FLAP* SELECTOR VALVE (LOCATED AT FUSELAGE STATION 420), AND ENERGIZE PINS "B(+)" AND "C(-)" WITH 28-VOLTS DC POWER. THE FLAPS SHOULD MOVE DOWN PAST THE NORMAL LIMIT OF TRAVEL AND ACTUATE THE HYDRAULIC LIMIT VALVE (46). AS THE LIMIT VALVE INDICATOR (47) REACHES THE "OFF" POSITION (SEE DETAIL C), GEARBOX OPERATION SHOULD STOP. IN THIS POSITION, THE INBOARD FLAPS SHOULD BE 23 (±2) DEGREES AND THE OUTBOARD FLAPS 28 (±2) DEGREES FROM THE UP POSITION. THE POSITION INDICATOR SHOULD SHOW APPROXIMATELY 5 PERCENT PAST "DN".

STEP 15

WITH THE FLAPS STOPPED IN THE FULLY DOWN POSITION OF STEP (14), DIMENSIONS "P" OF DETAIL D SHOULD NOT EXCEED 7/11 INCHES, AND THE UPPER SURFACE OF THE FLAPS SHOULD OVERLAP THE SHROUDS AT LEAST 0.35 INCH.

STEP 16

WITH THE FLAPS STILL IN THE FULLY DOWN POSITION OF STEP (14), CHECK THAT THERE IS A GAP OF 0.04 TO 0.06 INCH BETWEEN THE STOP BOLTS (51), AND THE STOP BLOCKS (49). ADJUST AS REQUIRED BY ADDING OR REMOVING SHIMS (48). (SEE DETAIL D.)

STEP 17

REMOVE THE TEST HARNESS FROM THE FLAP SELECTOR VALVE, AND RECONNECT THE AIRPLANE ELECTRICAL CONNECTOR TO THE VALVE.

STEP 18

CLOSE "ASYM 26 VAC" CIRCUIT BREAKER OR PLACE THE LEADING EDGE FLAPS IN THE FULLY UP POSITION, AND CONNECT THE FOUR ASYMMETRY POTENTIOMETER CABLES (24), (25), (37), AND (38), IF DISCONNECTED.

STEP 19

ADJUST CABLES (24), (25), (37), AND (38) FOR NO SLACK WITH THE FLAPS FULLY UP.

STEP 20

ADJUST THE ASYMMETRY POTENTIOMETERS AS FOLLOWS:
A. ATTACH A TEST HARNESS TO THE LEADING EDGE FLAP CONNECTOR OF THE ASYMMETRY CONTROL BOX (SEE DETAIL E). USE A TEST HARNESS SIMILAR TO THE ONE SHOWN IN THE SCHEMATIC DIAGRAM OF DETAIL L.

CAUTION

DO NOT CONNECT BOTH CONNECTORS TO *THE TEST HARNESS AT THE SAME TIME.*

JS-1-2-X0/6-2724-2

Figure 27-24. Wing Flap Rigging Instructions (Sheet 2 of 5)

B. CONNECT A TRIPLETT MODEL 630-T (OR EQUIVALENT) MULTIMETER TO TERMINALS 3 AND 4 OF THE TEST HARNESS (FOR OUTBOARD POTENTIOMETERS), AND SELECT THE 0.3-VOLT SCALE.

C. OPERATE THE FLAPS FROM ONE EXTREME POSITION TO THE OTHER EXTREME POSITION. OBSERVE THE TOTAL VOLTAGE VARIATION DURING THE FLAP TRAVEL. THE VOLTAGE VARIATION SHOULD NOT EXCEED ±0.15 VOLT.

D. ADJUST THE POTENTIOMETER CABLE ATTACHMENTS UNTIL THE VOLTAGE VARIATION IS MINIMUM AND EQUALLY DIVIDED ABOUT ZERO VOLTS.

NOTE

USE THE SAME TEST HARNESS AS SPECIFIED IN THE PRECEDING PARAGRAPH A (UNDER STEP 20) AND NOTE THAT BRIDGE POWER TO PIN B IS NOW 28 VDC BECAUSE OF THE CONNECTION BETWEEN PIN A AND B AT THE TERMINAL STRIP.

METER CHECKS WILL BE ACCOMPLISHED ON D.C. SCALES.

ON AIRPLANES LAC SERIAL 5001 THROUGH 5029, AND 5031 THROUGH 5045, THE LEADING EDGE FLAP ASYMMETRY POTENTIOMETER CABLES ARE ATTACHED WITH A STANDARD CABLE AND FITTING (FORK) TO FIXED ANGLE BRACKETS ON THE FLAP AFT BEAM. (SEE DETAILS J AND K.)

ON AIRPLANES LAC SERIAL 5046 AND UP, AND ON LAC SERIALS 5001 THROUGH 5029, AND 5031 THROUGH 5045 ON WHICH SERVICE BULLETIN 329-140 HAS BEEN INCORPORATED, A TURNBUCKLE-TYPE CABLE END FITTING IS PROVIDED FOR EASE OF ADJUSTMENT. (SEE DETAIL M.)

BY PULLING THE CABLES (24 AND 38), IT CAN BE DETERMINED WHICH CABLE SHOULD BE ADJUSTED. FOR AIRCRAFT WITHOUT ADJUSTABLE CABLE END FITTINGS, IF PULLING ONE CABLE DECREASES THE VOLTAGE, THE OPPOSITE CABLE ATTACH BRACKET (SEE DETAIL J) SHOULD BE SHIMMED. IN MOST CASES THE VOLTAGE VARIATION CAN BE REDUCED TO 0.05 VOLT. IF A RAPID VOLTAGE VARIATION IS DETECTED AT THE END OF THE "UP" TRAVEL, IT INDICATES THE POSSIBILITY OF A SLACK CABLE AND/OR A MECHANICAL STOP WHICH IS IMPROPERLY ADJUSTED AND RESULTS IN STOPPING ONE POTENTIOMETER TRAVEL BEFORE THE OTHER. IF THE VOLTMETER VARIES WITH AN ERRATIC MOVEMENT AT ANY INTERMEDIATE POINT DURING FLAP TRAVEL, THIS IS AN INDICATION OF A MALFUNCTIONING POTENTIOMETER. IN THIS CASE, THE POTENTIOMETERS SHOULD BE MONITORED INDIVIDUALLY TO DETERMINE WHICH POTENTIOMETER IS MALFUNCTIONING. TO MONITOR THE POTENTIOMETER INDIVIDUALLY, CONNECT A TRIPLETT 630-T MULTIMETER (SELECT RANGE FOR 28-VOLT DC) BETWEEN THE POTENTIOMETER WIPER AND GROUND. OPERATE THE FLAPS THROUGH THEIR FULL RANGE OF TRAVEL, AND OBSERVE THE METER FOR ERRATIC VOLTAGE VARIATION.

E. TO ADJUST THE POTENTIOMETER CABLE ATTACHMENT FOR AIRCRAFT WITHOUT THE ADJUSTABLE CABLE END FITTINGS, THE CABLES ARE ADJUSTED AS FOLLOWS:

(1) TO ELIMINATE SLACK CABLES IN THE FLAP UP POSITION, IT WILL BE NECESSARY TO FABRICATE NEW CABLE ATTACH CLIPS WHICH HAVE THE CABLE ATTACH HOLE LOCATED FORWARD OF THE PRESENT HOLE. (SEE DETAIL K.)

(2) TO BALANCE THE LEFT POTENTIOMETER WITH THE RIGHT, IT IS NECESSARY TO DECREASE CABLE "PULL-OUT" FOR ONE POTENTIOMETER BY ADDING SHIMS BETWEEN THE CABLE ATTACH CLIP AND AFT FACE OF BEAM WEB. SHIMS SHOULD BE FABRICATED FROM ALUMINUM LAMINATED SHIM STOCK WITH THE SAME THICKNESS USED UNDER BOTH CLIPS AT THE END OF THE CABLE WHERE ADJUSTMENT IS REQUIRED. (SEE DETAIL J.)

F. TO ADJUST THE POTENTIOMETER CABLES FOR AIRCRAFT WITH THE ADJUSTABLE CABLE END FITTINGS THE CABLES ARE ADJUSTED BY LOOSENING THE JAM NUT (68) AND TURNING THE ADJUSTMENT NUT (69) ON THE CABLE ASSEMBLY. (SEE DETAIL M.)

G. REPEAT (B), (C), (D), (E), AND (F) FOR CABLES (25) AND (37), USING TERMINALS 5 AND 6 OF THE TEST HARNESS.

H. DISCONNECT THE TEST HARNESS AND RECONNECT THE CONNECTOR TO THE CONTROL BOX.

RETURN THE FLAP SYSTEM TO NORMAL, AND PERFORM AN OPERATIONAL CHECKOUT OF THE FLAP SYSTEMS. IF THE TRAILING EDGE FLAPS NEED TO BE RIGGED, WAIT AND CHECK BOTH SYSTEMS AT THE SAME TIME.

WING FLAP RIGGING INSTRUCTIONS - TRAILING EDGE

NOTE

DISCONNECT THE ELECTRICAL CONNECTOR FROM THE LEADING EDGE FLAP SELECTOR VALVE FOR SYSTEM NO. 1 IF THE LEADING EDGE FLAPS ARE NOT TO BE ACTUATED DURING THIS PROCEDURE.

CAUTION

MAKE SURE THAT THE AFT STOP BOLTS (62) AND THE FORWARD STOP BOLTS (60) DO NOT INTERFERE WITH TRAILING EDGE FLAP ACTUATION. (SEE DETAIL G.)

STEP 1

DISCONNECT THE FLAP DRIVE TORQUE TUBES (10) AND (12) FROM THE OUTPUT SHAFTS OF THE TRAILING EDGE FLAP DRIVE GEARBOX (11). (SEE SHEET 5)

STEP 2

OPEN THE "ASMY 26 VAC" CIRCUIT BREAKER OR DISCONNECT THE ASYMMETRY POTENTIOMETER CABLES (1), (9), (15), AND (19) FROM THE TRAILING EDGE FLAPS. (SEE DETAIL H.)

STEP 3

DISCONNECT THE FOLLOWUP SWITCH LINK (59) FROM THE SWITCH SHAFT (53). (SEE DETAIL F.)

STEP 4

DISCONNECT THE CLOSURE DOOR LINKS (61) FROM THE CLOSURE DOORS (63). (SEE DETAIL G.)

STEP 5

CONNECT EXTERNAL DC POWER TO THE AIRPLANE, AND CONNECT HYDRAULIC POWER TO THE NORMAL HYDRAULIC SYSTEM.

STEP 6

PLACE THE FLAP AND BRAKE HYD SYS SWITCH ON THE MAIN INSTRUMENT PANEL IN THE "NORM-1" POSITION.

Figure 27-24. Wing Flap Rigging Instructions (Sheet 3 of 5)

STEP 7

MOVE THE FLAP CONTROL LEVEL TO THE "LANDING" POSITION, AND OPERATE THE FLAP DRIVE GEARBOX (11) UNTIL OPERATION STOPS AND THE HYDRAULIC LIMIT VALVE INDICATOR (47) INDICATES "OFF". (SEE DETAIL C)

STEP 8

DISCONNECT THE EXTERNAL ELECTRICAL POWER SOURCE; THEN RETURN THE FLAP CONTROL LEVER TO THE TRAILING EDGE "UP" POSITION.

CAUTION

ELECTRICAL POWER MUST BE DISCONNECTED BEFORE THE CONTROL LEVER IS RETURNED TO THE "UP" POSITION, OR UNCONTROLLED GEARBOX OPERATION WILL RESULT.

STEP 9

ROTATE THE OUTPUT SHAFT (45) OF THE FLAP DRIVE GEARBOX (11) 360 REVOLUTIONS COUNTERCLOCKWISE TO OBTAIN THE FULLY UP POSITION. (SEE DETAIL B) MAINTAIN THIS POSITION UNTIL AFTER COMPLETING STEPS (10)THROUGH (13).

STEP 10

IF THE "V" DIMENSIONS OF THE ACTUATORS DO NOT CORRESPOND TO THE FOLLOWING DIMENSIONS, DISCONNECT THE TORQUE SHAFTS (3), (5), (7), (14), (17) AND (20) FROM THE ACTUATORS (2), (4), (6), (8), (13), (16), (18) AND (21) AS REQUIRED, AND ADJUST AS NECESSARY. (SEE DETAIL G.) RECONNECT THE TORQUE SHAFTS UPON COMPLETION OF THIS STEP.

STEP 11

FROM THE FLAP-UP POSITION AS DESCRIBED IN STEP (10), MANUALLY ROTATE THE TORQUE SHAFTS (10) AND (12) AS REQUIRED TO FAIR THE FLAP TO THE WING. INBOARD AND OUTBOARD FLAPS MUST BE FAIRED TO WITHIN 0.06 INCH OF THE TRAILING EDGE OF THE WING.

NOTE

FINAL ADJUSTMENT MAY BE MADE BY DISCONNECTING THE TORQUE SHAFTS, PER STEP (10), FROM THE ACTUATORS AND THEM MOVING THE FLAPS AS REQUIRED BY ROTATING THE JACKSCREW INPUT SHAFTS (65) (SEE DETAIL G). ONE SHAFT ROTATION CHANGES JACKSCREW LENGTH APPROXIMATELY 0.011 INCHES. DO NOT EXCEED 5 TURNS IN EITHER DIRECTION.

STEP 12

RECONNECT THE TORQUE SHAFTS DISCONNECTED IN STEP (11), AND WHILE MAINTAINING THE FAIRED POSITION, CONNECT THE TORQUE SHAFTS (10) AND (12) TO THE OUTPUT SHAFTS OF THE GEARBOX (11).

STEP 13

ADJUST THE FOLLOWUP SWITCH AS FOLLOWS:
(SEE DETAIL F).

A. MANUALLY ROTATE THE SHAFT (53) UNTIL THE INDEX MARK (58) IS DISPLACED APPROXIMATELY 10 DEGREES CLOCKWISE FROM THE INDEX MARK (54).

B. CONNECT A TRIPLETT MODEL 630-T (OR EQUIVALENT) MULTIMETER TO THE CONNECTOR PINS "A" AND "G" AND SET THE MULTIMETER TO INDICATE CONTINUITY.

C. MANUALLY ROTATE THE SHAFT (53) COUNTERCLOCKWISE UNTIL THE METER INDICATES AN OPEN CIRCUIT.

D. WITH THE FLAPS UP AS IN STEP (10), CONNECT THE LINK (59) TO THE SWITCH SHAFT (53). DO NOT DISTURB THE POSITION OF THE INPUT SHAFT WHILE MAKING THIS CONNECTION IF THE LINK FAILS TO ALIGN WITH THE SHAFT, ADJUST THE LENGTH OF THE LINK (55).

STEP 14

CONNECT THE EXTERNAL HYDRAULIC POWER TO THE NORMAL HYDRAULIC SYSTEMS, AND CONNECT DC POWER TO THE AIRPLANE.

CAUTION

BEFORE PROCEEDING WITH FOLLOWING STEPS, ASSURE THAT ALL ACTUATOR ROD ENDS AND ALL SECTIONS OF TORQUE TUBES ARE PROPERLY CONNECTED. SINCE THE ASYMMETRY SYSTEM IS INOPERATIVE DURING THE FOLLOWING STEPS, AN IMPROPERLY CONNECTED ROD END OR TORQUE TUBE CAN RESULT IN SEVERE DAMAGE TO FLAP SECTIONS AND ADJACENT STRUCTURE. MONITOR FLAP SECTIONS CLOSELY DURING OPERATION TO ASCERTAIN THAT NO "OUT OF SYMMETRY" CONDITION DEVELOPS.

STEP 15

FROM THE TRAILING EDGE "UP" POSITION MOVE CONTROL LEVER TO THE "LANDING" POSITION. IN THE "LANDING" POSITION, THE POSITION INDICATOR SHOULD SHOW "DN" (±5 PERCENT) AND THE FLAP SURFACE SHOULD BE 50 (+0, -3 DEGREES) DOWN FROM THE FAIRED POSITION. ADJUST TRAVEL AS REQUIRED BY RAISING OR LOWERING THE LINK ATTACH BOLT (56) IN THE SLOTTED BRACKET (57) ON THE FLAP. (SEE DETAIL F.) TO DECREASE TRAVEL, RAISE THE BOLT. THE POSITION INDICATOR MAY BE ADJUSTED BY ROTATING THE POSITION TRANSMITTER IN ITS MOUNT. THE FLAP "TAKEOFF/APPROACH" POSITION IS SET BY ADJUSTMENT OF THE 40 PERCENT SWITCHES INSIDE JR1051. (SEE DETAIL F.) THE "40 PERCENT UP" SWITCH SHOULD OPERATE AT 40 (± 5 PERCENT) AS SHOWN ON THE INDICATOR WITH THE FLAP DOWN 20 (±2 DEGREES) WHEN THE FLAP IS OPERATED FROM THE "LANDING" POSITION TO THE "TAKEOFF/APPROACH" POSITION. THE "40 PERCENT DN" SWITCH SHOULD OPERATE BETWEEN THESE SAME LIMITS WHEN THE FLAP IS OPERATED FROM THE "UP" POSITION TO THE "TAKEOFF/APPROACH" POSITION. MOVE THE CONTROL LEVER TO THE "LANDING" POSITION AND GO TO STEP 16.

STEP 16

WITH THE TRAILING EDGE FLAPS FULLY DOWN, DISCONNECT THE ELECTRICAL CONNECTORS TO THE NO. 1 TRAILING EDGE WING FLAP SELECTOR VALVE LOCATED AT FUSELAGE STATION 539.

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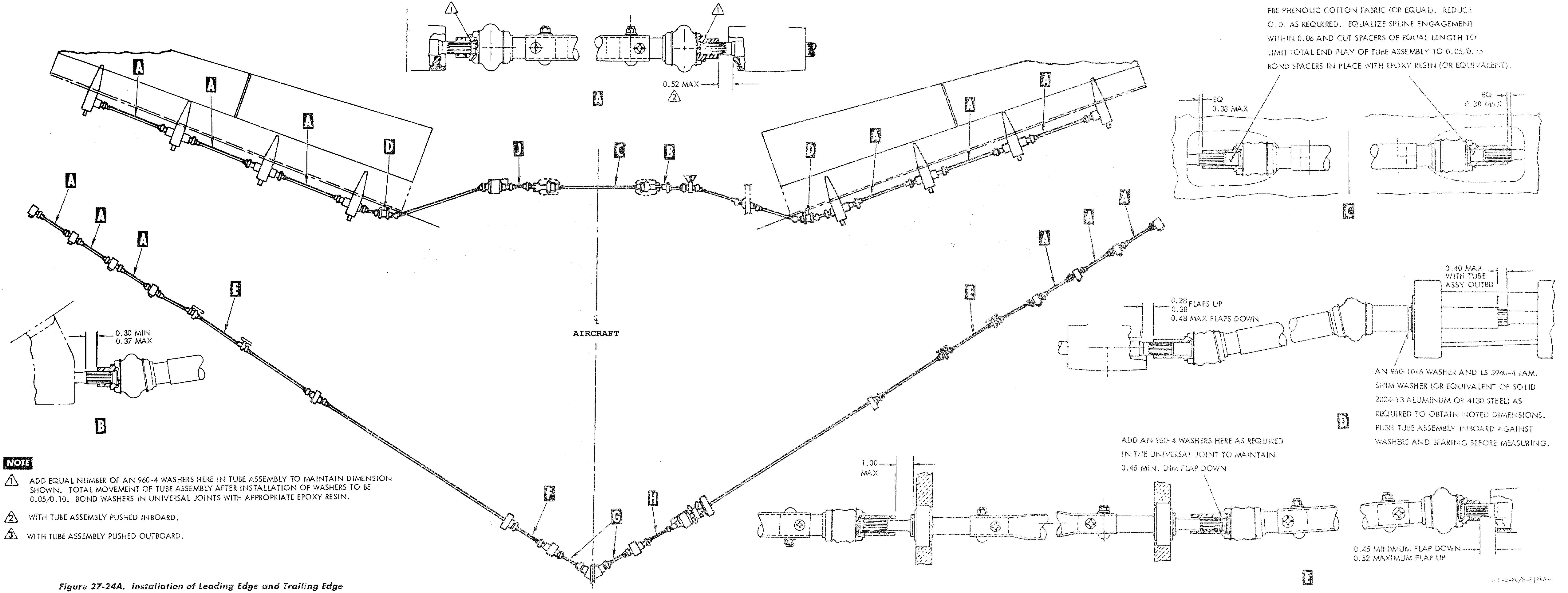
Figure 27-24. Wing Flap Rigging Instructions (Sheet 4 of 5)



0.355 OD X LENGTH AS REQUIRED OF MIL-P-79 TYPE FBE PHENOLIC COTTON FABRIC (OR EQUAL). REDUCE O.D. AS REQUIRED. EQUALIZE SPLINE ENGAGEMENT WITHIN 0.06 AND CUT SPACERS OF EQUAL LENGTH TO LIMIT TOTAL END PLAY OF TUBE ASSEMBLY TO 0.05/0.15 BOND SPACERS IN PLACE WITH EPOXY RESIN (OR EQUIVALENT).

AN 960-1016 WASHER AND LS 5940-4 LAM. SHIM WASHER (OR EQUIVALENT OF SOLID 2024-T3 ALUMINUM OR 4130 STEEL) AS REQUIRED TO OBTAIN NOTED DIMENSIONS. PUSH TUBE ASSEMBLY INBOARD AGAINST WASHERS AND BEARING BEFORE MEASURING.

ADD AN 960-4 WASHERS HERE AS REQUIRED IN THE UNIVERSAL JOINT TO MAINTAIN 0.45 MIN. DIM FLAP DOWN

NOTE

△1 ADD EQUAL NUMBER OF AN 960-4 WASHERS HERE IN TUBE ASSEMBLY TO MAINTAIN DIMENSION SHOWN. TOTAL MOVEMENT OF TUBE ASSEMBLY AFTER INSTALLATION OF WASHERS TO BE 0.05/0.10. BOND WASHERS IN UNIVERSAL JOINTS WITH APPROPRIATE EPOXY RESIN.

△2 WITH TUBE ASSEMBLY PUSHED INBOARD.

△3 WITH TUBE ASSEMBLY PUSHED OUTBOARD.

Figure 27-24A. Installation of Leading Edge and Trailing Edge Flap Torque Tubes (Sheet 1 of 2)

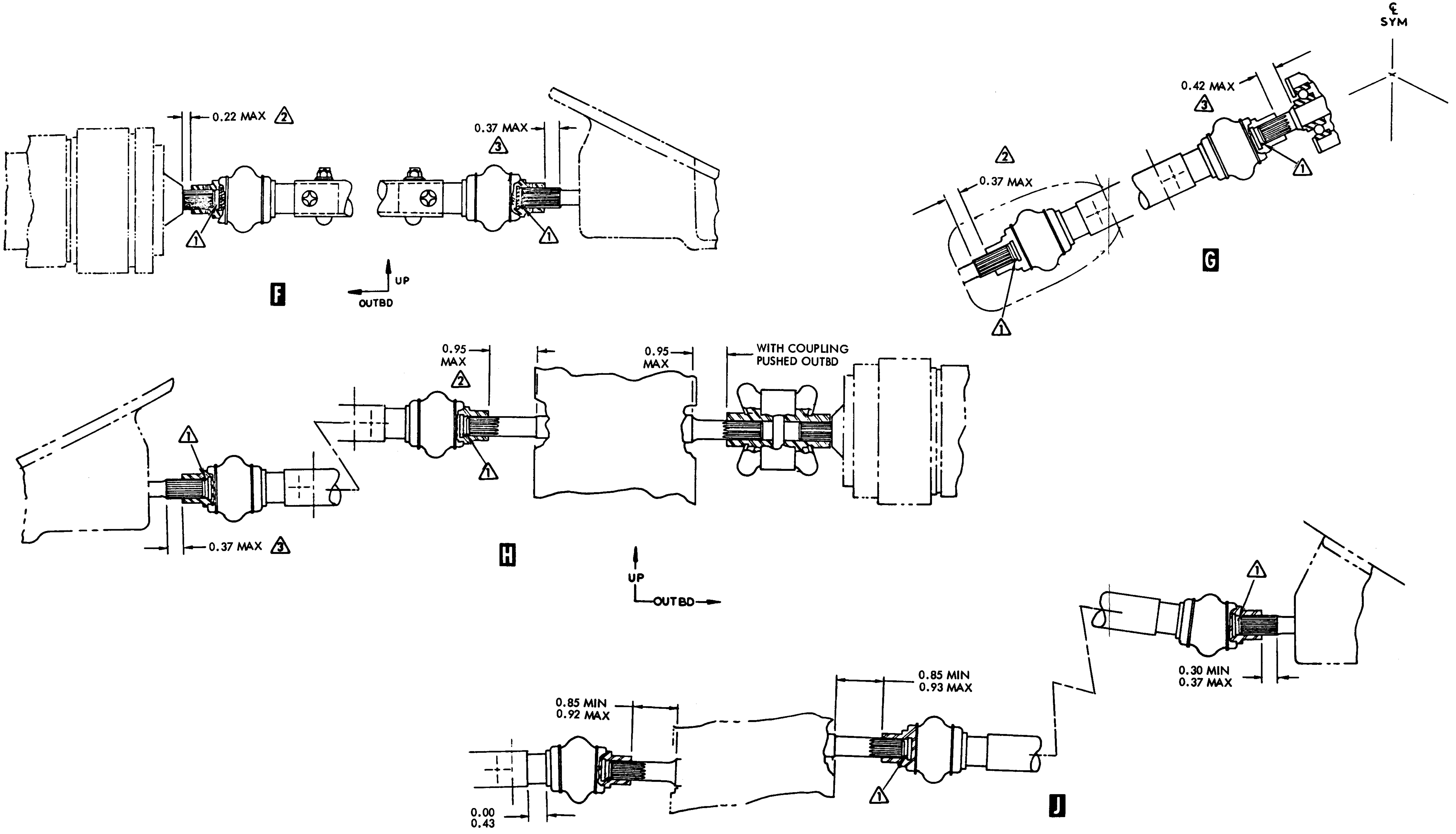


Figure 27-24A. Installation of Leading Edge and Trailing Edge Flap Torque Tubes (Sheet 2 of 2)

JS-1-2-X0/2-2724A-2

SECTION 27.5
SPEED BRAKE SYSTEM

Contents — Page

DESCRIPTION AND OPERATION 27-103
- General 27-103
- Operation 27-103
- Speed Brake Switch 27-104
- Speed Brake Selector Valve 27-104
- Speed Brake Actuating Cylinders 27-104
- Speed Brake Ground Control Handle 27-104
- Speed Brake Uplock Assembly 27-104
- Right Main Gear Squat Switch 27-104
- Speed Brake Warning Relay 27-104
- Speed-Brake-Up Switch 27-104
- Surge Damper 27-104
- Restrictor 27-104
- Check Valve 27-104A

OPERATIONAL CHECKOUT 27-104A
- Speed Brake System 27-104A

TROUBLESHOOTING 27-105
- Speed Brake System 27-105

REMOVAL AND REPLACEMENT 27-107
- Speed Brake Door Actuating Cylinders 27-107

ADJUSTMENT 27-107
- Speed Brake Uplock Assembly 27-107
- Speed Brake Door Step and Gap Requirements 27-107

FIGURES
- 27-25. Speed Brake System Schematic Diagram 27-109
- 27-26. Speed Brake Uplock Assembly 27-110

27.5.1 DESCRIPTION AND OPERATION.

27.5.1.1 GENERAL. (See figure 27-25.) The speed brake system contains a SPEED BRAKE switch, a selector valve, two hydraulic actuating cylinders, a speed brake door, and a speed brake uplock assembly. The speed brake is powered by the normal hydraulic system. The SPEED BRAKE switch energizes the speed brake selector valve to route hydraulic power to the actuating cylinders. When the speed brake down-position is selected, hydraulic power goes to the speed brake uplock cylinder to unlock the door as the actuating cylinders open the door. A speed brake uplock handle is mechanically connected to the uplock cylinder to permit the speed brake door to be manually unlocked from outside the airplane. A speed brake ground control handle can be pulled to the "SAFE" position to prevent accidental closing of the speed brake door. When the speed brake is not fully up and locked with the airplane on the ground, a SPEED BRAKE EXTEND light on the annunciator warning panel goes on and remains steady. When the speed brake is not fully up and locked with the airplane in the air and the gear control handle in the "LG DOWN" position, the SPEED BRAKE EXTEND light flashes.

27.5.1.2 OPERATION. (See figure 27-25.) Placing the SPEED BRAKE switch in the "EXT" position while the speed brake is retracted sends power to the No. 2 solenoid of the speed brake selector valve. This power energizes the valve to unlock the speed brake and extend the speed brake actuating cylinders to open the speed brake. As the speed brake door leaves the uplock, the speed-brake-up switch completes a grounding circuit to the right main gear squat switch and to the speed brake warning relay. If the airplane is in the air and the landing gear control handle is in the "LG DOWN" position, the grounding circuit energizes a keyer in the relay to make the SPEED BRAKE EXTEND warning light on the annunciator warning panel flash. If the landing gear control handle is not in the "LGDOWN" position, or if the airplane is on the ground, the keyer will be bypassed and the SPEED BRAKE EXTEND light will go on and remain steady.

27.5.1.3 SPEED BRAKE SWITCH. A three-position ("RETR," off, "EXT") SPEED BRAKE switch on the center console controls main bus power to the speed brake selector valve. This switch is spring loaded to the center or off position from "RETR". The "EXT" position is maintained.

27.5.1.4 SPEED BRAKE SELECTOR VALVE. The speed brake selector valve is a four-way, three-position, triple-solenoid valve. Energizing the No. 2 solenoid of the valve positions the valve to extend the speed brake door. Energizing the No. 1 and/or No. 3 solenoid(s) positions the valve to retract the speed brake. Power to energize the No. 1 solenoid passes through the speed brake retract circuit breaker. Power to energize the No. 2 or No. 3 solenoid of the valve passes through the speed brake normal circuit breaker. Solenoids No. 1 and No. 3 are both energized at the same time through the same switch. When de-energized, the speed brake selector valve opens both sides of the speed brake actuating cylinders to normal hydraulic system return. This valve is attached to the left hydraulic panel in the aft compartment of the airplane.

27.5.1.5 SPEED BRAKE ACTUATING CYLINDERS. Two double-acting actuating cylinders, one connected to each side of the speed brake door, actuate the speed brake door according to commands from the SPEED BRAKE control switch.

27.5.1.6 SPEED BRAKE GROUND CONTROL HANDLE.

a. On Airplanes 5001 thru 5068, a two position ("SAFE," "NORMAL") handle (inside a door on the left side of the fuselage near the nacelle-to-fuselage attach point) controls a controllable check valve in the down lines of the speed brake actuating cylinders. When in the pulled or "SAFE" position, this handle positions the valve to prevent retraction of the speed brake door. When in the fully-in or "NORMAL" position, this valve has no effect on the speed brake system. This ground control handle must be in the "NORMAL" position to close the cover door over it.

b. On Airplanes 5069 and up, a two-position ("SAFE," "NORMAL") handle is located on the speed brake opening on the left side of the aft bulkhead. It controls a controllable check valve in the down lines of the speed brake actuating cylinders. When in the pulled or "SAFE" position this handle positions the valve to prevent hydraulic retraction of the speed brake. If the speed brake door is closed manually with the handle in the "SAFE" position, the action of the door will push the handle to the "NORMAL" position. This will preclude flights with the handle inadvertently left in the "SAFE" position, in which the speed brake door could be opened but then could not be closed. When in the fully-in or "NORMAL" position, this valve has no effect on the speed brake system.

27.5.1.7 SPEED BRAKE UPLOCK ASSEMBLY. (See figure 27-26.) The speed brake uplock assembly contains an uplock cylinder, a door latch, and an uplock release handle. The uplock cylinder is actuated to the unlocked position when pressure is applied to the door-open sides of the actuating cylinders, and is spring loaded to return to the locked position in the absence of hydraulic pressure. An uplock handle aft of the speed brake opening and behind a door is mechanically attached to the uplock cylinder actuating rod to allow the uplock to be released manually from outside the airplane.

WARNING

When manually unlocking the speed brake door, make sure the ground control handle is in the "SAFE" position. Accidental actuation could result when power is applied to the airplane thereby causing injury to personnel or damage to equipment.

27.5.1.8 RIGHT MAIN GEAR SQUAT SWITCH. The right main gear squat switch completes a grounding circuit to a keyer in the speed brake warning relay when the right main gear is off the ground to supply a flashing warning when the landing gear control handle is in the "LG DOWN" position. This switch is positioned by extension and compression of the landing gear strut assembly.

27.5.1.9 SPEED BRAKE WARNING RELAY. A speed brake warning relay contains a keyer to provide a flashing signal on the annunciator warning panel when the airplane is in the air with the landing gear control handle in the "LG DOWN" position. Power to this keyer is supplied by the warning lights circuit breaker on the pilot's side circuit breaker panel.

27.5.1.10 SPEED-BRAKE-UP SWITCH. A speed-brake-up switch completes a grounding circuit to the SPEED BRAKE EXTEND warning light when the speed brake door is not up and locked. This switch is attached to the airplane structure near the speed brake uplock assembly.

27.5.1.11 SURGE DAMPER. A hydraulic surge damper valve is located in the pressure supply line to the speed brake system. The surge damper protects equipment by reducing the magnitude of pressure ripples in the hydraulic pressure lines.

27.5.1.12 RESTRICTOR. A two-way, 3.0-GPM restrictor is installed in the line to the head side of the speed brake actuating cylinders. This restrictor controls the open and close speed of operation of the speed brake assembly. Additionally, it provides that the uplock will unlock prior to speed brake extension.

27.5.1.13 CHECK VALVE. A check valve is installed in the hydraulic return line downstream of the speed brake selector valve. The primary purpose of this valve is to prevent inadvertent release of the uplock due to transient surges in the normal system return line.

27.5.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Portable Hydraulic System Test Stand	15 GPM at 2,700 PSI Capacity (MIL-H-5606 hydraulic fluid)	To pressurize hydraulic system.

27.5.2.1 SPEED BRAKE SYSTEM.

a. Make sure the SPEED BRAKE NORMAL and the WARNING LIGHTS circuit breakers are engaged.

27.5.2.1 (Cont).

b. Disengage the SPEED BRAKE RETRACT circuit breaker.

27.5.2.1 (Cont).

c. Place the speed brake ground control handle in the "NORMAL" position.

WARNING

Keep clear of the speed brake door when the ground handle is in the "NORMAL" position. Accidental actuation of the speed brake door can injure personnel critically.

d. Connect 28-volt, DC power to the airplane.

e. Connect the hydraulic test stand to the normal hydraulic system ground test connections, and adjust it for 3,000 PSI at zero flow and 2,700 PSI at 5 gallons per minute.

f. With the speed brake door closed, place the SPEED BRAKE switch on the center console in the "EXT" position.
RESULT: The speed brake door should unlock, and the door should open fully within 6 seconds. When the speed brake door leaves the uplock, the SPEED BRAKE EXTEND light on the annunciator warning panel should go on and remain steady.

g. Place the SPEED BRAKE switch in the "RETR" position.
RESULT: The speed brake door should close and lock within 6 seconds, and the SPEED BRAKE EXTEND light should go out.

h. With the landing gear control handle in the "LG DOWN" position, jumper the terminals of the right main gear squat switch to simulate an "in the air" condition. This may be accomplished on all airplanes except 5001 and 5002, by jumpering terminals 6 and 7 of terminal strip H12 located in the right-hand wing. On airplanes 5001 and 5002, jumper terminals 1 and 2 of terminal strip H6 in the right-hand wing to accomplish this.

27.5.2.1 (Cont).

i. Place the SPEED BRAKE switch in the "EXT" position.
RESULT: The speed brake door should extend, and the SPEED BRAKE EXTEND light should start flashing.

j. Remove the jumper wire installed in step (h).

k. Place the speed brake ground control handle in the pulled or "SAFE" position, and place the SPEED BRAKE switch in the "RETR" position.
RESULT: The speed brake door should not move.

l. Return the SPEED BRAKE switch to the "EXT" position; then return the speed brake ground control valve to normal.

m. Disengage the SPEED BRAKE NORMAL circuit breaker, engage the SPEED BRAKE RETRACT circuit breaker, and then place the SPEED BRAKE switch in the "RETR" position.
RESULT: The speed brake door should close and lock.

n. Place the SPEED BRAKE switch in the "EXT" position.
RESULT: There should be no actuation of the speed brake door.

o. Place the SPEED BRAKE switch in the "RETR" position, engage the SPEED BRAKE NORMAL circuit breaker, return the speed brake ground control valve to its original position, reduce hydraulic pressure to zero, and disconnect the hydraulic and electrical power sources from the airplane. Return the speed brake system to normal.

Note

Operational checkout of speed brake system should be accomplished in accordance with the above instructions every 100 flight hours.

27.5.3 TROUBLESHOOTING.

27.5.3.1 SPEED BRAKE SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SPEED BRAKE DOOR FAILS TO OPEN WHEN THE CONTROL SWITCH IS PLACED IN THE "EXT" POSITION.		
Disengaged or defective speed brake normal circuit breaker.	Check the speed brake normal circuit breaker.	Engage or replace the circuit breaker as needed.

27.5.3.1 SPEED BRAKE SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SPEED BRAKE DOOR FAILS TO OPEN WHEN THE CONTROL SWITCH IS PLACED TO THE "EXT" POSITION.		
Defective speed brake selector valve.	Check the No. 2 solenoid of the speed brake selector valve.	Replace the valve as needed.
Speed brake uplock not releasing.	Check the adjustment and operation of the speed brake uplock assembly.	Adjust the assembly or replace components as needed.
SPEED BRAKE DOOR FAILS TO CLOSE WHEN CONTROL SWITCH IS PLACED IN THE "RETR" POSITION.		
Disengaged or defective speed brake normal and speed brake retract circuit breakers.	Check both circuit breakers. If the door will extend, proceed with the next probable cause.	Engage or replace both circuit breakers as needed.
Defective speed brake selector valve.	Check the No. 1 and No. 3 solenoids of the selector valve.	Replace the selector valve as needed.
Speed brake ground control handle in the "SAFE" (pulled) position.	Check the position of the control handle and the operation of the controllable check valve.	Reposition the handle to "NORMAL," or repair or replace the check valve mechanism.
SPEED BRAKE DOOR OPERATES SLOWLY.		
Hydraulic pressure is applied to both sides of an actuator at the same time.	Check each speed brake actuator independently to confirm that hydraulic fluid is not transferring from the pressure side to the return through the actuator piston.	Replace the defective piston, or actuator.
"SPEED BRAKE EXTEND" WARNING LIGHT FAILS TO FLASH WITH AIRPLANE IN THE PROPER CONFIGURATION.		
Defective keyer in the speed brake warning relay.	Check the operation of the relay and the keyer.	Replace the relay as needed.
Improperly adjusted right main gear squat switch.	Check the adjustment and operation of the switch.	Adjust or replace the switch as needed.
Defective or improperly wired landing gear control unit.	Check the wiring and operation of the control unit.	Replace or rewire the control unit as needed.
Defective or improperly adjusted speed-brake-up switch	Check the switch adjustment. If the light goes on as the door opens, proceed with the next probable cause.	Replace or adjust the speed-brake-up switch as needed.
Defective or disengaged warning lights circuit breaker.	Check the warning lights circuit breaker.	Replace or engage the circuit breaker as needed.

Use and/or disclosure is governed by the statement on the title page of this document.

27.5.3.1 SPEED BRAKE SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
"SPEED BRAKE EXTEND" WARNING LIGHT FAILS TO GO ON WHEN SPEED BRAKE DOOR OPENS.		
Defective or improperly adjusted speed-brake-up switch.	Check the speed-brake-up switch adjustment and serviceability.	Replace or adjust the switch as needed.
Defective speed brake warning relay.	Check the operation of the speed brake warning relay.	Replace the warning relay as needed.
SPEED BRAKE EXTEND LIGHT FLASHES INTERMITTENTLY WITH AIRPLANE ON THE GROUND AND THE SPEED BRAKE DOWN.		
Defective right main gear squat switch.	Check the operation of the switch.	Replace the switch as needed.

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27.5.4 REMOVAL AND REPLACEMENT.

27.5.4.1 REPLACEMENT OF SPEED BRAKE DOOR ACTUATING CYLINDERS. Install the cylinders with the rods down. Install the door attaching bolts with their heads outboard. Install the cylinder-to-fuselage attaching bolts with their heads inboard. When installing the bolts, brush a light coat of Specification MIL-G-23827 grease on the shanks of the bolts. Adjust the cylinder rod ends to obtain the proper door uplock adjustment while maintaining the proper door steps and gaps.

27.5.5 ADJUSTMENT.

27.5.5.1 SPEED BRAKE UPLOCK ASSEMBLY. To identify components called out in this procedure, see figure 27-26.

a. With the speed brake door open and the ground control handle in the "SAFE" (pulled) position, place the speed brake handle (7) in the fully-up position so it is bottomed against the hinge stop (10).

b. With the handle (7) in the position of step (a), manually position the actuator shaft (11) in the fully aft (unlocked) position and pull the cotter pin (8) from the tee (9).

c. With the handle (7) in the position of step (a), and with the cotter pin (8) removed, release the actuator shaft (11) to the fully relaxed (locked) position and rotate the tee (9) as necessary to obtain a 0.06-inch gap between the forward face of the tee (9) and the fingers of the handle (7).

d. With the tee properly positioned, and with the hole vertical, hold the actuator shaft (11) in the fully aft position and insert a cotter pin through the tee (9) and the shaft (11). Clip the excess off the cotter pin (8) and partially wrap the clipped ends around the shaft (11).

e. With the actuator shaft (11) still in the extreme aft position, adjust the latch (1) to the vertical position, parallel to the front of the hinge (12).

f. Release the actuator shaft (11), place the speed brake ground control handle in the "NORMAL" (in) position, and retract the speed brake door.

g. With the speed brake door fully retracted, and with the speed brake uplock cylinder (6) in the locked position, adjust the latch (3) to obtain the proper contour steps at the front and rear of the door.

Note

> It may be necessary to adjust the speed brake actuating cylinder rod ends to obtain the proper steps.

h. With the door in the position obtained in step (g), back off the speed-brake-up switch (4) and then lower it until a click is heard. Lower the switch an additional 0.12 to 0.18 inch from this position.

i. Secure all components as required, and then perform an operational checkout of the speed brake system.

27.5.5.2 SPEED BRAKE DOOR STEP AND GAP REQUIREMENTS. The forward step should be from 0.06 inch inside the fuselage contour to 0.03 inch outside the fuselage contour. The rear step should be from 0.03 inch inside to 0.06 inch outside the fuselage contour. The sides of the door should have steps of no more than 0.06 inch inside or outside the fuselage contour.

Note

> To obtain the proper steps, it may be necessary to adjust both the speed brake door actuating cylinder rod ends and the speed brake uplock assembly.

There should be a gap of between 0.06 and 0.19 inch between the perimeter of the door and the fuselage skin. There should be a gap of from 0 to 0.15 inch between the airplane structure and the front of the door-to-fuselage attaching fitting.

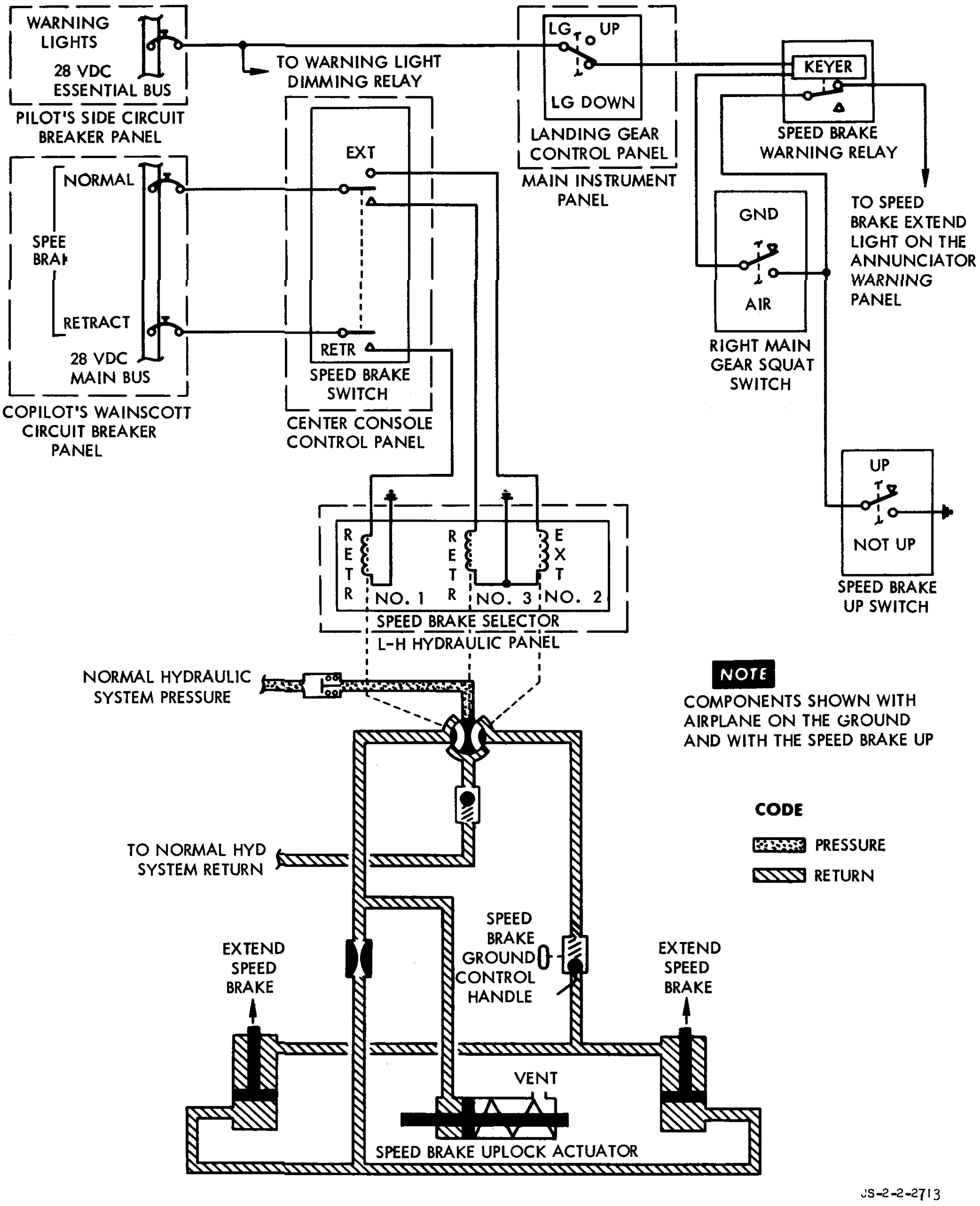
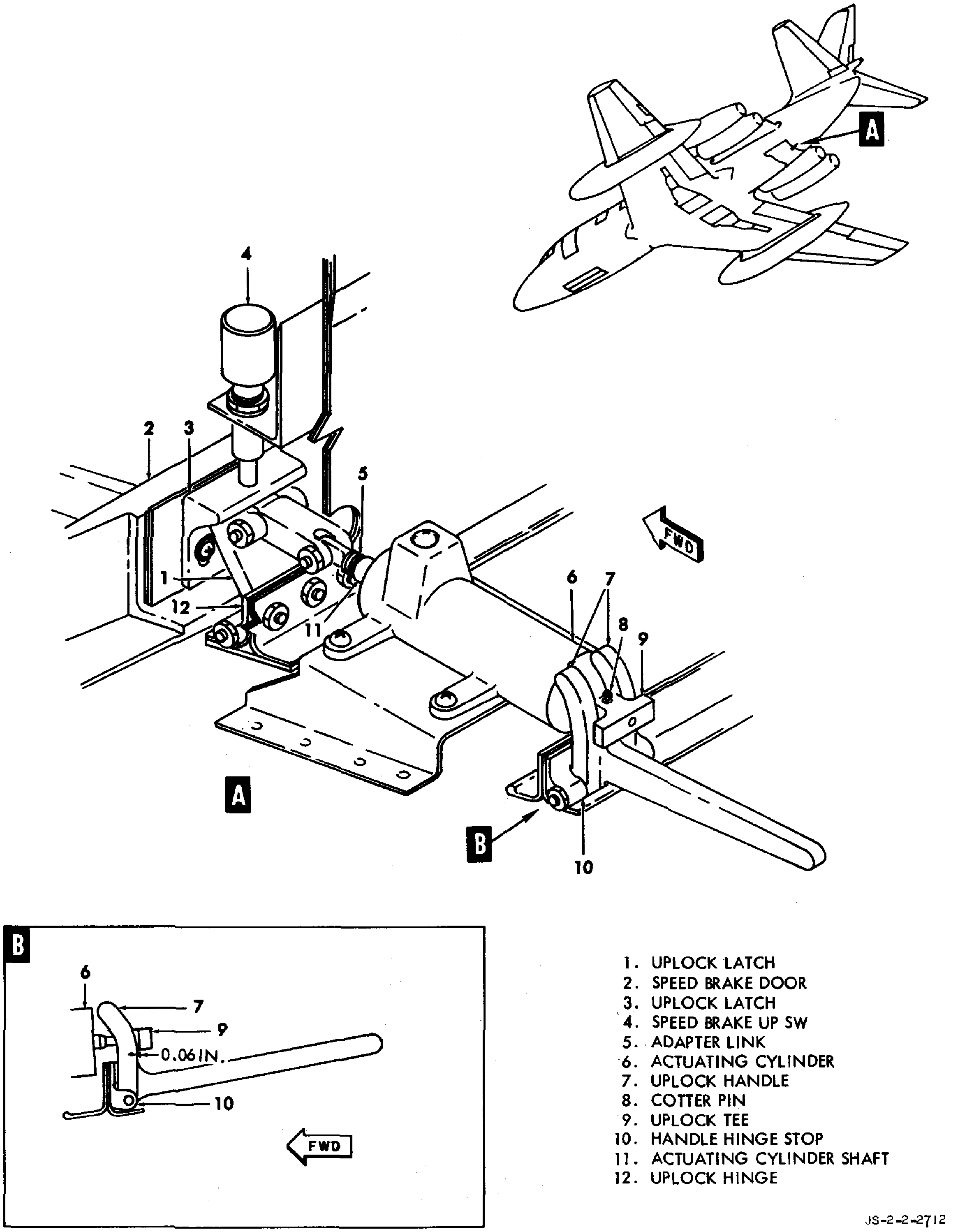

WARNING LIGHTS
28 VDC ESSENTIAL BUS
PILOT'S SIDE CIRCUIT BREAKER PANEL
TO WARNING LIGHT DIMMING RELAY
LG UP
LG DOWN
LANDING GEAR CONTROL PANEL
MAIN INSTRUMENT PANEL
KEYER
SPEED BRAKE WARNING RELAY
TO SPEED BRAKE EXTEND LIGHT ON THE ANNUNCIATOR WARNING PANEL
NORMAL
SPEE BRAI
RETRACT
28 VDC MAIN BUS
COPILOT'S WAINSCOTT CIRCUIT BREAKER PANEL
EXT
RETR
SPEED BRAKE SWITCH
CENTER CONSOLE CONTROL PANEL
GND
AIR
RIGHT MAIN GEAR SQUAT SWITCH
UP
NOT UP
SPEED BRAKE UP SWITCH
RETR
NO. 1
RETR
NO. 3
EXT
NO. 2
SPEED BRAKE SELECTOR
L-H HYDRAULIC PANEL
NORMAL HYDRAULIC SYSTEM PRESSURE
TO NORMAL HYD SYSTEM RETURN
EXTEND SPEED BRAKE
SPEED BRAKE GROUND CONTROL HANDLE
EXTEND SPEED BRAKE
VENT
SPEED BRAKE UPLOCK ACTUATOR
NOTE
COMPONENTS SHOWN WITH AIRPLANE ON THE GROUND AND WITH THE SPEED BRAKE UP
CODE
PRESSURE
RETURN
JS-2-2-2713

Figure 27-25. Speed Brake System Schematic Diagram



1. UPLOCK LATCH
2. SPEED BRAKE DOOR
3. UPLOCK LATCH
4. SPEED BRAKE UP SW
5. ADAPTER LINK
6. ACTUATING CYLINDER
7. UPLOCK HANDLE
8. COTTER PIN
9. UPLOCK TEE
10. HANDLE HINGE STOP
11. ACTUATING CYLINDER SHAFT
12. UPLOCK HINGE

JS-2-2-2712

Figure 27-26. Speed Brake Uplock Assembly

SECTION 27.6
STALL WARNING SYSTEM

DASH 6 JETSTAR

Contents Page

DESCRIPTION AND OPERATION 27-111
- General 27-111
- Operation 27-112
- Stall Warning System No. 1 Operation 27-112
- Stall Warning System No. 2 Operation 27-112
- Stall Warning System Test Circuits 27-112
- Stall Warning System Control Switches 27-112
- Stall Warning System Off-Indicating Lights 27-112
- Transducer Heater Relays 27-112
- Angle-of-Attack Transducers 27-112
- Altitude Pressure Switches 27-113
- Stall Warning and Automatic Ignition Relays 27-113
- Automatic Ignition On Light 27-113
- Shaker Relays 27-113
- Stall Warning Test Relay 27-113
- Control Column Shaker Assembly 27-113
- Test Switches 27-113

OPERATIONAL CHECKOUT 27-113
- Stall Warning System 27-113

TROUBLESHOOTING 27-116
- Stall Warning System 27-116

ADJUSTMENT 27-117
- Control Column Shaker Assembly Clearance Adjustment 27-117
- Transducer Vane Angles 27-117

FIGURES

- 27-27. Auto Ignition System 27-119
- 27-28. Stall Warning System Schematic Diagram 27-121

27.6.1 DESCRIPTION AND OPERATION.

27.6.1.1 GENERAL. (See figure 27-28.) A stall warning system warns the pilots as the airplane nears a stall attitude and, at altitudes above approximately 25,000 feet, supplies automatic engine ignition to prevent engine flameout. Stall warning is provided by a shaker assembly attached to the copilot's control column. Insurance against engine flameout is provided by a circuit which energizes the ignition relays in the engine start and ignition system to provide automatic ignition to all engines.

a. Transducer assemblies attached to the right side of the forward fuselage area have vanes which are exposed to the airstream. These vanes sense airplane attitude in relation to the airstream, and actuate switch contacts in the transducer assembly to control power to the stall warning circuits. Altitude pressure switches select engine ignition according to airplane altitude. Stall-warning-test switches provide a means of checking the system while the airplane is on the ground. A stall warning test relay is energized during testing to prevent ignition power from reaching the engines.

b. The stall warning system is a dual, fail-safe system with components in each half of the system labeled system No. 1 or system No. 2. System No. 2 is the normal operating system with No. 1 system as the back-up. The No. 1 system operates when the airplane has reached a "nose-up" attitude 1.5 degrees higher than is necessary to operate system No. 2.

27.6.1.2 OPERATION. (See figure 27-28.) Control switches on the instrument panel glare shield control 28-volt, DC power to the transducers. Transducers sense the airplane attitude and route power to the stall warning circuits according to airplane altitude. At altitudes above 25,000 (±2,500) feet, a transducer vane angle of 18-1/4 (±1/4) degrees from its counterclockwise stop operates the shaker motor and provides ignition to the engines. At altitudes below 25,000 (±2,500) feet, a transducer vane angle of 25 (±1/4) degrees from its counterclockwise stop operates the shaker motor.

27.6.1.3 STALL WARNING SYSTEM NO. 1 OPERATION. (See figure 27-28.) Placing the NO. 1 STALL WARNING SYS switch in the "ON" position supplies 28-volt, DC power to the No. 1 transducer and energizes the No. 1 transducer heater relay. Energizing the heater relay not only supplies power to the transducer heater, but also breaks the grounding circuit to the No. 1 system off-indicating light. As the transducer vane rotates 18-1/4 (±1/4) degrees from its counterclockwise stop, its contacts supply power to the altitude pressure switch. If the airplane is below 25,000 (±2,500) feet, the circuit stops here. If the airplane is above 25,000 (±2,500) feet, the power passes through the altitude pressure switch to energize the stall warning and automatic ignition relay. Energizing this relay energizes the shaker relay, supplies engine igniter power to the ignition relays through the air ignition control circuit breakers, and supplies power to the AUTO IGN ON light. Pulling a fire emergency control handle blocks ignition power to the corresponding engine. As the transducer vane rotates to a position 25 (±1/4) degrees from its counterclockwise stop, power is routed through the transducer contacts, through the left main gear squat switch (when the squat switch is in the in-the-air position), and energizes the shaker relay. Energizing the shaker relay routes power to the shaker motor. The altitude pressure switch has no effect on this circuit.

27.6.1.4 STALL WARNING SYSTEM NO. 2 OPERATION. (See figure 27-28.) Operation of the No. 2 system is identical to that of the No. 1 system except that power from the 25 (±1/4)-degree contacts of the transducer passes through the right main gear squat switch instead of the left gear squat switch, and power must pass through the de-energized No. 1 shaker relay as well as the energized No. 2 shaker relay to operate the shaker motor.

27.6.1.5 STALL WARNING SYSTEM TEST CIRCUITS. (See figure 27-28.) One "A" and one "B" test switch for each system is provided for ground checkout of the stall warning system. Each switch is a two-position ("TEST," "OFF") switch guarded in the "OFF" position. Placing an "A" test switch in the "TEST" position bypasses the main gear squat switch in the corresponding system. Placing a "B" test switch in the "TEST" position bypasses the altitude pressure switch in the corresponding system to allow testing of the high altitude position of the system while the airplane is on the ground, and also energizes the stall warning test relay to prevent ignition power from reaching the engines during testing. When the automatic ignition functions of the stall warning system are tested, the AUTO IGN ON light goes on as each engine position is selected to show that the circuits are operating properly. The main components of the stall warning system are as follows: two stall warning system control switches, two stall warning system off-indicating lights, two transducer heater relays, two angle-of-attack transducers, two altitude pressure switches, two stall warning and auto ignition relays, one AUTO IGN ON INDICATOR light, two shaker relays, one stall warning test relay, one shaker assembly, two "A" test switches, and two "B" test switches.

27.6.1.6 STALL WARNING SYSTEM CONTROL SWITCHES. A stall warning system control switch for each system controls power to the solenoid of its transducer heater relay and to its angle-of-attack transducer. Placing the switch in the "ON" position energizes the transducer heater relay to remove the ground circuit from the system off-indicating light and also routes power to the heater in the transducer. System control switches are guarded in the "ON" position.

27.6.1.7 STALL WARNING SYSTEM OFF-INDICATING LIGHTS. One red off-indicating light for each system goes on to indicate that the stall warning system is not turned on. The grounding circuit that controls the light passes through the transducer heater relay. When the relay is energized, the light loses its grounding circuit and goes out.

27.6.1.8 TRANSDUCER HEATER RELAYS. One transducer heater relay for each system controls power to the heater in the transducer and also controls the system off-indicating light. These relays are energized by their respective stall warning system control switches.

27.6.1.9 ANGLE-OF-ATTACK TRANSDUCERS. One transducer for each system senses airplane attitude by a vane exposed to the airstream. As the airplane attitude changes, the vane rotates to position

27.6.1.9 (Cont).

switch contacts inside the transducer assembly. These contacts complete circuits at vane angles of 18-1/4 (±1/4) and 25 (±1/4) degrees from their counterclockwise stops. At 18-1/4 (±1/4) degrees, the transducer routes power to the altitude pressure switch. At 25 (±1/4) degrees, the transducer routes power to the shaker relay through the main gear squat switch.

27.6.1.10 ALTITUDE PRESSURE SWITCHES. One altitude pressure switch for each system senses airplane altitude by static air pressure. Switch contacts complete a circuit from the transducer to the stall warning and auto ignition relay at altitudes of 25,000 (±2,500) feet or more. At altitudes of less than 25,000 (±2,500) feet, the switch contacts complete a circuit from the solenoid of the stall warning and auto ignition relay to test switch "B".

27.6.1.11 STALL WARNING AND AUTOMATIC IGNITION RELAYS. One stall warning and automatic ignition relay for each system energizes at altitudes of 25,000 (±2,500) feet or more when the transducer vane is at an angle of 18-1/4 (±1/4) degrees or more to supply ignition to the engines. These relays also control the AUTO IGN ON light on the copilot's control column cover; when either relay energizes, the light goes on. Another function of these relays is to complete a circuit from the system control switch to the shaker relay contacts when the system is turned on and the relay is energized. (See figure 27-27.)

27.6.1.12 AUTOMATIC IGNITION ON LIGHT. One AUTO IGN ON indicator light glows blue when either one of the stall warning and auto ignition relays is energized.

27.6.1.13 SHAKER RELAYS. One shaker relay for each system controls power to the single shaker. The shaker relay energizes when the main gear squat switch is in the "in-the-air" position and the transducer vane is at an angle of 25 (±1/4) degrees, or when the stall warning and auto ignition relay is energized and the system is turned on. Power to the shaker from the No. 1 system passes directly from the shaker relay to the shaker, but shaker power from the No. 2 system must pass through the energized No. 2 shaker relay and then through the de-energized No. 1 shaker relay.

27.6.1.14 STALL WARNING TEST RELAY. One stall warning test relay for both systems makes it possible to test the stall warning system without routing ignition power to non-operating engines. This relay is energized by placing either the No. 1 or the No. 2 "B" test switch in the "TEST" position.

27.6.1.15 CONTROL COLUMN SHAKER ASSEMBLY. One shaker assembly is used for both systems. It consists essentially of an electric motor with an eccentric weight attached to the armature to provide a shaking action when the motor is energized. The shaker is attached to the copilot's control column.

27.6.1.16 TEST SWITCHES. One "A" and one "B" test switch for each system makes it possible to test the stall warning system. Placing the "A" test switch in the "TEST" position bypasses the corresponding main gear squat switch thereby making it possible to test the stall warning system with the airplane on the ground. Placing the "B" test switch in the "TEST" position bypasses the corresponding altitude pressure switch, energizes the stall warning test relay, and completes a circuit from the ENG START SEL switch to the AUTO IGN ON light through either of the stall warning and auto ignition relays.

27.6.2 OPERATIONAL CHECKOUT.

27.6.2.1 STALL WARNING SYSTEM. A protractor to measure the angle of transducer vane rotation, a controllable source of vacuum, a DC voltmeter, and jacks to jack the airplane are necessary to perform this test.

CAUTION

Do not exceed 60 seconds continuous operation of the shaker. After 60 seconds continuous operation, turn the system off for 60 seconds before resuming the test.

a. Disengage the four AIR IGN CONTROL circuit breakers. Leave them disengaged through step (ae).

b. Jack the airplane in accordance with Chapter 7. The wheels must clear the ground. Actuate the main gear squat switches to the "on the ground" position.

c. Install the protractor on the upper (No. 1) transducer vane.

d. Make sure the following circuit breakers are in the positions indicated:

STALL WARN SYS NO. 1	STALL WARN SYS NO. 2	Position
CONTROL	CONTROL	Engaged
WARN LT	WARN LT	Engaged
HEATER	HEATER	Disengaged

Make sure the four ENG IGNITER PWR circuit breakers are engaged.

e. Make sure the system control switches on the glare shield are in the "OFF" positions.

f. Make sure the FIRE EXT AGENT SWITCH is in the "OFF" and guarded position.

27.6.2.1 (Cont).

g. With both transducer vanes against their counterclockwise stops, apply 28-volt, DC power to the airplane service connections, and observe the stall warning system off-indicating lights on the glare shield.
RESULT: The system off-indicating lights should glow red.

h. Place the system No. 1 control switch in the "ON" position.
RESULT: The corresponding off-indicating light should go off.

i. Hold the system No. 1 "A" test switch in the "TEST" position, and rotate the upper transducer vane slowly. Observe the angle of vane rotation necessary to energize the shaker.
RESULT: The shaker should energize when the vane is 25 (±0.5) degrees from its counterclockwise stop.

j. Continue to rotate the vane to its clockwise stop, and observe the shaker operation.
RESULT: The shaker should continue to operate.

k. With the shaker still operating, release the system No. 1 "A" test switch.
RESULT: Shaker operation should cease when the "A" test switch is released.

l. Return the upper vane to its counterclockwise stop; then hold the system No. 1 "B" test switch in the "TEST" position, and rotate the upper vane clockwise to its stop. Observe the angle at which the shaker starts to operate.
RESULT: The shaker should start to operate at 18.25 (±0.5) degrees of vane rotation from its counterclockwise stop. The shaker should continue to operate through step (m).

m. With the system No. 1 "B" test switch still in the "TEST" position, hold the vane at its clockwise stop, and then rotate the ENG START SEL switch on the overhead start panel to engine positions "NO. 1," "NO. 2," "NO. 3," and "NO. 4," observing the AUTO IGN ON Light at each switch selection. When the selector switch is in the "NO. 1" and "NO. 4" positions, pull the corresponding fire emergency control handle and return it to normal.
RESULT: The AUTO IGN ON light should glow blue at each engine selection. The AUTO IGN ON light should go off in the "NO. 1" and "NO. 4" positions when the corresponding fire emergency control handles are pulled; the light should go on again when the fire handles are returned to normal.

n. Release test switch "B" for the No. 1 system.
RESULT: Shaker operation should stop.

o. Rotate the ENG START SEL switch from "NO. 4," to "NO. 3," "NO. 2," "NO. 1," and then "OFF," observing the AUTO IGN ON light at each position.
RESULT: The AUTO IGN ON light should not go on in any of the positions.

p. Return the upper vane to its counterclockwise stop, remove the protractor from the upper vane, and install it on the lower vane.

q. Place the system No. 1 switch in the "OFF" position, and observe the system off-indicating light.
RESULT: The off-indicating light for the No 1 system should go on.

r. Repeat steps (h) through (o) for system No. 2 using the lower transducer vane. The results should be the same, except that they should be for system No. 2.

s. Return the lower vane to its counterclockwise stop, and then remove the protractor from the vane.

t. Perfrom step 34.10.2.6.9 and observe safety precautions as noted in 34.10.2.6.8 pertaining to the third static system. Place both vanes against the clockwise stops and hold them there for the remainder of this test.

CAUTION

Disconnect and plug the third static system tubing at the autopilot's altitude control unit, the autopilot airspeed differential pressure switch, the overspeed warning switch, and the mach trim compensator. Ensure that all plugged system tubing is leak proof per step 34.10.2.6.8.

u. Place the system No. 1 control switch in the "ON" position, and adjust the vacuum source slowly to simulate an altitude of 25,000 (±2,500) feet.
RESULT: At a vacuum simulating 25,000 (±2,500) feet, the shaker should operate.

v. Place the system No. 1 switch in the "OFF" position.
RESULT: The shaker should cease operation.

w. Increase the vacuum to simulate 28,000 feet; then place the No. 2 system control switch in the "ON" position.
RESULT: The shaker should operate.

x. Reduce the vacuum slowly, and note when the shaker ceases operating.
RESULT: The shaker should cease operating when the simulated altitude reaches approximately 25,000 feet.

y. Place the system No. 2 switch in the "OFF" position and allow the third static system to bleed to ambient pressure. Remove the seal and vacuum source from the airplane. Continue with next operational checkout step.

z. Actuate the main gear squat switches to simulate an "in the air" position.

27.6.2.1 (Cont).

aa. Place both the No. 1 and the No. 2 system control switches in the "ON" position.
RESULT: The shaker should operate.

ab. Alternately actuate the left and right main gear squat switches to the "on the ground" position,

27.6.2.1 (Cont)

and observe shaker operation.
RESULT: With both squat switches actuated to the "on the ground" position, the shaker should not operate. With either switch in the "in the air" position, the shaker should operate.

ac. Return the control switches to the "OFF" position, return the vanes to their counterclockwise stops, and return the squat switches to simulate an "on the ground" condition.

ad. Engage the No. 1 and No. 2 system HEATER circuit breakers, and turn on both systems. Observe transducer vane temperatures.
RESULT: Both vanes should become warm to the hand.

ae. Disengage the following circuit breakers:

Stall Warn Sys No. 1 Control

Stall Warn Sys No. 2 Control

No. 1 Eng Igniter Power

No. 2 Eng Igniter Power

No. 3 Eng Igniter Power

No. 4 Eng Igniter Power

af. Engage the four AIR IGN CONTROL circuit breakers and the AIR IGN CONTROL PWR circuit breaker. Make sure the HYD SHUTOFF circuit breakers for engines No. 2 and No. 3 are engaged.

ag. Locate the four igniter relays in the aft compartment at fuselage station 670 and waterline 98, and connect a DC voltmeter to terminal XI of the No. 1 ignition relay and to ground.
RESULT: The voltmeter should indicate zero volts.

ah. Place the ENG START SEL switch in the "NO. 1" position, and hold the AIR START switch in the "AIR START" position. Observe the voltmeter.
RESULT: The voltmeter should indicate 28 volts.

ai. Momentarily place the system No. 1 "B" test switch in the "TEST" position, and observe the voltmeter.
RESULT: The voltmeter indication should drop to zero when the "B" test switch is in "TEST." When the test switch is returned to "OFF," the voltage should return to 28 volts.

aj. Repeat step (ai) using the No. 2 system "B" test switch. The results should be the same.

ak. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 2 ignition relay and to ground.

al. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 2" position. The results should be the same.

am. Pull the fire emergency control handle for engine No. 2 and then return it to normal. Observe the voltmeter.
RESULT: The voltmeter indication should drop to zero when the fire emergency control handle is pulled, and it should return to 28 volts when the handle is restored to its normal position.

an. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 3 ignition relay and to ground.

ao. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 3" position; repeat step (am), using the fire emergency control handle for engine No. 3. The results should be the same.

ap. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 4 ignition relay and to ground.

aq. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 4" position. The results should be the same.

ar. Release the AIR START switch, and place the ENG START SEL switch in the "OFF" position.

as. Disconnect the external power from the airplane, and disconnect the voltmeter from the No. 4 ignition relay.

at. Make sure that the six stall warning circuit breakers are engaged and that the four ENG IGNITER PWR circuit breakers are disengaged.

au. Place the stall warning system No. 1 and No. 2 control switches in the "OFF" position.
RESULT: The two stall warning off-indicating lights should not go on.

av. With the upper transducer vane against its clockwise stop, select airplane battery power. Observe the off-indicating lights.
RESULT: The No. 2 stall warning system off-indicating light should go on.

aw. Place the No. 1 and No. 2 stall warning system control switches in the "ON" position.
RESULT: The No. 2 system off-indicating light should remain on; the No. 1 off-indicating light should remain off. The upper transducer vane should become warm; the lower vane should not.

ax. Place the system No. 1 "A" test switch in the "TEST" position momentarily.
RESULT: The control column shaker should operate.

27.6.2.1 (Cont).

ay. Place both stall warning system control switches in the "OFF" position, return both transducer vanes to their counterclockwise stops, engage the four ENG IGNITER PWR circuit breakers, and return the battery switch to "OFF."

27.6.2.1 (Cont).

az. Return the main gear squat switches to normal, and lower the airplane from the jacks. Return the stall warning system to normal.

27.6.3 TROUBLESHOOTING.

27.6.3.1 STALL WARNING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
STICK SHAKER DID NOT OPERATE. AIRPLANE BELOW 20,000 FEET IN STALL MANEUVER.		
Defective shaker motor, resistor or wiring.	Check motor operation. Check resistor and wiring for open circuit.	Replace shaker assembly or resistor. Repair wiring.
System No. 1 shaker relay contacts defective.	Check relay operation and continuity through contacts.	Replace shaker relay.
STICK SHAKER OPERATES IN FLIGHT WITH AIRPLANE NOT IN STALL CONDITION.		
Defective angle of attack vane.	Alternately switch off system No. 1 and system No. 2.	Replace angle of attack vane.
AIRPLANE ON GROUND, SYSTEM NO. 1, SYSTEM NO. 2 OPERATING OK, AND TURNED OFF: UPPER VANE TO FIRST INDICATING MARK (18 1/4°); TEST SWITCH "A" OFF; SWITCH "B", SYSTEM NO. 1 OPERATED TO TEST. SHAKER DOES NOT OPERATE.		
Circuit breaker pulled or tripped on System No. 1.	Red warning light for system No. 1 is on.	Reset circuit breaker, or check for fault in circuit.
Defective angle of attack vane.	Applying 28 volts, DC to test switch "B" terminals operates shaker.	Replace angle of attack vane.
Defective test switch "B".	Same as above.	Replace test switch.
Defective altitude switch.	Remove disconnect plug from altitude switch. Determine that 28 volts, DC is on pin "B". Jumper pins "A" and "B" to operate shaker.	Replace altitude switch.
Stall warning and auto ignition relay defective, or diode shorted.	Remove disconnect plug from altitude switch. Determine that 28 volts, DC is on pin "B". Jumper pins "A" and "B", but shaker does not operate.	Replace relay or diode.
Defective wiring.	Test for open circuits, grounded circuits, high impedance and incorrect connections.	Repair wiring.

27.6.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SAME AS ABOVE TROUBLE, EXCEPT MAKE SETUP FOR SYSTEM NO. 2 TO OPERATE.		
AIRPLANE ON GROUND, SYSTEM NO. 1 ON, SYSTEM NO. 2 OPERATING OK AND TURNED OFF, UPPER VANE TO SECOND. MARK (25°), TEST SWITCH "B" OFF, TEST SWITCH "A" TO TEST, SHAKER DOES NOT OPERATE.		
Circuit breaker pulled or tripped on system No. 1.	Red warning light for system No. 1 is on.	Reset circuit breaker or check for fault in circuit.
Defective angle of attack vane.	Determine that 28 volts, DC is not on switch terminals. Apply 28 volts, DC to test switch "A" terminals, and shaker operates.	Replace vane.
Defective test switch "A".	Same as above.	Replace test switch.
Shaker relay or diode defective.	Determine 28 volts, DC is on switch terminals.	Replace relay or diode.

27.6.4 ADJUSTMENT.

27.6.4.1 CONTROL COLUMN SHAKER ASSEMBLY CLEARANCE ADJUSTMENT. For these procedures, refer to the elevator rigging instructions in Section 27.1.

27.6.4.2 TRANSDUCER VANE ANGLES. Attachment provisions for transducers automatically position the vanes at the proper angle of attack during transducer installation.

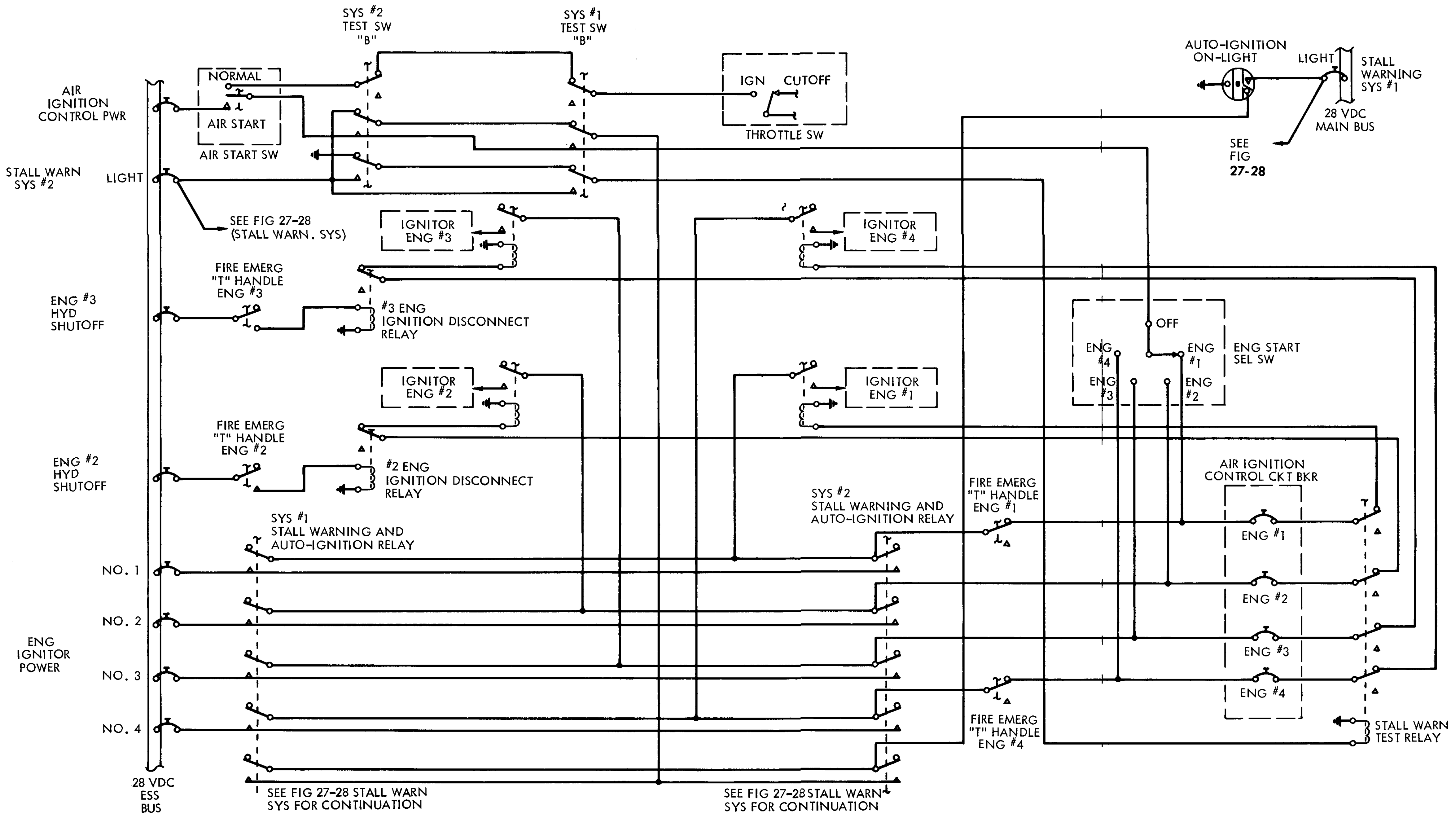


Figure 27-27. Auto Ignition System

Changed 1 April 1968

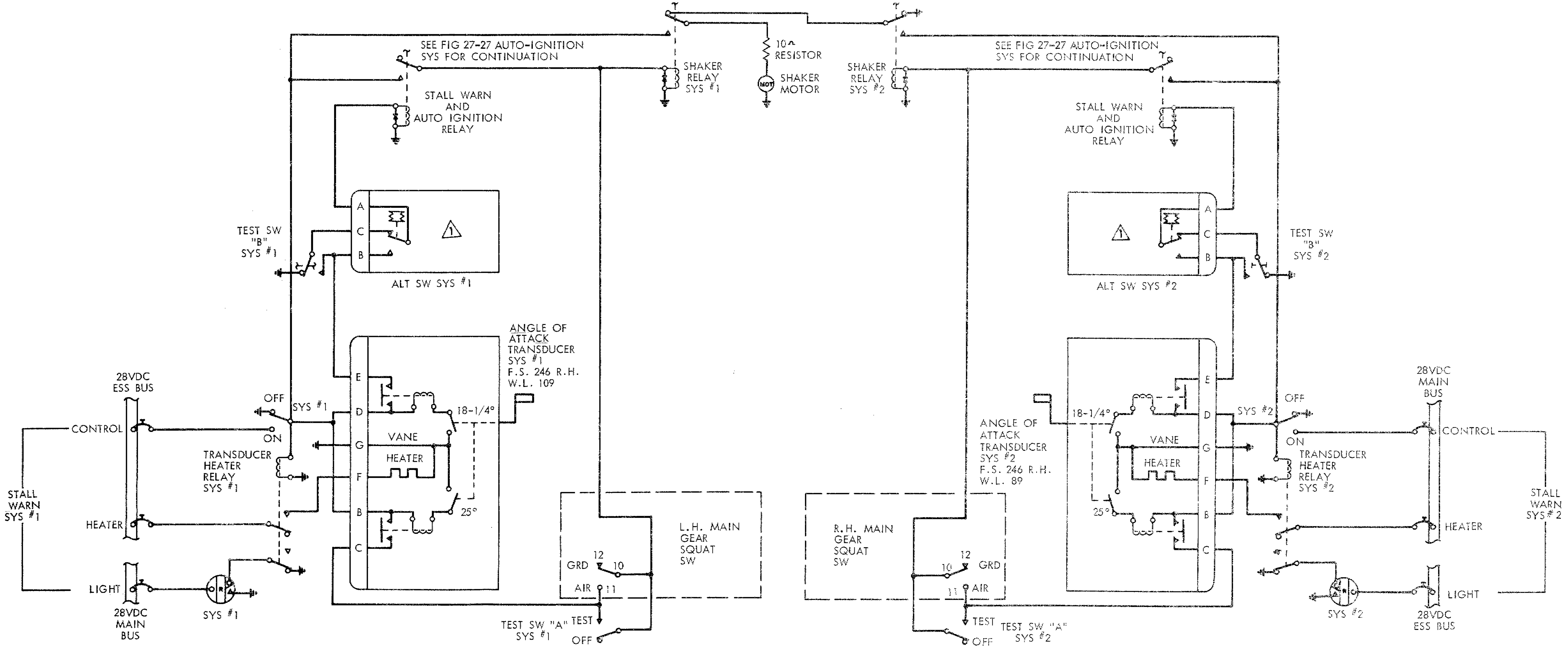


Figure 27-28 Stall Warning System Schematic Diagram (Airplanes 5001 through 5092, 5094 through 5096)

SECTION 27.7

STALL WARNING SYSTEM

DASH 8 JETSTAR

Contents Page

DESCRIPTION AND OPERATION 27-123
General 27-123
Control Panel 27-123
Angle of Attack Transducers 27-124
Transducer Heater Relays 27-124
Altitude Pressure Switches 27-124
Amplifier Switches 27-124
Accelerometers 27-124
Time Delay Relays 27-124
Stall Warning and Automatic Ignition Relays 27-124
Automatic Ignition Indicator Light 27-124
Shaker Relays 27-124
Stall Warning Test Relay 27-124
Shaker Assembly 27-124
Test Switches 27-124
Operation 27-124

OPERATIONAL CHECKOUT 27-125
Stall Warning System 27-125

TROUBLESHOOTING 27-128
Stall Warning System 27-128

ADJUSTMENT 27-130
Control Column Shaker Assembly Clearance Adjustment 27-130
Transducer Vane Angles 27-130

FIGURES
27-29 Stall Warning and Auto Ignition Schematic Diagram 27-131

27.7.1 DESCRIPTION AND OPERATION.

27.7.1.1 GENERAL. (See figure 27-29.) A stall warning system warns the pilots as the airplane nears a stall attitude and, at altitudes above approximately 20,000 feet, supplies automatic engine ignition to prevent engine flameout. Stall warning is provided by a shaker assembly attached to the copilot's control column. Insurance against engine flameout is provided by circuits which energize the ignition relays in the engine start and ignition system to provide automatic ignition to all engines.

a. Transducer assemblies attached to the right side of the forward fuselage area have vanes which are exposed to the airstream. These vanes sense airplane attitude in relation to the airstream, and actuate switch contacts in the transducer assembly to control power to the stall warning circuits. Altitude pressure switches select engine ignition according to airplane altitude. Stall-warning-test switches provide a means of checking the system while the airplane is on the ground. A stall warning test relay is energized during testing to prevent ignition power from reaching the engines.

b. The stall warning system is a dual, fail-safe system with identical components in each half of the system labeled system No. 1 or system No. 2. System No. 2 is the normal operating system with No. 1 system as a back-up. The No. 1 system operates when the airplane has reached a "nose-up" attitude 1.50 degrees higher than is necessary to operate system No. 2.

27.7.1.2 CONTROL PANEL. The stall warning control panel is mounted on the edge of the glareshield on the aircraft center line and consists of the two system power switches and two indicator lights. A red indicator light ON indicates that the system is turned OFF. The lights will extinguish when the systems are turned ON.

27.7.1.3 ANGLE OF ATTACK TRANSDUCERS. One transducer for each system senses airplane attitude by a vane exposed to the airstream. As the airplane attitude changes, the vane rotates to position switch contacts and a potentiometer inside the transducer assembly. The switch contacts close to apply power to the system at vane angles of 18.25 (±.25) and 25 (±.25) degrees, respectively from the counterclockwise stop. The potentiometer is connected to the amplifier switch.

27.7.1.4 TRANSDUCER HEATER RELAYS. One transducer heater relay for each system controls power to the vane heater and also controls the system OFF indicating light. These relays are energized by their respective system control switches.

27.7.1.5 ALTITUDE PRESSURE SWITCHES. Two altitude pressure switches for each system sense airplane altitude by static air pressure. One pressure switch actuates at an altitude of 20,000 ±2,000 feet and the other at an altitude of 32,500 ± 2,500 feet.

27.7.1.6 AMPLIFIER SWITCHES.. There is one amplifier switch assembly in each system. The switch assembly contains solid state circuitry that amplifies the signal received from the transducer potentiometer and closes the normally open contacts when the input signal corresponds to a vane rotation of 15.25 (±.25) degrees from the counterclockwise stop.

27.7.1.7 ACCELEROMETERS. An accelerometer in each system is set to close normally open switch contacts whenever a positive force of 1.3±.1 G's is imposed on the airplane's vertical axis.

27.7.1.8 TIME DELAY RELAYS. A 2PDT time delay relay is used in each system to supply power to the igniters through the 32,500 foot altitude switch. The relay is energized by the accelerometer switch and has a delay of 5 seconds on drop-out when the coil is de-energized.

27.7.1.9 STALL WARNING AND AUTOMATIC IGNITION RELAYS. One stall warning and automatic ignition relay for each system is energized at altitudes between 20,000 feet and 32,500 feet by a transducer switch when the vane is at an angle of 18.25 degrees or more. At altitudes above 32,500 feet the relays are energized by the amplifier switch when the vane is at an angle of 15.25 degrees or more or by the time delay relays when the accelerometer switches close.

27.7.1.10 AUTOMATIC IGNITION INDICATOR LIGHT. One AUTO IGN ON indicator light, located on the copilot's control column cover, glows blue when either one of the stall warning and auto ign relays is energized.

27.7.1.11 SHAKER RELAYS. One shaker relay for each system controls power to the single shaker motor. The shaker relay is energized at altitudes below 20,000 feet by a transducer switch when the vane is at an angle of 25 degrees or more and above 20,000 feet any time one of the stall warning and auto ign relays is energized with the following exception. When the stall warning and auto ign relay is energized above 32,500 feet by the accelerometer and the vane angle is less than 15.25 degrees, the shaker relay will not be energized.

27.7.1.12 STALL WARNING TEST RELAY. One stall warning test relay makes it possible to test either stall warning system without applying power to the igniters of non-operating engines. This relay is energized by placing test switch "B", "C" or "D" of either No. 1 or No. 2 system in the TEST position.

27.7.1.13 SHAKER ASSEMBLY. One shaker assembly is used for both systems. It consists essentially of an electric motor with an eccentric weight attached to the armature to provide a shaking action when the motor is energized. The shaker is attached to the copilot's control column.

27.7.1.14 TEST SWITCHES. Each system has four test switches, labeled "A", "B", "C" and "D", to test the various circuits on the ground. These switches are located on the copilot's control column cover. With the vanes pushed up against the clockwise stops, placing test switch "A" in the TEST position bypasses the corresponding main gear squat switch and energizes the shaker relay through the 25 degree transducer switch. Placing test switch "B" in the TEST position energizes the stall warning and auto ign relay through the 18.25 degree transducer switch and the normally closed contacts of the 20,000 foot altitude switch. Placing test switch "C" in the TEST position energizes the stall warning and auto ign relay through the amplifier switch and the normally closed contacts of the 32,500 foot altitude switch. Placing test switch "D" in the TEST position energizes the time delay relay and the stall warning and auto ign relay through the time delay relay contacts and the normally closed contacts of the 32,500 foot altitude switch.

27.7.1.15 OPERATION. (See figure 27-29.) When the No. 1 and No. 2 system control switches are placed in the ON position, the corresponding indicator lights extinguish and heat is applied to the vanes. The stall warning system is inoperative when the aircraft weight is on the main gear. Once the aircraft is airborne and is below an altitude of 20,000 feet the shaker unit will operate any time the vane angle is above 25 degrees. The shaker relay is energized by the 25 degree switch through the squat switch. In flight the airflow over the two vanes is different and the lower vane, System 2, will engage the shaker first, then after the upper vane, System 1, has moved 1.5 degrees more it will engage the shaker. As the vane angle decreases to below the 25 degree point the shaker will automatically be turned off. When above an altitude of 20,000 feet, the shaker and automatic ignition system will both be engaged when the vane angle is above 18.25 degrees. The stall warning and auto ign relay is energized by the 18.25 degree switch through the 20,000 foot altitude switch and the shaker relay is energized through con-

27.7.1.15 (Cont).

tacts of the stall warning and auto ign relay. As the vane angle decreases below the 18.25 degree point the shaker and igniters will automatically be turned off. When above an altitude of 32,500 feet, the shaker and auto ignition system will both be engaged when the vane angle is above 15.25 degrees. The signal from the transducer potentiometer at a vane angle of 15.25 degrees causes the amplifier switch to actuate supplying power through the 32,500 foot altitude switch to energize the stall warning and auto ign relay and in turn the shaker relay. As the vane angle decreases below the 15.25 degree point the shaker and igniters will automatically be turned off. When above an altitude of 32,500 feet and the vane angle is below 15.25 degrees the auto ignition system will be automatically energized if the aircraft experiences a positive vertical force of 1.3 G's.

a. The accelerometer switch energizes the time delay relay which supplies power through the 32,500 foot altitude switch to energize the stall warning and auto ign relay. As the force decreases below 1.3 G's the igniters will automatically be turned off after a 5 second delay. If, while the igniters are energized by the accelerometer, the vane angle should reach 15.25 degrees, the shaker would also be energized by voltage from the amplifier switch through the time delay relay and the stall warning and auto ign relay. In the event that a fire occurs in one engine in flight, and the fire handle is pulled, the circuit to that igniter relay is opened to prevent the auto-ign relay from energizing the igniter. On engines 1 and 4 this is done by routing the wire through the fire handle switches. On engines 2 and 3 this is accomplished by relays which are energized by voltage from the fire handle switches, when the handles are pulled. The blue auto ignition ON light will illuminate any time the auto ign system is engaged.

27.7.2 OPERATIONAL CHECKOUT.

27.7.2.1 STALL WARNING SYSTEM. A protractor to measure the angle of transducer vane rotation, a controllable source of vacuum, a DC voltmeter, and jacks to jack the airplane are necessary to perform this test.

CAUTION

Do not exceed 60 seconds continuous operation of the shaker. After 60 seconds continuous operation, turn the system off for 60 seconds before resuming the test.

a. Disengage the four AIR IGN CONTROL circuit breakers. Leave them disengaged through step (ae).

b. Jack the airplane in accordance with Chapter 7. The wheels must clear the ground. Operate main gear squat switches to the "on the ground" position.

c. Install the protractor on the upper (No. 1) transducer vane.

d. Make sure the following circuit breakers are in the positions indicated:

STALL WARN SYS NO. 1	STALL WARN SYS NO. 2	Position
CONTROL	CONTROL	Engaged
WARN LT	WARN LT	Engaged
HEATER	HEATER	Disengaged

Make sure the four ENG IGNITER PWR circuit breakers are engaged.

e. Make sure the system control switches on the glare shield are in the "OFF" positions.

f. Make sure the FIRE EXT AGENT SWITCH is in the "OFF" and guarded position.

g. With both transducer vanes against their counterclockwise stops, apply 28-volt, DC power to the airplane service connections, and observe the stall warning system off-indicating lights on the glare shield.
RESULT: The system off-indicating lights should glow red.

h. Place the system No. 1 control switch in the "ON" position.
RESULT: The corresponding off-indicating light should go off.

i. Hold the system No. 1 "A" test switch in the "TEST" position, and rotate the upper transducer vane slowly. Observe the angle of vane rotation necessary to energize the shaker.
RESULT: The shaker should energize when the vane is 25 (±0.5) degrees from its counterclockwise stop.

j. Continue to rotate the vane to its clockwise stop, and observe the shaker operation.
RESULT: The shaker should continue to operate.

k. With the shaker still operating, release the system No. 1 "A" test switch.
RESULT: Shaker operation should cease when the "A" test switch is released.

l. Return the upper vane to its counterclockwise stop; then hold the system No. 1 "B" test switch in the "TEST" position, and rotate the upper

27.7.2.1 (Cont).

vane clockwise to its stop. Observe the angle at which the shaker starts to operate.
RESULT: The shaker should start to operate at 18.25 (±0.5) degrees of vane rotation from its counterclockwise stop. The shaker should continue to operate through step (m).

m. With the system No. 1 "B" test switch still in the "TEST" position, hold the vane at its clockwise stop, and then rotate the ENG START SEL switch on the overhead start panel to engine positions "NO. 1," "NO. 2," "NO. 3," and "NO. 4," observing the AUTO IGN ON Light at each switch selection. When the selector switch is in the "NO. 1" and "NO. 4" positions, pull the corresponding fire emergency control handle and return it to normal.
RESULT: The AUTO IGN ON light should glow blue at each engine selection. The AUTO IGN ON light should go off in the "NO. 1" and "NO. 4" positions when the corresponding fire emergency control handles are pulled; the light should go on again when the fire handles are returned to normal.

n. Release test switch "B" for the No. 1 system.
RESULT: Shaker operation should stop.

o. Rotate the ENG START SEL switch from "NO. 4," to "NO. 3," "NO. 2," "NO. 1," and then "OFF," observing the AUTO IGN ON light at each position.
RESULT: The AUTO IGN ON light should not go on in any of the positions.

p. Return the upper vane to its counterclockwise stop; then hold the system No. 1 "C" test switch in the TEST position and rotate the upper vane clockwise to its stop. Observe the angle at which the shaker starts to operate.
RESULT: The shaker should start to operate and the AUTO IGN ON light should illuminate at 15.25 (±0.5) degrees of vane rotation from its counterclockwise stop.

q. With the shaker still operating, release the system No. 1 "C" test switch.
RESULT: The shaker operation should cease and the AUTO IGN ON light should extinguish.

r. Return the upper vane to its counterclockwise stop; then hold the system No. 1 "D" test switch in the TEST position.
RESULT: The AUTO IGN ON light should illuminate.

s. Rotate the upper vane clockwise to its stop. Observe the angle at which the shaker starts to operate.
RESULT: The shaker should start to operate at 15.25 (±0.5) degrees of vane rotation from its counterclockwise stop. The AUTO IGN ON light should remain illuminated.

t. With the shaker still operating, release the system No. 1 "D" test switch.
RESULT: The shaker operation should cease and the AUTO IGN ON light should extinguish.

u. Return the upper vane to its counterclockwise stop, remove the protractor from the upper vane, and install it on the lower vane.

v. Place the system No. 1 switch in the "OFF" position, and observe the system off-indicating light.
RESULT: The off-indicating light for the No. 1 system should go on.

w. Repeat steps (h) through (o) for system No. 2 using the lower transducer vane. The results should be the same, except that they should be for system No. 2.

x. Return the lower vane to its counterclockwise stop, and then remove the protractor from the vane.

y. Perform step 34.10.2.6.9 and observe safety precautions as noted in 34.10.2.6.8 pertaining to the third static system. Place both vanes against the clockwise stops.

CAUTION

Disconnect and plug the third static system tubing at the autopilot's altitude control unit, the autopilot airspeed differential pressure switch, the overspeed warning switch, and the mach trim compensator. Ensure that all plugged system tubing is leak proof per step 34.10.2.6.8.

z. Place the system No. 1 control switch in the "ON" position, and adjust the vacuum source slowly to simulate an altitude of 20,000 (±2,000) feet.
RESULT: At a vacuum simulating 20,000 (±2,000) feet, the shaker should operate.

aa. Place the system No. 1 switch in the "OFF" position.
RESULT: The shaker should cease operation.

ab. Increase the vacuum to simulate 23,000 feet; then place the No. 2 system control switch in the "ON" position.
RESULT: The shaker should operate.

ac. Reduce the vacuum slowly, and note when the shaker ceases operating.
RESULT: The shaker should cease operating when the simulated altitude reaches approximately 18,000 feet (±2,000).

ad. Place the system No. 2 switch in the OFF position and the system No. 1 switch in the ON position.

27.7.2.1 (Cont).

ae. Increase the vacuum until the shaker operates. Rotate the upper vane counterclockwise just enough to disengage the shaker.

af. Increase the vacuum slowly to simulate an altitude of 32,500 (±2,500) feet.
RESULT: At a vacuum simulating 32,500 (±2,500) feet the shaker should operate.

ag. Place the system No. 1 switch in the OFF position.
RESULT: The shaker should cease to operate.

ah. Reduce the vacuum to simulate an altitude of 23,000 feet. Place the system No. 2 switch in the ON position and the shaker should operate.

ai. Rotate the lower vane counterclockwise just enough to disengage the shaker.

aj. Increase the vacuum slowly to simulate an altitude of 32,500 (±2,500) feet.
RESULT: At a vacuum simulating 32,500 (±2,500) feet the shaker should operate.

ak. Place the system No. 2 switch in the OFF position.
RESULT: The shaker should cease to operate.

al. Allow the third static system to bleed to ambient pressure. Remove the seal and vacuum source from the airplane. Place both vanes against the clockwise stops for the remainder of this test.

am. Actuate the main gear squat switches to simulate an "in the air" position.

an. Place both the No. 1 and the No. 2 system control switches in the "ON" position.
RESULT: The shaker should operate.

ao. Alternately actuate the left and right main gear squat switches to the "on the ground" position, and observe shaker operation.
RESULT: With both squat switches actuated to the "on the ground" position, the shaker should not operate. With either switch in the "in the air" position, the shaker should operate.

ap. Return the control switches to the "OFF" position, return the vanes to their counterclockwise stops, and reinsert the wedges in the squat switches to simulate an "on the ground" condition.

aq. Engage the No. 1 and No. 2 system HEATER circuit breakers, and turn on both systems. Observe transducer vane temperatures.
RESULT: Both vanes should become warm to the hand.

ar. Disengage the following circuit breakers:

Stall Warn Sys No. 1 Control

Stall Warn Sys No. 2 Control

No. 1 Eng Igniter Power

No. 2 Eng Igniter Power

No. 3 Eng Igniter Power

No. 4 Eng Igniter Power

as. Engage the four AIR IGN CONTROL circuit breakers and the AIR IGN CONTROL PWR circuit breaker. Make sure the HYD SHUTOFF circuit breakers for engines No. 2 and No. 3 are engaged.

at. Locate the four igniter relays in the aft compartment at fuselage station 670 and waterline 98, and connect a DC voltmeter to terminal XI of the No. 1 ignition relay and to ground.
RESULT: The voltmeter should indicate zero volts.

au. Place the ENG START SEL switch in the "NO. 1" position, and hold the AIR START switch in the "AIR START" position. Observe the voltmeter.
RESULT: The voltmeter should indicate 28 volts.

av. Momentarily place, in turn, system No. 1 test switches "B", "C" and "D" in the TEST position, and observe the voltmeter.
RESULT: The voltmeter should drop to zero when the "B", "C" or "D" test switch is in the TEST position. When the test switch is returned to OFF, the voltage should return to 28 volts.

aw. Repeat step (ai) using the No. 2 system "B" test switch. The results should be the same.

ax. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 2 ignition relay and to ground.

ay. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 2" position. The results should be the same.

az. Pull the fire emergency control handle for engine No. 2 and then return it to normal. Observe the voltmeter.
RESULT: The voltmeter indication should drop to zero when the fire emergency control handle is pulled, and it should return to 28 volts when the handle is restored to its normal position.

ba. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 3 ignition relay and to ground.

bb. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 3" position; repeat step (am), using the fire emergency control handle for engine No. 3. The results should be the same.

27.7.2.1 (Cont).

bc. Release the AIR START switch, and connect the voltmeter to terminal X1 of the No. 4 ignition relay and to ground.

bd. Repeat steps (ah) through (aj) with the ENG START SEL switch in the "NO. 4" position. The results should be the same.

be. Release the AIR START switch, and place the ENG START SEL switch in the "OFF" position.

bf. Disconnect the external power from the airplane, and disconnect the voltmeter from the No. 4 ignition relay.

bg. Make sure that the six stall warning circuit breakers are engaged and that the four ENG IGNITER PWR circuit breakers are disengaged.

bh. Place the stall warning system No. 1 and No. 2 control switches in the "OFF" position.
RESULT: The two stall warning off-indicating lights should not go on.

bi. With the upper transducer vane against its clockwise stop, select airplane battery power. Observe the off-indicating lights.
RESULT: The No. 2 stall warning system off-indicating light should go on.

bj. Place the No. 1 and No. 2 stall warning system control switches in the "ON" position.
RESULT: The No. 2 system off-indicating light should remain on; the No. 1 off-indicating light should remain off. The upper transducer vane should become warm; the lower vane should not.

bk. Place the system No. 1 "A" test switch in the "TEST" position momentarily.
RESULT: The control column shaker should operate.

bl. Place both stall warning system control switches in the OFF position, return both transducer vanes to their counterclockwise stops, engage the four ENG IGNITER PWR circuit breakers, and return the battery switch to OFF.

bm. Return the main gear squat switches to normal and lower the airplane from the jacks. Restore the aircraft to normal.

27.7.3 TROUBLESHOOTING.

27.7.3.1 STALL WARNING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
STICK SHAKER DID NOT OPERATE. AIRPLANE BELOW 20,000 FEET IN STALL MANEUVER.		
Defective shaker motor, resistor or wiring.	Check motor operation. Check resistor and wiring for open circuit.	Replace shaker assembly or resistor. Repair wiring.
System No. 1 shaker relay contacts defective.	Check relay operation. Check continuity through contacts A2-A3, check continuity through contacts A2-A1 (energized).	Replace shaker relay.
STICK SHAKER OPERATES IN FLIGHT WITH AIRPLANE NOT IN STALL CONDITION.		
Defective angle of attack vane.	Alternately switch off system No. 1 and system No. 2.	Replace angle of attack vane.
SHAKER DOES NOT OPERATE. AIRPLANE ON GROUND, BOTH VANES AGAINST CW STOP. ONE SYSTEM ON, TEST SWITCH "A" IN TEST. OTHER SYSTEM OFF.		
Defective transducer.	Check for continuity between pins B and C.	Replace transducer.

27.7.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SHAKER DOES NOT OPERATE. AIRPLANE ON GROUND, BOTH VANES AGAINST CW STOP. ONE SYSTEM ON, TEST SWITCH "A" IN TEST. OTHER SYSTEM OFF (CONT).		
Defective Test switch "A".	Check switch operation and continuity.	Replace switch.
Defective shaker relay.	Check relay operation and continuity.	Replace relay.
Defective shaker motor, resistor, or wiring.	Check motor operation; check resistor and wiring for open circuit.	Replace shaker assembly or resistor. Repair wiring.
SAME AS ABOVE EXCEPT TEST SWITCH "B" IN TEST.		
Defective transducer.	Check for continuity between pins D and E.	Replace transducer.
Defective 20,000 foot altitude switch.	Check for continuity between pins A and C.	Replace altitude switch.
Defective test switch "B".	Check switch operation and continuity.	Replace switch.
Defective contacts on test switch C.	Check for continuity between terms 2 and 3.	Replace switch.
Defective contacts on test switch D.	Check for continuity between terms 5 and 6.	Replace switch.
Defective 32,500 ft. altitude switch.	Check for continuity between pins A and C.	Replace altitude switch.
Defective stall warning and auto ignition relay.	Determine that 28 volts DC is on term X1 of shaker relay.	
SAME AS ABOVE EXCEPT TEST SWITCH "C" IN TEST.		
Defective test switch "C".	Check switch operation and continuity.	Replace switch.
Defective transducer.	Check operation of potentiometer in transducer (Ref. figure 27-29).	Replace transducer.
Defective amplifier switch.	Determine that 28 volts DC is on term 1 of test switch "C".	Replace amplifier switch.
SAME AS ABOVE EXCEPT TEST SWITCH "D" IN TEST.		
Defective test switch "D".	Check switch operation and continuity.	Replace switch.

27.7.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SAME AS ABOVE EXCEPT TEST SWITCH "D" IN TEST (CONT).		
Defective time delay relay.	Check continuity between pins B & D and G & E, apply 28 volts DC to pin A and ground pin H, and check continuity between pins B & C and G & F.	Replace time delay relay.
AUTO IGNITION LIGHT FAILS TO ILLUMINATE WHEN TEST SWITCH "B", "C" OR "D" IS IN TEST.		
Auto ignition light.	Press to test.	Replace bulb.
Defective contacts on stall warning and auto ign relay.	Determine that 28 volts DC is on term 2 of auto ign light.	Replace relay.
SYSTEM SWITCH ON AND CORRESPONDING RED LIGHT FAILS TO EXTINGUISH.		
Circuit breaker open, defective switch on wiring.	Reset circuit breaker. Check switch operation; check continuity of wiring.	Replace circuit breaker or switch. Repair wiring.

27.7.4 ADJUSTMENT.

27.7.4.1 CONTROL COLUMN SHAKER ASSEMBLY CLEARANCE ADJUSTMENT. For these procedures, refer to the elevator rigging instructions in Section 27.1.

27.7.4.2 TRANSDUCER VANE ANGLES. Attachment provisions for transducers automatically position the vanes at the proper angle of attack during transducer installation.

NOTE

ALL WIRE SHIELDS ARE GROUNDED.

Figure 27-29 Stall Warning And Auto Ignition System Schematic Diagram (Airplanes 5093, 5097 and up)

SECTION 27.8

CONTROL CABLES - GENERAL REQUIREMENTS

Contents Page

General 27-133
Preservation for Prolonged Inactivity 27-133
Preservation 27-133
Depreservation 27-133

Figures
27-30 Control Cable Data 27-134

27.8.1 GENERAL.

Aircraft control cables are made of extra flexible, preformed, corrosion-resistant steel. Cables vary in size from 1/16 to 3/8 inch diameter. Control cables 1/16 and 3/32 inch in diameter consist of 7 strands of 7 wires each and are referred to as 7 × 7 cable. Control cables 1/8 inch in diameter and larger consist of 7 strands of 19 wires each and are referred to as 7 × 19 cable.

The number of wires in a 7 × 7 cable (49) and in 7 × 19 cable (133) is one reason abrupt failure is uncommon. However, progressive failure over extended periods and use can show in service. Tests have shown that cables may have broken wires and still be capable of carrying the design load. Loss of cable strength because of broken wires was proven to be dependent upon the concentration at a specific point rather than on the total number in the cable.

The JetStar control cables should be inspected at regular intervals by passing a cloth along the length of the cable. Broken wires will be evident where the cloth is snagged. Particular attention should be given to areas where control cables pass over pulleys or go through fair-leads.

Any 7 × 19 cable that has three broken wires or less per inch, per strand provided that not more than six broken wires exist in a one inch running length shall be considered serviceable. Any 7 × 7 cable that has two broken wires or less per strand, per inch, provided that not more than three broken wires exist in a one inch length shall be considered serviceable. The maximum number of broken wires shall not occur in any two consecutive inches of cable; that is, if six wires are broken in one inch of 7 × 19 cable, none would be allowed in the next consecutive inch. Any cable not within the above specified limits shall be considered unserviceable and must be replaced. A maximum of three broken wires per inch shall be allowable in the length of cables passing over pulleys, drums, or through fairleads. The critical fatique area is the working length of the cable; therefore, when broken wires appear in this area the cable tension should be released and the cable closely inspected for defects.

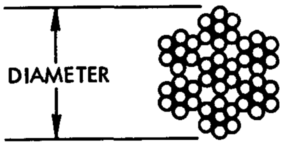
If a corroded or badly worn cable is found during inspection, it should be replaced even if the number of broken wires is less than that specified for replacement. If the serviceability of a corroded cable is in doubt, examine it to see if the corrosion covers only the outer surface or inner strand. Relieve the tension on the cable and carefully untwist the cable to visually inspect the interior. If corrosion covers the inner strand, the cable should be replaced. If corrosion is only present on the surface, the corrosion should be removed with a clean dry rag or fiber brush.

27.8.2 PRESERVATION FOR PROLONGED INACTIVITY.

27.8.2.1 PRESERVATION. For aircraft storage in excess of 30 days, apply brush coat of MIL-G-21164 grease (See figure 12-15) to control cables. Apply a continuous coat but avoid excess.

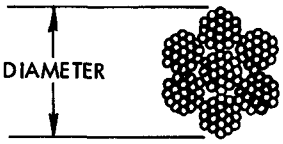
27.8.2.2 DEPRESERVATION. Wipe off excess grease to avoid buildup of dirt and dust in service.

CABLE SPECIFICATIONS			
NOMINAL DIA. OF CABLE	STRANDS BY WIRES/STRAND	MINIMUM BREAKING LOAD - POUNDS	TEST LOAD (APPROX. 60% OF MINIMUM BREAKING LOAD
1/16	7 X 7	480	288
3/32	7 X 7	920	552
1/8	7 X 19	2000	1200
5/32	7 X 19	2800	1680
3/16	7 X 19	4200	2520
7/32	7 X 19	5600	3360
1/4	7 X 19	7000	4200
9/32	7 X 19	8000	4800
5/16	7 X 19	9800	5880
11/32	7 X 19	12500	7500
3/8	7 X 19	14400	8640



7 X 7

(7 STRANDS OF 7 WIRES EACH)



7 X 19

(7 STRANDS OF 19 WIRES EACH)

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Figure 27-30. Control Cable Data

CHAPTER 28
FUEL

TABLE OF CONTENTS

Section		Page
28.0	FUEL SUPPLY SYSTEM	28-3
28.1	FUEL VENT SYSTEM	28-17
28.2	FUEL TANKS	28-21
28.3	FUEL QUANTITY INDICATING SYSTEM	28-45
28.4	DEFUELING AND JETTISONING SYSTEMS	28-55
28.5	FUEL SYSTEM TESTING	28-59

SECTION 28.0
FUEL SUPPLY SYSTEM

Contents Page

DESCRIPTION AND OPERATION 28-3
General 28-3
Fuel Management Panel 28-3
Fuel Warning Lights 28-3
Secondary Fuel Controls 28-4
Fuel Supply System 28-4
Main Tank Boost Pumps 28-5
External Tank Boost Pumps 28-5
Ejectors 28-5
Fuel System Valve Installation 28-5
Fuel Line Check Valves 28-5
Tank Interconnect Valves 28-5
Fuel Air Bleed Valve 28-5

TROUBLESHOOTING 28-6

REMOVAL AND REPLACEMENT 28-7
Main Tank Boost Pumps 28-7
External Tank Boost Pumps 28-8
Ejectors 28-8
Sump Low Level Warning Switch 28-8
Fuel Line Check Valves 28-8
Tank Interconnect Valves 28-8
Line Couplings 28-8

FIGURES
28-1. Fuel System Components Locations 28-10
28-2. Fuel System Flow Diagram 28-13
28-3. External Tank Pump Installation 28-14
28-4. Fuel System Schematic Diagram 28-15
28-4A. Flared Tube Fitting Torque Values 28-16A

28.0.1 DESCRIPTION AND OPERATION.

28.0.1.1 GENERAL. (See figure 28-1.) All the fuel is held in four integral wing tanks and two external, wing-mounted tanks. The main tanks in each wing are connected by a tank interconnect line and valve. All six fuel tanks are connected by a crossfeed manifold. The primary controls for the fuel supply system are on the fuel management panel convenient to the pilot and copilot. The secondary controls are on the copilot's side of the main instrument panel, adjacent to the fuel management panel, and on the copilot's control column cover. A fuel boost pump in each tank supplies fuel to the engine-driven fuel pumps. A fuel jettison system is an integral part of the fuel system.

28.0.1.2 FUEL MANAGEMENT PANEL. Located on the fuel management panel are a fuel quantity indicator for each main tank, a dual-pointer fuel quantity indicator for both external tanks, four crossfeed valve switches, a crossfeed isolation shutoff valve switch, a LOW FUEL PRESS warning light for each engine, a PRESS LOW light for each external tank, a BOOST PUMP switch for each tank, and a LOW SUMP LEVEL warning light for each main tank. A schematic diagram on the front of the panel connects all fuel system switches, indicators, and warning lights.

28.0.1.3 FUEL WARNING LIGHTS.

a. The PRESS LOW warning light for each external tank goes on when the external tank fuel boost pressure drops to between 25-1/2 and 24 psi. The LOW FUEL PRESS warning light for each engine goes on when the engine fuel supply pressure is below 8 (±1) PSI. A pressure switch at the boost pump outlet, or a second pressure switch at the engine inlet, turns the light on according to the location at which low pressure exists. The boost pump outlet pressure switch is wired through

28.0.1.3 (Cont).

the boost pump control switch, and is inoperative when the boost pump is turned off. The light glows steadily to indicate either of these conditions. The same light is also connected, through a flasher unit, to a differential pressure switch across the boost stage of the engine-driven fuel pump. A flashing light thus indicates a failure of the engine-driven pump boost stage.

b. During operation on external tank fuel, with main tank boost pumps also operating, the pressure switch in each main tank boost pump discharge line remains open so long as main tank boost pump pressure is normal. Should this pump pressure drop below the pressure switch setting, the pressure switch closes and turns on the applicable LOW FUEL PRESS light on the fuel management panel. If the corresponding main tank boost pump is then switched off and the light goes out, loss of main tank boost pump pressure is confirmed. If the light had been turned on by low fuel pressure at the engine inlet, switching off the main tank boost pump would have had no effect on the light. If an engine-driven pump boost stage failure had turned the light on, this could have been identified by the light flashing instead of glowing steadily.

c. The LOW SUMP LEVEL warning light for each main tank goes on when 300 (±100) pounds of fuel remain in the sump.

28.0.1.4 SECONDARY FUEL CONTROLS. The secondary controls consist of the L and R FUEL JETTISON switches; the No. 1, No. 2, No. 3 and No. 4 FUEL TANK ISOLATION switches; and the 1 & 2 and the 3 & 4 MAIN TANK INTERCONNECT switches. The FUEL JETTISON switches control the fuel jettison valves. Each FUEL TANK ISOLATION switch controls an isolation valve, and the MAIN TANK INTERCONNECT switches control the main tank interconnect valves. The secondary fuel control switches are on the copilot's side of the main instrument panel except for the MAIN TANK INTERCONNECT switches which are on the copilot's control column cover.

28.0.1.5 FUEL SUPPLY SYSTEM.
(See figures 28-2 and 28-4.)

WARNING

Prolonged operation of fuel boost pumps without fuel in the tank shall be avoided.

WARNING

Whenever circuit breakers open or have been found open in any pump circuitry, remove power from the airplane immediately and accomplish a thorough continuity check and a visual check of exposed circuitry components.

a. Each of the four integral wing tanks normally supplies fuel directly to its corresponding engine. Each external tank is connected directly to the crossfeed manifold, through which it can supply fuel to any or all of the engines. The fuel boost pumps in the main tanks remain on when operating the fuel system under normal conditions. The crossfeed valves are opened and the external tank boost pumps turned on to use the fuel in the external tanks.

b. The minimum output pressure of the external tank boost pumps is greater than the maximum output pressure of the main tank boost pumps. During the operation of the main and external tank pumps, all the fuel consumed by the engines comes from the external tanks. Check valves in the fuel lines between the crossfeed manifold and the main tanks prevent the external tank boost pumps from emptying fuel into the main tanks. When the external tank fuel supply is exhausted and the output pressure of the external tank boost pump decreases below the output pressure of the main tank boost pumps, or the external tank pump is turned off, the main tank pumps automatically supply the engines. Check valves in the fuel lines from the external tanks prevent fuel from entering the external tanks after the boost pumps are turned off. The check valves in the discharge lines from the main and external tanks incorporate orifices for thermal relief of the lines when downstream valves are closed.

c. The crossfeed manifold connects all the fuel tanks and engine fuel supply lines. The crossfeed valves control the flow of fuel through the crossfeed manifold. The crossfeed separation valve divides the crossfeed manifold between the No. 2 and the No. 3 engine fuel supply lines and remains closed except when an abnormal condition requires it to be opened to maintain spanwise balance. The fuel lines from the external tanks connect to the crossfeed manifold on each side of the crossfeed separation valve.

d. The main fuel tank isolation valves permit any main fuel tank to be isolated from the crossfeed manifold, so engines may be supplied from the external tanks by suction feed, if an external tank boost pump fails. For example, fuel may be fed from the left external tank to No. 1 engine by closing No. 1 tank isolation valve and No. 2 crossfeed valve, and opening No. 1 crossfeed valve. The No. 1 engine then obtains fuel by suction feed from the left external tank, even though the boost pump in that tank may be inoperative. Both No. 1 and No. 2 engines may similarly be supplied from the left external tank by closing

28.0.1.5 (Cont).

No. 1 and No. 2 tank isolation valves and opening No. 1 and No. 2 crossfeed valves, or fuel may be supplied to No. 2 engine from the left external tank by closing No. 2 tank isolation valve and No. 1 crossfeed valve and opening No. 2 crossfeed valve. The No. 3 and No. 4 tank isolation valves permit similar operation on the right side of the airplane.

e. Shutoff valves installed in the engine fuel supply lines shut off the fuel supply to prevent fuel from feeding an engine fire. The fire emergency handles control these valves.

CAUTION

The engine-driven fuel pumps are lubricated by the fuel they are pumping. Always retard

28.0.1.5 (Cont).

the throttle to the "OFF" position before pulling the fire emergency handle, if time permits. This will prevent the pumps from running dry. If the engine is shut down with the fire emergency handle, inspect all engine fuel pump control filters for evidence of fuel pump damage.

28.0.1.6 MAIN TANK BOOST PUMPS. The fuel tank boost pumps are 28-volt, DC motor-driven submerged pumps with a discharge pressure of 18 to 25 PSI (5001 thru 5045) or 17 to 24 PSI (5046 and up) at a flow of 0 to 7,300 pounds per hour.

28.0.1.7 EXTERNAL TANK BOOST PUMPS. The external fuel tank boost pumps are 28-volt, DC, motor-driven submerged pumps with a discharge pressure of 27 to 38 PSI (on airplanes 5001 thru 5045) or 30 to 38 PSI (on airplanes 5046 and up) at a flow of 0 to 6,200 pounds per hour.

28.0.1.8 EJECTORS. Three ejectors are provided in each main tank using approximately 10 gallons of fuel per minute for motive or primary flow. This flow is supplied from the corresponding tank boost pump through the ejector check valve. Ejector operation is continuous whenever the pump is operating with fuel in the tank. The fuel transfer ejector induces a flow from the tank into the sump to keep the sump full. The vent drain ejector transfers the fuel that enters the vent system from the vent drain tank or compartment into the sump. The sub-sump ejector transfers fuel from the outboard, aft corner of the sump into the sub-sump at the pump inlet to reduce the quantity of trapped fuel during climb at extremely low fuel levels. The fuel transfer ejector and the vent drain ejector are provided with integral check valves in the induced flow line to prevent draining the sump when the boost pump is not operating. The sub-sump ejector does not have a similar check valve on airplanes 5041 and up.

28.0.1.9 FUEL SYSTEM VALVE INSTALLATION.

a. Except as indicated in paragraph 28.0.1.12, all the shutoff valves shown in figure 28-2 are located in an enclosed valve compartment in the aft fuselage compartment. This compartment is vented, and it is drained overboard through a drain mast on the fuselage bottom skin. A vapor-proof barrier isolates the fuel valves from the remainder of the aft compartment. Access to the valves is through a sealed-zipper fabric closure (on airplanes 5001 through 5045) or through a removable panel (on airplanes 5046 and up). From the valve compartment, fuel lines to the engines are routed through the aft compartment in rubberized fabric sleeves to prevent escape of fuel or fuel vapor in the event of a line or coupling leak.

b. The tank isolation valves, the crossfeed valves and the engine shutoff valves are all 1-1/4-inch tube size, 28 volt DC, motor operated, sliding gate valves with provision for pressure relief in case of thermal expansion of the fuel trapped in the lines. These valves are normally controlled from the cockpit, but, for ground maintenance, a lever is provided on the valve for manual operation.

c. The crossfeed separation valve and the jettison valves are the same as above except that they are 1-1/2-inch tube size with no provision for pressure relief.

d. The manual defuel valve is a 1-1/2-inch tube size, manually operated, sliding gate valve. The operating lever is accessible through a door on the underside of the fuselage, as is the attaching point for connecting a defueling hose or line. A restricting orifice is installed with this valve to prevent defueling at fuel flow rates that could damage the fuel pumps.

e. The jettison line drain valve is provided to drain fuel and condensate from the jettison line. This valve is a "push to drain" valve that may be operated from below the fuselage.

WARNING

Each time the jettison line drain valve is used, it should be checked to see that there is no binding in the grommet where it passes through the fuselage skin. Binding at this point can keep the valve spring from closing the valve.

28.0.1.10 FUEL LINE CHECK VALVES. In the fuel line from each tank is a 1-1/4-inch check valve. This valve is provided with a small orifice in the flapper for thermal relief.

28.0.1.11 TANK INTERCONNECT VALVES. These are 1-1/2-inch tube size, 28 volt DC, motor operated, sliding gate valves with lever for manual operation. Each valve is in a line connecting the forward and aft tank sumps in each wing. The tank interconnect valves are normally kept closed, but may be opened in case of malfunction of other valves which traps fuel in one of the main tanks and prevents it being used. Under these conditions, to assure fuel flow from the tank in which it is trapped to the tank from which it can be used, the interconnect valve is only opened when there is less than approximately 400 pounds of fuel remaining in the usable tank.

28.0.1.12 FUEL AIR BLEED VALVE. A 1/4-inch line size, 28 volt DC, solenoid operated shutoff valve is provided at each engine fuel inlet to vent

28.0.1.12 (Cont).

an air or vapor locked condition in the engine fuel feed line. Such a condition can occur when a tank is inadvertently run dry in flight, when an empty system is refueled, or from other causes. This valve simply provides a method for bleeding the fuel line overboard at the engine so that any air or vapor in the line may be removed. Its use is recommended whenever difficulty in starting an engine is encountered. The valves are controlled by a single, guarded FUEL AIR BLEED LINE switch on the overhead start panel, and by the five-position ENG SEL switch. The ENG SEL switch determines which one of the four valves will be operated when the FUEL AIR BLEED LINE switch is placed in the "OPEN" position. In this position, the switch energizes an AIR BLEED OPEN light on the panel to alert the crew.

28.0.2 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FUEL SHUTOFF VALVES FAIL TO OPERATE.		
Defective control switch.	Perform continuity and visual check of affected switch and circuit.	Repair circuit or replace switch as necessary.
Valve motor burned out.	If the above continuity check shows power to the motor terminals and the connection is tight, the motor is defective.	Replace defective valve.
Motor drive shaft is sheared.	With electric power connected to the airplane, switch the valve control from one position to another and see if motor runs.	Replace the valve if motor operates but valve does not change positions.
PUMP FAILS TO PRODUCE REQUIRED FLOW OR PRESSURE.		
Low voltage at pump terminals.	Check voltage at bus.	If bus voltage is low, troubleshoot the electrical supply system.
Clogged pump inlet strainer.	Remove the pump and visually inspect the strainer.	Clean the strainer.
Defective pump.	If voltage is correct and strainer is not clogged, the pump is defective.	Replace defective pump.
PUMP FAILS TO OPERATE.		
Defective switch on fuel management panel.	Perform visual and continuity check of switch.	Replace defective switch.
Defective pump.	If above continuity check shows power to the terminals, the pump is defective.	Replace defective pump.
EXTERNAL FUEL TANK LOW PRESSURE WARNING LIGHT FLICKERS INTERMITTENTLY.		
External tank fuel low pressure switch out of adjustment.	Substitute a new pressure switch.	Replace pressure switch.
External tank boost pump fails to produce sufficient pressure.	See above.	See above.

28.0.2 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW SUMP LEVEL WARNING LIGHT COMES ON WITH 400 OR MORE POUNDS OF FUEL IN TANK.		
Inoperative fuel transfer ejector.	Substitute a new ejector.	Replace fuel transfer ejector.
Leaky tank flapper check valves.	Locate leaky valve or valves.	Repair or replace leaky valves.
FUEL LEAKAGE BETWEEN MAIN TANKS.		
Leak in inter-tank seal.	Check for tank-to-tank leakage.	Replace or repair seal.
FUEL QUANTITY INCREASES IN EXTERNAL TANK WITH MAIN TANK PUMPS RUNNING.		
Leak in external tank fuel line check valve.	Substitute new check valve.	Replace check valve.

28.0.3 REMOVAL AND REPLACEMENT.

28.0.3.1 MAIN TANK BOOST PUMPS. The main tank boost pumps are installed on the inside of a boost pump access door in the inboard tank bulkhead, between the wing and the fuselage. Access to the inboard tank bulkhead is provided by removing a section of the wing lower fillet.

28.0.3.1.1 REMOVAL.

a. Defuel the tank, remove the section of wing fillet beneath the boost pump access door, and disconnect the electrical connector from the pump.

b. Disconnect the ejector fuel supply line, the fuel fitting on the pump outlet line, and the flexible pump drain hose.

c. Remove the boost pump access door.

Note

On tanks No. 1 and No. 4, the main access door will not come out of the tank with the boost pump attached. Remove the small access door on the main door and reach into the tank to support the pump with one hand while removing the pump mounting bolts with the other hand. When the pump is free, lay it on its side on the bottom of the tank out of the way of the main access door. Then unbolt and remove the main access door to remove the pump from the tank.

d. Remove the boost pump and O-ring from the access door.

28.0.3.1.2 SAFETY PRECAUTIONS. Observe all of the safety precautions contained in Chapter 12, paragraph 12.1.1, and paragraph 28.2.1.4 of this chapter prior to removal and replacement of components.

WARNING

Observe the following LEL (lower explosive limit) vapor concentration levels during fuel tank maintenance work. Above 20% LEL is not considered "firesafe". Purge tanks with clean air down to 20% LEL before opening. Between 20% & 5% LEL enter tanks only with full face, air supplied respirators. Between 5% & 1.5% LEL use half face, vapor chemical cartridge respirators. Below 1.5% LEL plus 19.5% oxygen is considered "healthsafe".

28.0.3.1.3 INSTALLATION. Replace the O-ring, and install the boost pump in reverse order of removal.

Note

Fuel boost pumps having shaft seal leakage may not necessarily need replacement. Observe leakage from the seal drain before taking further action. If there is a steady drip, or if leakage exceeds one drop per minute, replace the pump. Pumps having leakage rates of less than one drop per minute should be operated, and the leakage rate again checked. If the leakage rate is more than three drops in 10 minutes after 30 minutes of pump operation, replace the pump. Note that, after periods of inactivity, pumps should be operated for several hours before replacing them for shaft seal leakage. (This is applicable to both main and external tank pumps.)

28.0.3.2 EXTERNAL TANK BOOST PUMPS.

28.0.3.2.1 REMOVAL.

a. Defuel the tank, and remove the boost pump access door.

b. With the wing flap lowered, disconnect the electrical leads from the pump at the terminals in the tank fairing. Feed the leads back through the conduit.

c. Remove the bolt and cap securing the outlet fitting (see figure 28-3).

d. Remove the bolts that secure the pump to the pump mounting bracket.

e. Remove the pump and the pump seal.

28.0.3.2.2 INSTALLATION. Replace the pump seal, and install the boost pump in reverse order of removal.

CAUTION

Do not exceed 30 inch-pounds torque on the 1/4 inch aluminum mounting bolts.

Note

The open side of the scarfed seal drain mast should be facing aft when installation is complete.

28.0.3.3 EJECTORS. All fuel ejectors are mounted inside the fuel tank and may be removed and replaced with the boost pump access door removed. (See paragraph 28.0.3.1.1)

28.0.3.4 SUMP LOW LEVEL WARNING SWITCH. This switch is mounted inside the fuel tank and may be removed and replaced with the boost pump access door removed (see paragraph 28.0.3.1.1)

28.0.3.5 FUEL LINE CHECK VALVES. The check valves in the lines from the No. 1, No. 2 and left external tank are accessible through the left fuel check valve access door located in the lower aft wing fillet. Valves from No. 3, No. 4 and right external tanks are located in the right fuel check valve access door.

28.0.3.6 TANK INTERCONNECT VALVES. Removal of the lower left wing fillet is required to obtain access to the interconnect valve between tanks 1 and 2. Removal of the right fillet will give access to the valve between tanks 3 and 4.

28.0.3.7 LINE COUPLINGS. Installation of the following fuel line couplings is to be made with the torque value shown:

Coupling Type	Size (Inches)	Torque (Inch-Pounds)
3605-12D	3/4	60-84
3605-16D	1	120-156
3605-20D	1 1/4	180-216
3605-24D	1 1/2	324-360
3605-28D	1 3/4	348-384

Note

Torque values for flared tube fittings are shown in figure 28-4A.

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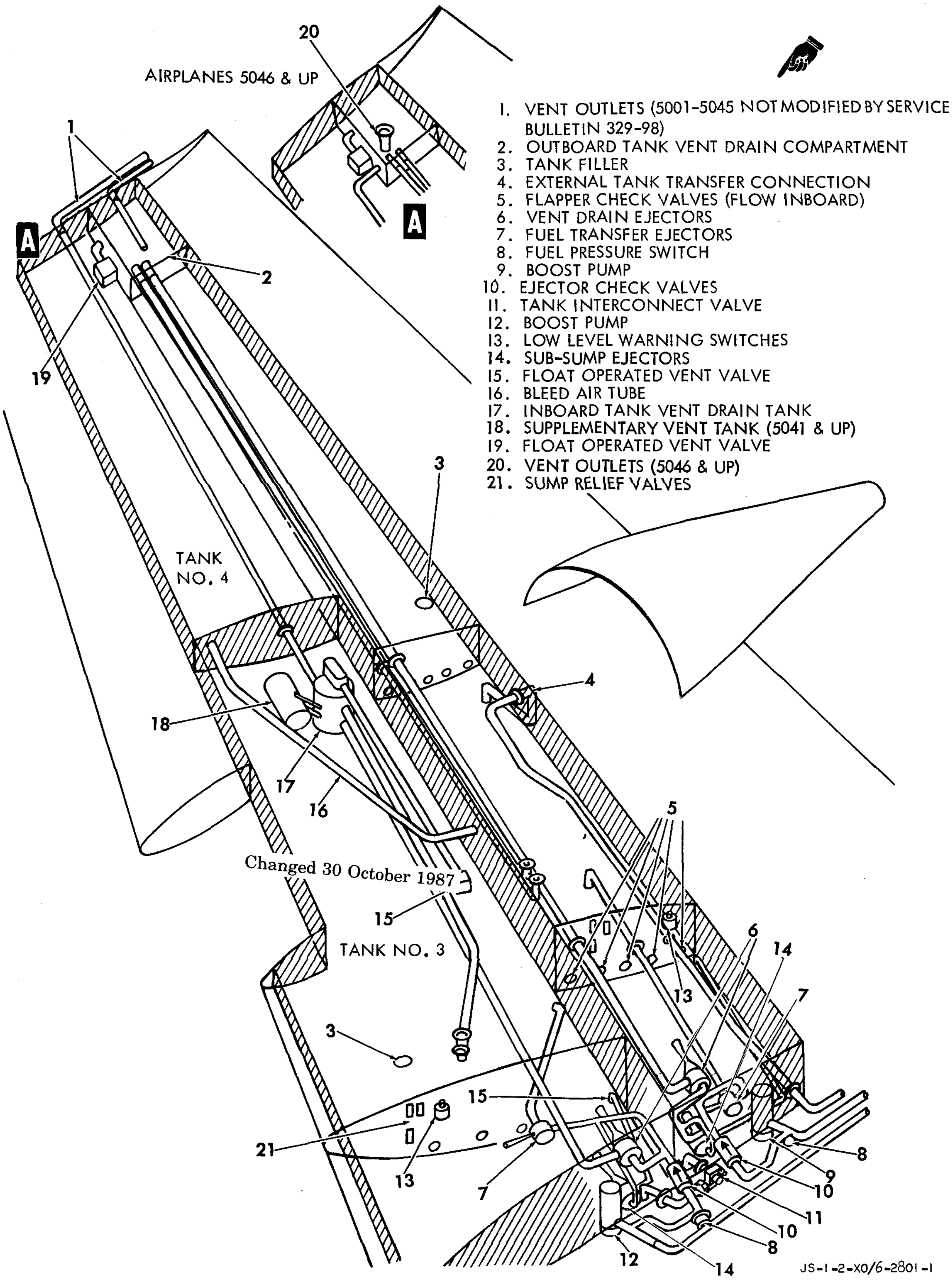

1. VENT OUTLETS (5001-5045 NOT MODIFIED BY SERVICE BULLETIN 329-98)
2. OUTBOARD TANK VENT DRAIN COMPARTMENT
3. TANK FILLER
4. EXTERNAL TANK TRANSFER CONNECTION
5. FLAPPER CHECK VALVES (FLOW INBOARD)
6. VENT DRAIN EJECTORS
7. FUEL TRANSFER EJECTORS
8. FUEL PRESSURE SWITCH
9. BOOST PUMP
10. EJECTOR CHECK VALVES
11. TANK INTERCONNECT VALVE
12. BOOST PUMP
13. LOW LEVEL WARNING SWITCHES
14. SUB-SUMP EJECTORS
15. FLOAT OPERATED VENT VALVE
16. BLEED AIR TUBE
17. INBOARD TANK VENT DRAIN TANK
18. SUPPLEMENTARY VENT TANK (5041 & UP)
19. FLOAT OPERATED VENT VALVE
20. VENT OUTLETS (5046 & UP)
21. SUMP RELIEF VALVES

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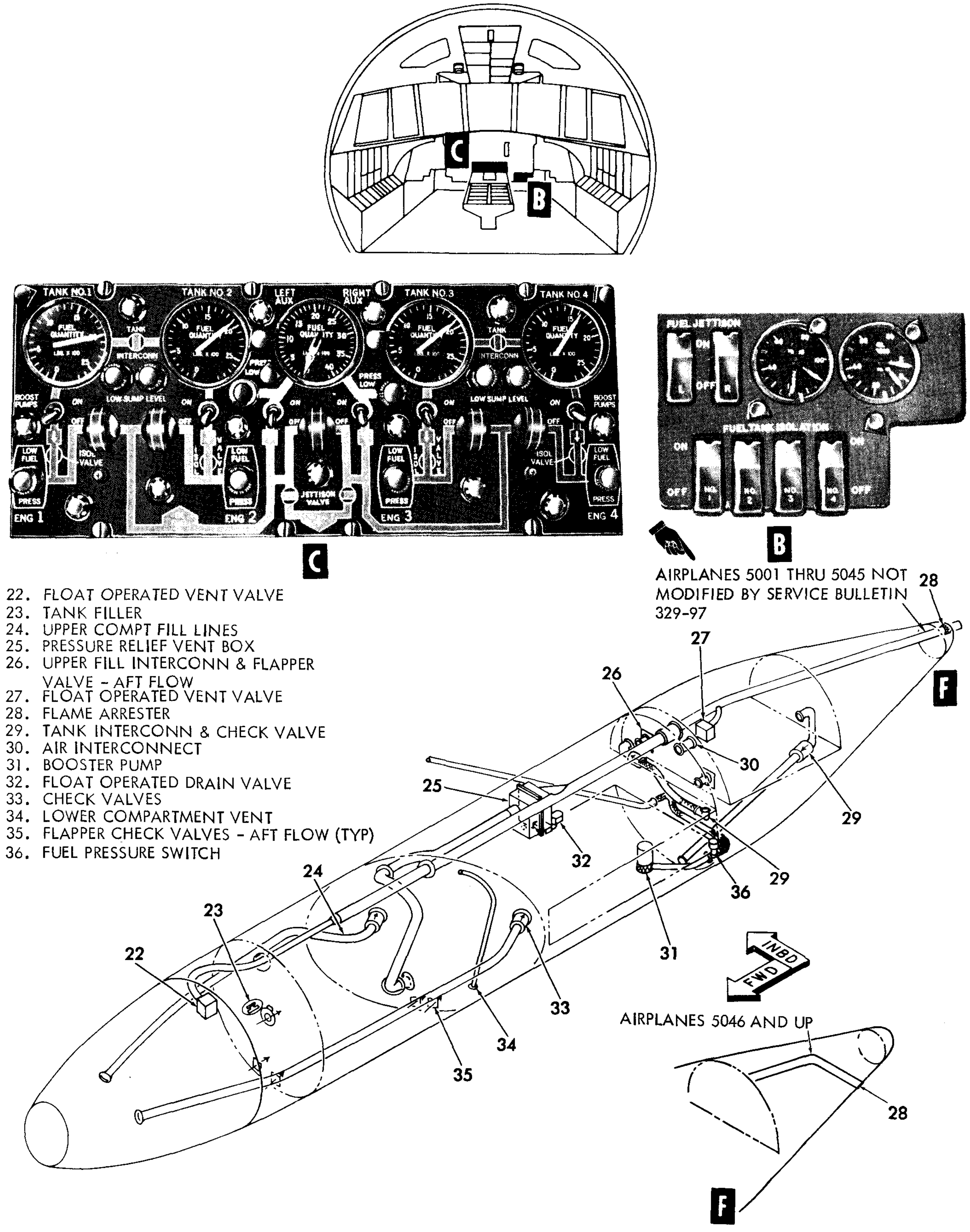
Figure 28-1. Fuel System Components Location (Sheet 1 of 3)



22. FLOAT OPERATED VENT VALVE
23. TANK FILLER
24. UPPER COMPT FILL LINES
25. PRESSURE RELIEF VENT BOX
26. UPPER FILL INTERCONN & FLAPPER VALVE - AFT FLOW
27. FLOAT OPERATED VENT VALVE
28. FLAME ARRESTER
29. TANK INTERCONN & CHECK VALVE
30. AIR INTERCONNECT
31. BOOSTER PUMP
32. FLOAT OPERATED DRAIN VALVE
33. CHECK VALVES
34. LOWER COMPARTMENT VENT
35. FLAPPER CHECK VALVES - AFT FLOW (TYP)
36. FUEL PRESSURE SWITCH

Figure 28-1. Fuel System Components Location (Sheet 2 of 3)

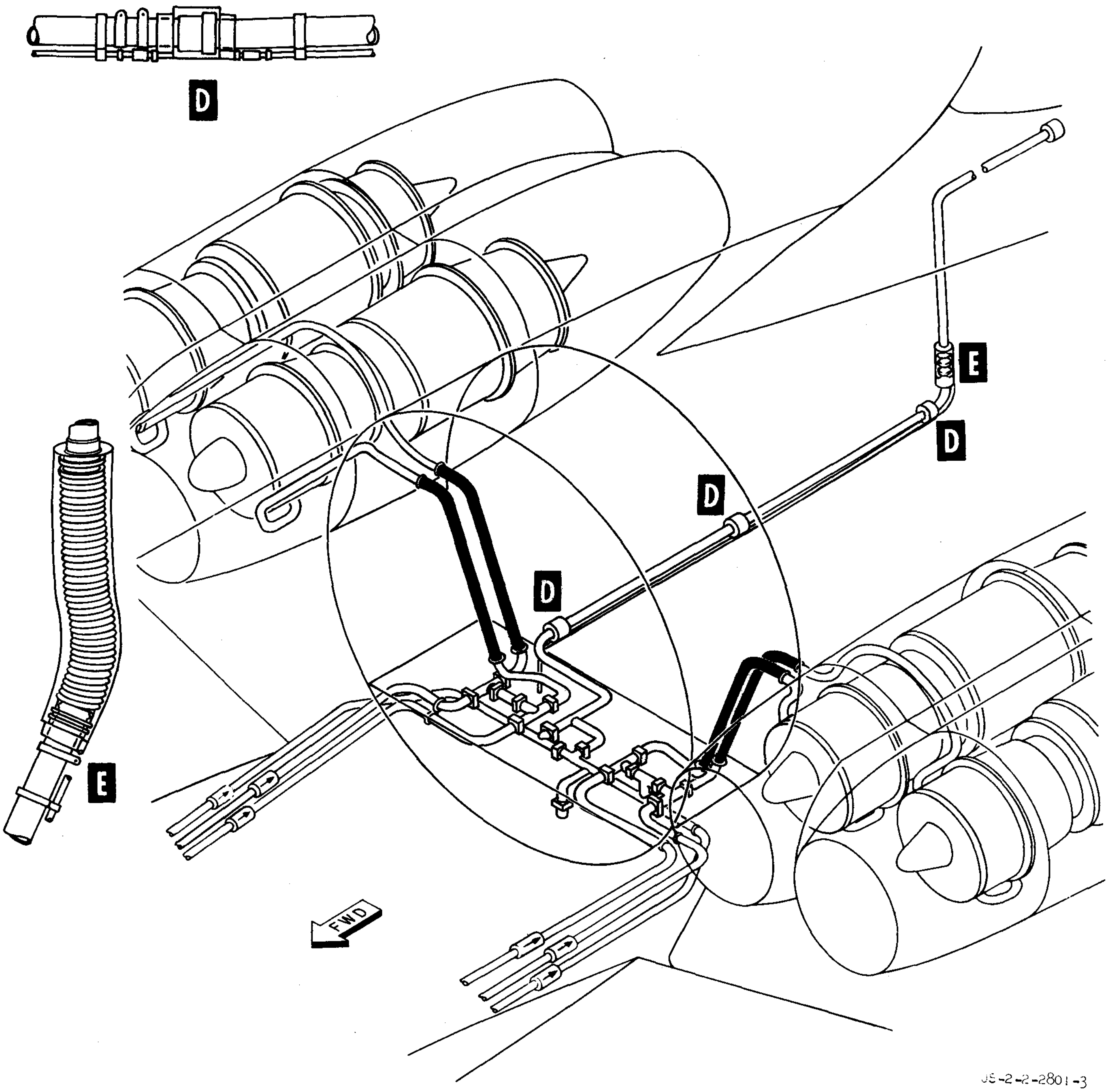
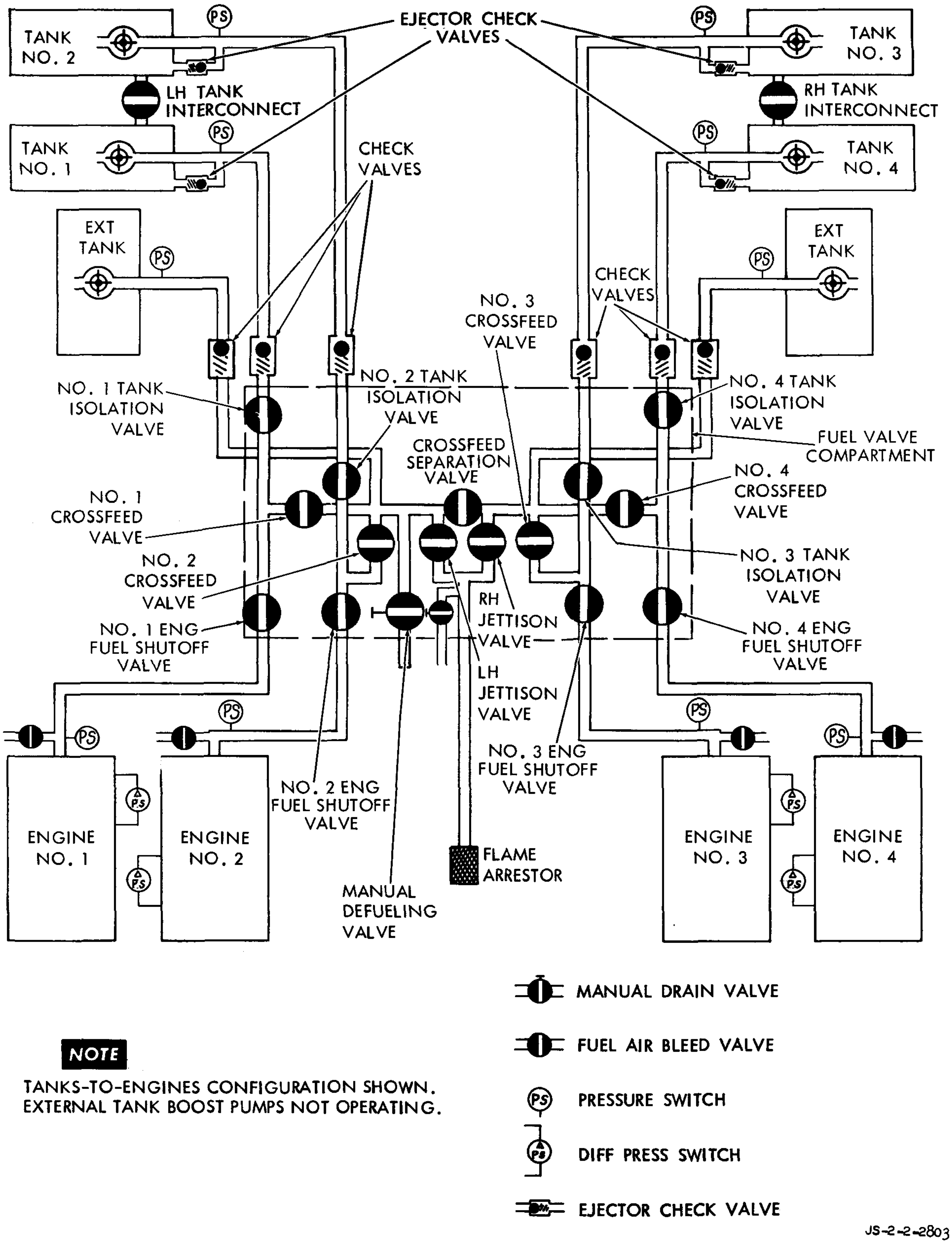


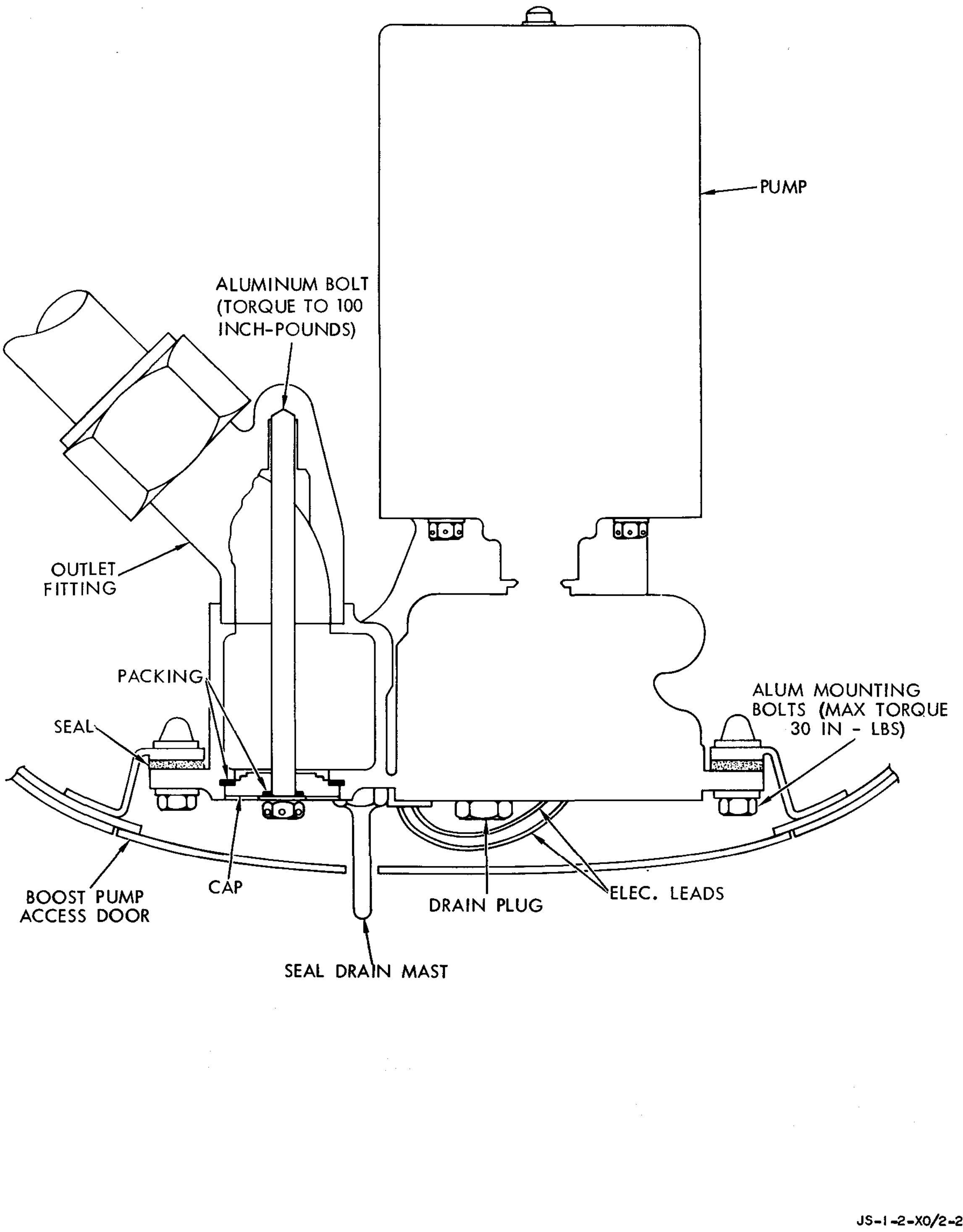
Figure 28-1. Fuel System Components Location (Sheet 3 of 3)



NOTE

TANKS-TO-ENGINES CONFIGURATION SHOWN.
EXTERNAL TANK BOOST PUMPS NOT OPERATING.

Figure 28-2. Fuel System Flow Diagram



JS-1-2-X0/2-2803

Figure 28-3. External Tank Pump Installation

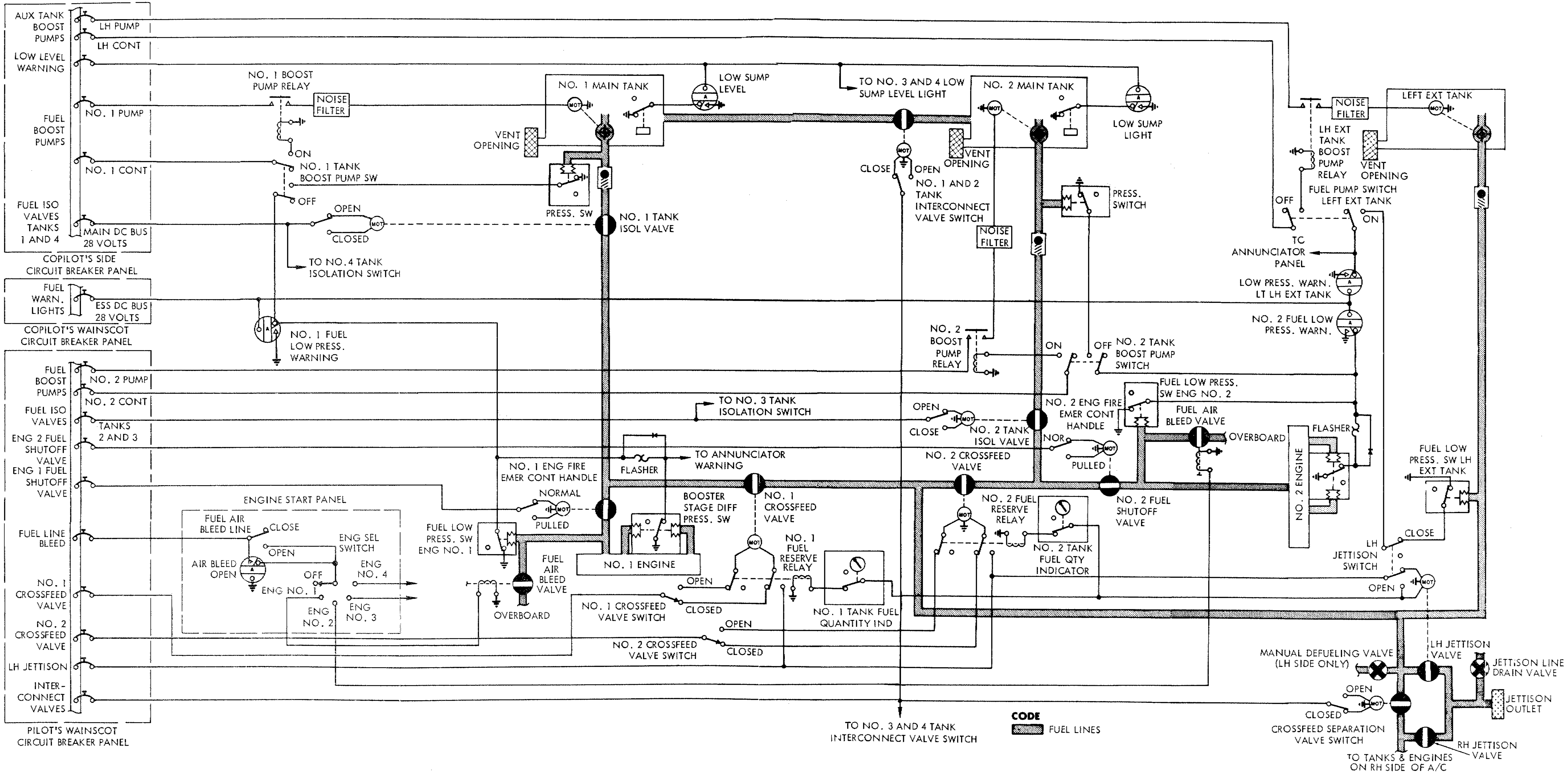


Figure 28-4. Fuel System Schematic Diagram

FLARED TUBE FITTING TORQUE VALUES (INCH-LBS)

SIZE		ALUM. TUBING	STEEL TUBING	AN 924 JAM NUTS *	AND 10050 PORTS *	NAS 596 (STEEL) RIGID CONNECTOR***
3/16"	(-3)	-	90-100	65-80	65-75	-
1/4"	(-4)-	40-65	135-150	85-105	95-105	60-96
5/16"	(-5)	60-80	180-200	105-125	125-135	66-108
3/8"	(-6)	75-125	270-300	120-150	155-165	72-120
1/2"	(-8)	150-250	450-500	240-280	280-305	144-232
5/8"	(-10)	200-350	650-700	320-380	380-405	204-360
3/4"	(-12)	300-500	900-1000	500-600	550-600	300-540
1"	(-16)	500-700	1200-1400	720-880	880-900	504-696
1 1/4"	(-20)	600-900	-	960-1200	900-1000	600-900
1 1/2"	(-24)	600-900**	-	1200-1440	900-1000	600-900

* FOR BOTH ALUM. & STEEL FITTINGS

** 1400-1600 INCH-LBS. FOR 0.049 WALL ALUM. TUBING

***USED IN FIRE EXTINGUISHER SYSTEM (REFER TO CHAPTER 26)

TORQUE VALUES FOR FLEXIBLE HOSE ASSEMBLIES GENERALLY ARE THE SAME AS THOSE SPECIFIED FOR THE SAME TYPE TUBE FITTING.

FOR TORQUE VALUES OF FLARELESS TUBE FITTINGS SEE FIGURE 29-3

JS-1-2-X0/2-2804A

Figure 28-4A. Flared Tube Fitting Torque Values

SECTION 28.1
FUEL VENT SYSTEM

Contents Page

DESCRIPTION AND OPERATION 28-17
  General 28-17
  Main Tank Venting 28-17
  Sump Venting 28-17
  Vent System Fuel Removal 28-17
  External Tank Venting 28-17

OPERATIONAL CHECKOUT 28-17

TROUBLESHOOTING 28-18

REMOVAL AND REPLACEMENT 28-18

FIGURES
  28-5. Fuel Vent System Components Locations 28-19

28.1.1 DESCRIPTION AND OPERATION.

28.1.1.1 GENERAL. The fuel vent system serves the following functions: (1) To prevent excessive pressure differentials between the inside of the tank and the aircraft ambient for all conditions of aircraft attitude, maneuver, altitude change, and fuel quantity. (2) To provide a method of catching any fuel that enters the vent system and returning it to the sump before it spills overboard. (3) To provide a method of removing large quantities of fuel from the tank to prevent serious structural damage in case of check valve and boost pump failure. (4) To provide a reservoir to assure sufficient fuel expansion space.

28.1.1.2 MAIN TANK VENTING. Two vent inlets are provided in each main tank. The inboard inlet has a double bellmouth which is open at all times. The outboard inlet is protected by a float valve, and is open whenever the inboard inlet is flooded and at low fuel levels. Vent inlet and line sizes have been selected to prevent any adverse pressure differentials. An air bleed line is provided to prevent a pocket of trapped air in one corner of tanks 1 and 4 when the aircraft is in the climb condition. Small holes are provided in the stiffeners of the wing upper surface to remove and permit the movement of air in the top of the tank at extremely high fuel levels.

28.1.1.3 SUMP VENTING. The main tank sumps normally have a fuel level higher than the tank level, and fuel is constantly being introduced into the sump by the fuel transfer ejectors. These factors result in a peculiar problem for venting in this area. Venting is normally accomplished by a series of small holes at the top of the sump rib. For climb, a small vent tube connecting the upper, forward corner of the sump with the upper, forward corner of the tank is provided. In addition to this, sump relief valves are provided to prevent a pressure build up in the sump due to the transfer of fuel by the ejectors or by check valve failure.

28.1.1.4 VENT SYSTEM FUEL REMOVAL. Any fuel that enters the main tank vent system will accumulate in the vent tank or vent drain compartment. The vent drain ejector, which operates anytime the tank boost pump is on, then removes the fuel at this point and returns it to the tank sump. The vent tanks and drain compartments also serve as reservoirs of expansion space for the main tanks.

28.1.1.5 EXTERNAL TANK VENTING. Venting of the external tank is accomplished through two float operated valves. One valve is located near the forward end of the external tank, and the other near the aft end. Both are located high in the tank. Normal attitudes of the aircraft result in the forward float valve being open under all conditions except dive, at which time the aft valve opens. At low fuel levels both valves are normally open. The underwing compartment has a small vent tube to vent this area to the top of the tank. No method of removing fuel entering the external tank vent system is provided as this, by tank design, is a rather remote possibility. A reverse pressure relief valve is provided to give a path of fuel overboard in case of check valve failure.

28.1.2 OPERATIONAL CHECKOUT. The vent system should be assumed to be operating satisfactorily if fuel system operation is normal and no fuel is being lost overboard from the vent outlets.

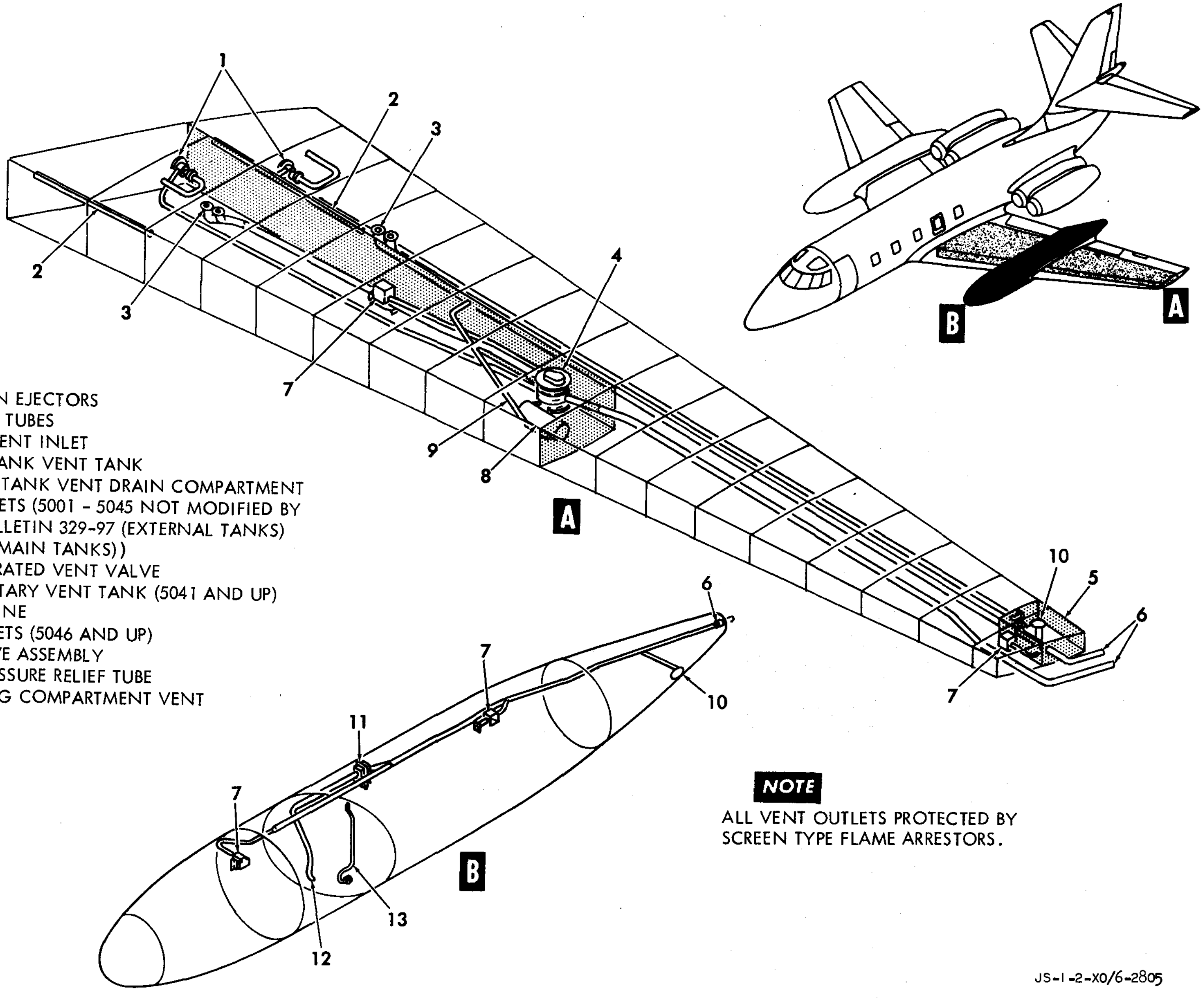
28.1.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FUEL SPILLING FROM MAIN TANK VENT OUTLET.		
Float valve leaking or stuck open.	If fuel is coming from tank No. 1 or 4, remove vent drain compartment cover and see if fuel is coming from (1-inch) float valve line. If coming from tank No. 2 or 3, remove float valve access cover and inspect valve.	Replace float valve.
Vent drain ejector check valve leaking or stuck open.	If fuel is coming from tank No. 1 or 4, see if fuel is entering vent drain compartment from (3/4-inch) vent drain line. If fuel is coming from tank No. 2 or 3, inspect ejector through boost pump access door.	Replace vent drain ejector.
Intertank leakage.	Make intertank leakage check per note after paragraph 28.2.3.3.	Replace or repair intertank seal.
Leak from tank into vent lines.	Determine location of leak.	Repair leak.
FUEL SPILLING FROM EXTERNAL TANK VENT OUTLET.		
Aft float valve leaking or stuck open.	Fuel flows from vent with full tank, during climb, rotation or acceleration.	Replace aft check valve.
Relief valve leaking.	Fuel flow from vent with full tank.	Repair or replace relief valve.
Forward float valve leaking or stuck open.	Fuel slugs from vent during "round-out" or climb immediately after descent or breaking.	Replace forward check valve.

28.1.4 REMOVAL AND REPLACEMENT. Access for removal and replacement of components of the fuel vent system in the main tanks, except for the vent drain ejectors, is by removing the wing tank cover panels in accordance with the instructions in section 28.2. The vent drain ejectors are accessible through the fuel boost pump access openings at the wing root rib. The access holes above the float valves in the main tanks are for inspection purposes only, and they are too small for float valve removal and replacement. Access to the vent system components in the external tank is through the access doors which are exposed when the tank is removed from the wing and, for the aft float valve only,

28.1.4 (Cont).

through an access door exposed when the wing trailing edge flaps are lowered. The actual removal and replacement procedures for the vent system components are straightforward, being in accordance with standard aircraft practices. Note that all bonding jumpers disturbed during maintenance operations must be reinstalled in their original locations with finish removed to assure good electrical contact. Also, flexible rubber fabric line couplings should be installed with a thin coat of MIL-S-8802, Class A-1/2, sealant (see figure 28-12) on the outside of the tube at the clamp position.



1. VENT DRAIN EJECTORS
2. SUMP VENT TUBES
3. INBOARD VENT INLET
4. INBOARD TANK VENT TANK
5. OUTBOARD TANK VENT DRAIN COMPARTMENT
6. VENT OUTLETS (5001 - 5045 NOT MODIFIED BY SERVICE BULLETIN 329-97 (EXTERNAL TANKS) OR 329-98 (MAIN TANKS))
7. FLOAT OPERATED VENT VALVE
8. SUPPLEMENTARY VENT TANK (5041 AND UP)
9. AIR BLEED LINE
10. VENT OUTLETS (5046 AND UP)
11. RELIEF VALVE ASSEMBLY
12. REVERSE PRESSURE RELIEF TUBE
13. UNDER WING COMPARTMENT VENT

NOTE

ALL VENT OUTLETS PROTECTED BY SCREEN TYPE FLAME ARRESTORS.

Figure 28-5. Fuel Vent System Components Locations

SECTION 28.2
FUEL TANKS

Contents

Page

DESCRIPTION AND OPERATION 28-21
- General 28-21
- Integral Fuel Tanks 28-21
- External Fuel Tanks 28-21
- Safety Precautions 28-22

REMOVAL AND INSTALLATION 28-22
- General 28-22
- Removal of Integral Fuel Tank Top Access Panels 28-22
- Alternate Method for Access To Wing Tanks 28-23
- Installation of Integral Fuel Tank Top Access Panels 28-24
- Removal of External Fuel Tank 28-25
- Installation of External Fuel Tank 28-26
- Replacement of Condensate Drain Valve O-Ring 28-26

LEAK DETECTION AND REPAIR 28-26
- General 28-26
- Types of Leaks and Recommended Action 28-27
- Leak Detection 28-28
- Fuel Tank Sealant Application and Leak Repair 28-29
- Temporary External Repairs for Fuel Tank Leakage 28-32
- Fuel Soak and Stand Pipe Test for External Leakage 28-32
- Tank to Tank Leakage 28-32

PRESERVATION 28-32
- Alternate Preservation Method 28-33
- Preservation of Open Fuel Tanks 28-33

FIGURES

28-6. Wing Support for Tank Cover Removal 28-34
28-7. Integral Fuel Tank Upper Access Panel Permissible Deflection 28-36
28-8. Location of Fuel Tank Mechanical Seals 28-37
28-9. Upper Wing Panel Attachments 28-38
28-10. Drilled Screw Method of Tank Leak Detection 28-42
28-11. Typical Sealant Application 28-43
28-12. Sealing Materials 28-44
28-12A. Fuel Tank Drain Valve (Airplanes 5046 and up) 28-44B

28.2.1 DESCRIPTION AND OPERATION.

28.2.1.1 GENERAL. All of the fuel is held in four integral wing tanks and two external, wing-mounted tanks.

28.2.1.2 INTEGRAL FUEL TANKS. (See figure 28-6.) Each wing is divided into two main integral tanks. (Refer to Section 12.1 for information on tank capacities.) A sump is formed at the inboard end of each tank by a rib which is completely sealed except for a vent at the top. Flapper check valves installed in the bottom of the rib permit one-way fuel flow into the sump. Pressure relief valves are installed on sump ribs to prevent over-pressurization of the sump. A fuel boost pump, a fuel transfer ejector, a sub-sump ejector, a vent drain ejector, a low fuel level warning switch, a drain valve, and associated plumbing lines are installed in each sump. A baffle is installed approximately at midspan of the aft tank to control surging. Poppet type water drain valves are mounted on the bottom wing panel at the lowest point in each sump and the outer wing compartment. Each tank has a flush-type filler cap in the upper wing surface. On airplanes 5001, 5002, 5004 thru 5040, and all airplanes that have not incorporated Service Bulletin No. 329-70, a float-actuated rod prevents installation of the filler cap if the tank is overfilled, thus providing for the correct expansion space. On airplanes 5003, 5041 and up, and airplanes incorporating Service Bulletin 329-70, a new tank design prevents the possibility of filling the expansion space.

28.2.1.3 EXTERNAL FUEL TANKS. (Refer to chapter 12 for information on tank capacity.) The wings divide each tank into two compartments. The fuel in the upper compartment flows down to the lower compartment through an interconnecting line. Baffles

28.2.1.3 (Cont).

are installed in the compartment to prevent surging of fuel during flight. Check valves on the baffle next to the boost pump prevent the fuel from flowing away from the pump. Fuel is pumped from the external tanks to the crossfeed manifold by the boost pumps. Each tank is filled through a flush-type filler opening in the top of the tank, located so the expansion space cannot be filled. Water drain valves are installed in the bottom of the tank to permit drainage of condensed moisture and sediment.

28.2.1.4 SAFETY PRECAUTIONS. Observe all of the safety precautions contained in Chapter 12 when performing fuel tank maintenance, in addition to observing the following special precautions:

a. Electrically ground the airplane at both the forward and aft static ground connections.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

b. Special procedures must be followed to support the airplane before removing the large access panels over the integral wing tanks. Refer to the wing tank access panel removal procedures in this section for instructions.

c. Use only explosion proof flashlights, blowers, and other equipment around aircraft which have contained fuel. Remove electric power from aircraft when personnel enter fuel tanks. Keep one person outside and supply forced ventilation while any person is working inside tank. If fuel vapors are present, air supplied masks will be used.

d. Exercise care to ensure that no damage occurs to sealant inside tanks. Use only DDD-R-30 clean cotton rags or cellulose sponges for cleaning. Worker outer clothing should not be made from wool, silk, dacron, or nylon. Use wrapping paper or plastic film instead of cloth for protective covering.

WARNING

Observe the following LEL (lower explosive limit) vapor concentration levels during fuel tank maintenance work. Above 20% LEL is not considered "firesafe." Purge tanks with clean air down to 20% LEL before opening. Between 20% & 5% LEL enter tanks only with full face, air supplied respirators. Between 5% & 1.5% LEL use half face, vapor chemical cartridge respirators. Below 1.5% LEL plus 19.5% oxygen is considered "healthsafe".

28.2.2 REMOVAL AND INSTALLATION.

28.2.2.1 GENERAL. The two large access panels on top of the wing provide access to the integral fuel tanks and carry part of the wing load. It is necessary to support the airplane in a fixed position before removing either of these panels to relieve the load on the panel attaching screws, and to prevent damage to the structure while the panel is removed. Special equipment is necessary for this purpose (see figure 28-6). In addition, the external fuel tanks must be removed to reach all the fasteners in the wing panels.

28.2.2.2 REMOVAL OF INTEGRAL FUEL TANK TOP ACCESS PANELS. To avoid structural distortion of the airplane while the wing tank cover panels are removed, and to facilitate alignment of the attaching screw holes when the panels are reinstalled, a special method of supporting the airplane during these operations must be used. Fuselage support at station 490 is required, and either additional support at station 608, or addition of weight to the nose section of the fuselage to stabilize it in the fore and aft direction, is also required.

a. Adapt two 10-ton-capacity aircraft tripod jacks to carry the locally-made cradles shown in detail A, figure 28-6. When these cradles are mounted on the jacks, the bottom of each cradle channel should be approximately 47-1/2 inches from the floor with the jack at midtravel. The jacks should be provided with conventional safety locks to prevent inadvertent lowering under load. The cradles should be mounted on the jacks in such a manner that they cannot slide or become disengaged from the jacks under side loads.

b. If it is desired to use an additional support at fuselage station 608, locally manufacture a tripod-based fuselage cradle as shown in detail "B", figure 28-6. The contour and bevel of the cradle should conform to those of the fuselage at station 608, and the load-carrying capacity of the assembly should be approximately 5 tons.

c. Defuel all four wing tanks and both external tanks in accordance with the defueling procedures in Chapter 12.

d. Jack the airplane at the fuselage nose and the wing jack points in accordance with the procedure in Chapter 7. Maintain the airplane in a level attitude.

e. Remove the external tank from the wing containing the affected integral tank in accordance with the instructions in Paragraph 28.2.2.5.

f. Remove fairings as necessary to expose the lower member of the fuselage frame at station 490.

28.2.2.2 (Cont).

g. Position the two fuselage frame cradles and jacks at fuselage station 490, with each cradle centered approximately 32 inches from the centerline of the airplane, so that the horizontal portion at each end of the lower frame member is engaged by each padded channel. On the right-hand side of the fuselage, it may be necessary to loosen a hydraulic tubing clamp to enable the cradle to be positioned correctly. Raise each cradle with its jack to place its channel in contact with the frame member, but do not raise the cradles sufficiently to remove any load from the wing jacks. Lock the cradle jacks in this position.

h. If the aft fuselage cradle is to be used, position it at fuselage station 608, and raise it to place the cradle approximately 1/2 inch below the fuselage skin in the frame area. If the aft fuselage cradle is not to be used, proceed with step (j).

CAUTION

Be careful to position the cradle, if used, at the frame location, and not against unsupported skin panels, or the skin could be damaged when load is applied to the cradle.

i. Secure the aft fuselage cradle to prevent it slipping aft by lashing it in position with tiedown ropes or equivalent between the top of the cradle tripod and each main landing gear structure.

j. If desired, instead of using the cradle at fuselage station 608, weight may be added to the fuselage in the nose landing gear area to maintain the center of gravity location well forward of the fuselage station 490 cradles.

28.2.2.2 (Cont).

k. Recheck the fuselage station 490 cradles and the nose jack for security and proper contact with the airplane.

l. Maintaining the airplane level, lower it until firm contact is made with the fuselage station 490 cradles. Monitor the fuselage station 608 cradle, if used, and lower it as required to prevent contact with the fuselage.

m. Lower the wing jacks until the fuselage load is stabilized and carried by the fuselage station 490 cradles and the nose jack. The wing jacks should carry no load.

n. Raise the fuselage station 608 cradle, if used, until it is in firm contact with the fuselage. Position the four wing cradles in accordance with the locations shown in figure 28-6, and raise them into firm contact with the wing at these locations.

o. Lower the wing jacks enough to disengage them from the jack pads. They may then be removed from beneath the airplane, if desired.

CAUTION

The weight of the airplane is now carried only by the nose jack and the fuselage station 490 cradles. Avoid unnecessary movement of personnel in the airplane, as the resultant weight shifts could affect its stability on these supports.

Note

If access is required only to integral tank areas in the outboard end of the wing, it is permissible to remove only certain outboard fasteners of the panels and to deflect the panels upwards in accordance with the limitations shown in figure 28-7. Refer to paragraph 28.2.2.3 for detailed instructions. If removal of the complete panels is required, accomplish all the following steps.

p. Remove the top wing to fuselage fairing and the attaching bolts which secure the affected panel to the fuselage. Remove the fuel quantity indicating system probes per Section 28.3.

q. Refer to figure 28-9 and remove all the screws around the edge and across the width of the panel, including the two screws outboard of ribs at box beam stations 70.6 and 109.4, and remove the panel. Due to the forward panel covering part of the aft tank, remove both panels if access to the aft tank is required.

CAUTION

Do not disturb any of the airplane supporting provisions while any panels are removed. Before removing the panels, make sure all the screws and wing upper access panel to fuselage attaching bolts are removed. Remove all access covers and filler caps in the panels, and use the holes as handholds to lift the panels. Work from the outboard end. Do not use pry bars or screwdrivers between the panels and the wing structure, or the spar caps maybe seriously damaged.

r. Place the removed panels on a clean table, with the inside of the panel upward. Place a covering of wrapping paper or similar material over the panels to exclude dirt, dust, and other foreign material. Do not cover the panels with cloth material as lint may filter down on to the panel.

28.2.2.3 ALTERNATE METHOD FOR ACCESS TO WING TANKS. Complete removal of the integral fuel tank top access panels is not always necessary, when the area within the tank to which access is required is not adjacent to the wing root. In these circumstances, it is permissible to remove only some of the panel attaching screws, and to deflect the panel upwards by raising the outboard tip. This upward deflection of the panels is limited by the possibility of excessive bending causing structural damage to the integral stiffeners or to the edges. The limits of upward deflection of the panel tip versus the distances from the panel tip to the furthest inboard screws remaining installed are shown in figure 28-7.

CAUTION

If the required access cannot be obtained within the limits of tip deflection and screw removal, the complete panel must be removed in accordance with the instructions in paragraph 28.2.2.2. If the deflection limits are inadvertently exceeded, inspect the panel carefully in the areas of the last remaining installed screws to see if any permanent deformation has occurred in the edges of the panel and/or across the integral stiffeners of the panel just outboard of these screws. Also remove these screws to see if the screw threads, the countersunk holes in the panel, or the plate nuts have been damaged. If no damage has been found in any of these places, the panel may be reinstalled. Otherwise, if the panel itself is damaged it must be replaced with a new panel, or if the plate nuts or screws are damaged they must be replaced as necessary.

28.2.2.3 (Cont).

Proceed as follows for partial lifting of the upper access panels:

a. Support the airplane as directed in steps (a) through (d) and (f) through (o), paragraph 28.2.2.2.

b. Calculate from the graph in figure 28-7 how far inboard from the tip of the panel the screws must be removed to gain access to the desired area.

c. If the calculation shows that the last loosened screws are to be inboard of the external fuel tank, or in the external fuel tank area, remove the external tank in accordance with the instructions in this section.

d. Remove the panel attaching screws, starting at the tip and working inboard. Note that the first screws to be left installed, at the inboard end of the panel section to be raised, should all be located in the same box beam.

CAUTION

This will be the area of transition between the raised section of the panel and the installed section. Do not allow one row of screws to carry all the bending load. Four or more screws should be progressively tighter, going towards the wing root, so as to distribute the bending load and thus to avoid a sharp bend in the panel and excessive loading of individual screws.

e. Support the panel outboard tip in the area within 6 inches of the outboard end, and raise the tip a distance not to exceed the limits shown in the graph in figure 28-7.

f. When the tip has been raised to the desired level, secure the means of support so that the tip cannot be inadvertently dropped while work is being performed on the wing tanks. Install safety blocks between the raised panel and the wing, as required. The blocks must rest on the wing, but as required above, the support should be within 6 inches of the tip and the safety blocks must not contact the underside of the panel.

28.2.2.4 INSTALLATION OF INTEGRAL FUEL TANK TOP ACCESS PANELS. Thin strips of rubber seal are installed around critical areas of the wing to form a compression seal when the panels are installed (See figure 28-8.) A seal is located around each of the removable upper wing surfaces and access doors as follows:

Forward root rib access door seals
Aft root rib access door seals
Forward upper wing surface seals
Aft upper wing surface seals
Sump bulkhead seals
Dry bay seals
Inter-tank bulkhead seals.

Always make sure these seals are clean and in place before reinstalling the access panels. Replace any seals that have cuts, tears, or rough spots. Each time the upper wing surfaces are removed, inspect the inter-tank bulkhead seals and replace any that have a permanent set of 0.20 inch or more below the seal retainer. In some instances, sealing compound has been applied at specific points around the edge of the access panel to supplement the sealing action of the mechanical seals. This old sealant must be removed and new sealant applied before the panel is replaced. Use the following procedure to install a wing access panel:

a. Mark the area around the edge of the panel which contained the sealing compound so this area can be identified later.

b. Scrape all the old sealant from the edge of the wing access panel and from the mating surface on the wing with a micarta scraper.

c. After all sealant is removed, clean the area with K-U-L (Kelite), Stoddard High Flash Solvent (P-S-661), or aliphatic naphtha. Naphtha is the preferred fluid, but is more volatile and should be used cautiously. Apply the fluid with clean cellulose sponges only, and wipe dry immediately with a clean, dry cellulose sponge.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

WARNING

Exercise extreme care when using Stoddard High Flash Solvent and aliphatic naphtha. These solvents are highly flammable.

CAUTION

Do not use plastic sponges such as rayon or nylon since they quickly develop static charges of electricity when rubbed and may ignite the cleaner.

28.2.2.4 (Cont).

d. Inspect the interior of all dome nuts for the presence of liquid sealant and if present, remove the sealant. Inspect all dome nuts for damage and security of attachment to structure.

Note

If all sealant is not removed from the dome nuts, the panel installation screws may bottom on the sealant and ruin the dome nuts. Any damaged dome nuts should be replaced.

e. Apply a coating of Johnson Paste Wax or Simoniz Car Wax (Paste) to the rubber seals installed on the access panel ribs and on the top surface of the wing along the edges where the sealant was removed.

f. Mix a batch of MIL-S-8802, Class B sealing compound according to instructions contained in this section.

g. Apply a thin coating of Johnson Paste Wax or Simonize Car Wax (Paste) as a parting agent and then a thin coating of the Class B sealing compound to the wing structure immediately adjacent to the seal groove that

28.2.2.4 (Cont).

will mate with the access panel edge where the old sealant was removed.

Note

Apply the sealant in a thin bead to avoid the dome nuts.

h. Position the access panel in place on the wing while the sealant is still wet, and install all the panel mounting screws. (See figure 28-9.)

CAUTION

Make sure the proper screws are used to secure the panel. If possible, use the same screws in the same locations from which they were removed. If new screws are used, make sure they are of the proper length. Screws too long will not permit the panel to seal and may damage the dome nuts. Special nonmagnetic screws are used in the outboard tips near the compass flux valves (See figure 28-9.) These screws must be removed and installed with nonmagnetic tools (refer to Section 34.6).

Seal the screws attaching the panel to the wing ribs by applying a small amount of the Class B compound under the head of the screw before it is tightened.

i. Wipe off all excess sealant immediately after all screws are tightened.

j. Install the wing upper access panel to fuselage attaching bolts and fairing.

k. Install the fuel quantity indicating system probes per Section 28.3.

l. Position the wing jacks, previously removed, and raise them to engage the jack pads. Raise the airplane with the wing and nose jacks, in a level attitude, enough to remove the load from the cradles. Make sure the three jacks are now bearing the weight of the airplane.

m. Lower the cradles clear of the airplane frame and surfaces. Remove the tiedown ropes from the aft fuselage cradle and main landing gear. Move all the cradles clear of the airplane. Remove the supplementary weights from the nose landing gear area of the fuselage, if these were used instead of the aft fuselage cradle. Tighten the hydraulic tubing clamp at station 490, right-hand side, as necessary. Install the fairings previously removed. Install the external tank in accordance with the instructions in this section.

n. Make sure the main and nose landing gear ground safety locks are installed.

o. Lower the airplane to the ground and remove the jacks.

p. Fuel the airplane, perform an operational check-out of the fuel system, and check the tanks for leakage.

28.2.2.5 REMOVAL OF EXTERNAL FUEL TANK.

Note

For removal and installation purposes, the external tank may be considered as three major sections: the forward section, the aft section, and the bottom fairing. The forward section extends on the top and bottom of the wing to the rear beam where it joins with the aft section. The bottom fairing moves with and is attached to the flap.

a. Observe all of the safety precautions contained in Chapter 12, paragraph 12.1.1 and paragraph 28.2.1.4 of this chapter.

b. Drain the tank of all fuel.

c. Remove all tank fairings.

d. Disconnect the tank-to-airplane plumbing and wiring.

e. On the top of the wing, disconnect the plumbing between the forward and aft sections of the tank, and remove the fittings from the forward section.

f. On the bottom of wing, disconnect the fuel interconnect line which connects the forward and aft sections of the tank.

g. On top of the wing, disconnect the two close-tolerance bolts which connect the aft section of the tank to the forward section. Remove the aft section.

h. Support the forward section of tank.

i. Remove the fitting on the aft end of the forward section from the tank and from the top of the wing rear beam.

j. Remove the fittings on the aft end of the forward section from the tank and from the bottom of the wing rear beam.

k. Remove the bolts from the fitting assembly on each side of the forward section which secure the tank to the front beam, and remove the fittings.

28.2.2.5 (Cont).

l. Disconnect the two close-tolerance bolts which connect the forward section of the tank to the front beam.

m. Remove the forward section.

28.2.2.6 INSTALLATION OF EXTERNAL FUEL TANK. Tank installation is the reverse of the removal.

Note

> The fuel line which connects the forward and aft sections of the tank under the wing is a structural member as well as a fuel interconnection. Do not alter the length of this line, or tank alignment will be affected.
>
> When the two close-tolerance bolts, removed in 28.2.2.5.f, are reinstalled be sure that the O.D. of the washer used on the threaded end measures 0.68 inch or larger (LS 1785-416 or equal). Aircraft 5001 through 5052 were delivered with washers having an O.D. of 0.50 inch. These washers should be replaced with the larger O.D. washers at time of next tank removal and replacement.

28.2.2.7 REPLACEMENT OF CONDENSATE DRAIN VALVE O-RING. On airplanes 5001 thru 5045, the O-ring seals on the fuel tank drain valves can be replaced by the following method:

a. Defuel the tank in which the valve is located.

b. Lock the drain valve in the open position by pushing up and rotating the valve stem.

c. After all fuel is drained from the tank, insert a preformed wire hook through the drain hole, hook it over the outer edge of the O-ring, and pull the seal from its seat. Remove the seal through the drain hole.

Note

> An adequate tool for removing the O-ring can be fashioned from lockwire. Sharpen a piece of wire to a point, and bend the wire to the shape of a small fishhook.

d. Insert the replacement O-ring through the drain hole, and place it in its approximate position with a blunt tool such as a screwdriver.

e. Unlock the valve stem, and let it snap to the closed position. This should force the O-ring into its retaining groove on the valve stem.

f. Operate the valve until visual inspection assures that the seal is properly positioned and seated.

Note

> On airplanes 5046 and up, and on previous airplanes on which valves have been replaced with the preferred spare part, a new condensate drain valve allows O-ring replacement without defueling the tank. A secondary seal within the valve allows removal of the valve seat to facilitate O-ring seal replacement. (See figure 28-12A).

28.2.3 LEAK DETECTION AND REPAIR.

28.2.3.1 GENERAL. Both integral and external fuel tanks are sealed on the inside with a sealing compound which cures into a stiff, rubbery material. Great care is exercised in the original application of the sealing compound to assure correct installation. This sealant will have a long, trouble-free service life, if maintained properly. When it is necessary to break this seal, or if a leak should develop, open the tank and repair the sealant from the inside. External repairs are permitted only under the conditions specified in this section for such temporary repairs.

a. The original seal on airplanes 5001 thru 5045 is applied to the tanks by performing the following in the order listed:

1. Cleaning	- All surfaces are thoroughly cleaned to assure proper adhesion of the sealant.
2. Sealing of mating faying surfaces (during assembly sealing).	- All faying surfaces of both external and integral tanks receive an application of sealing compound before they are joined.
3. Fillet and brush sealing (post-assembly sealing).	- The entire fuel tank is sealed by the application of fillets of sealing compound to all seams and joints.
4. Coating of fasteners	- Two thin coats of brush-type sealant are brushed over all fasteners passing through the tank barrier, and over all fillets.
5. Application of corrosion protective coating	- The bottoms of integral tanks are coated with a corrosion protective compound to protect them from the corrosive action of water.

28.2.3.1 (Cont).

6. Application of sealant protective coating - The entire tank is filled with a liquid protective agent and then drained. This provides a thin protective coat over the entire tank, to protect the sealant from attack by fuel and to provide additional protection against corrosion of the tank structure.
7. Curing - After each application of sealant and protective coating material, each is allowed to cure for a specified time at a specified temperature.
8. Testing - The tank is pressure and vacuum tested for leaks. If none is found, it is filled with fuel and soaked for an additional test period before it is accepted as leakfree.

b. The original seal on airplanes 5046 and up is applied to the tanks by performing the following in the order listed:

1. Cleaning - All surfaces are thoroughly cleaned to assure proper adhesion of the sealant.
2. Sealing of mating faying surfaces (during assembly sealing) - All faying surfaces of both external and integral tanks receive an application of sealing compound before they are joined.
3. Application of corrosion protective coat - The entire tanks are coated with a corrosion protective compound to protect them from the corrosive action of water.
4. Fillet and brush sealing (post-assembly sealing) - The entire fuel tank is sealed by the application of fillets of sealing compound to all seams and joints.
5. Coating of fasteners - Two thin coats of brush-type sealant are brushed over all fasteners passing through the tank barrier and over all fillets.
6. Application of final corrosion protective coating - A second coat of the corrosion protective compound is applied to the lower surfaces of all integral tanks in order to insure complete coverage and coating continuity.
7. Curing - After each application of sealant and protective coating material, each is allowed to cure for a specified time at a specified temperature.
8. Testing - The tank is pressure and vacuum tested for leaks. If none is found, it is filled with fuel and soaked for an additional test period before it is accepted as leakfree.

28.2.3.2 TYPES OF LEAKS AND RECOMMENDED ACTION. Fuel leaks are classified into four types, which reflect the rate of leakage. For an accurate measure of leakage, wipe clean the area surrounding the leak, and apply talcum powder to this area. After observing the leak source for 30 minutes, measure the wet area, and classify the leak by the following definitions:

a. Slow Seep - A fuel leak that wets an area around the source up to 3/4 inch in diameter.

28.2.3.2 (Cont).

b. Seep - A fuel leak that wets an area up to 1-1/2 inches in diameter around the leak source, but does not flow, run, or drip.

c. Heavy Seep - A fuel leak that wets an area up to 3 inches in diameter, but does not flow, run, or drip.

d. Running Leak - A leak that allows fuel to drip or run from the structure.

28.2.3.2 (Cont).

Fuel leaks are further described by stating their location in relation to open or closed airplane areas. This is important, since leaks in some areas require immediate repair, while leaks in other areas are not considered to be potential flight hazards. The following table indicates the recommended action for the various types of leaks in relation to their location:

28.2.3.2 (Cont).

CAUTION

Leaks may be caused by structural failure and not by sealant discrepancies. Do not attempt to repair such leaks by adding more sealant.

FUEL LEAK LOCATION	TYPE OF LEAK	RECOMMENDED ACTION
Upper or lower skin surface, exposed area of external tank, or wing leading edge inboard of tank.	Slow seep, seep, or heavy seep.	Clean surface, record, and inspect frequently.
	Running leak.	Ground airplane, and repair immediately.
Rear beam and wing tip.	Slow seep or seep.	Return to home field, and repair before further flight.
	Heavy seep or running leak.	Ground airplane, and repair immediately.
Area under external tank, front beam, or root rib.	Slow seep, seep, heavy seep, or running leak.	Ground airplane, and repair immediately.

28.2.3.3 LEAK DETECTION. The source of a leak, which may be several inches from its external indication, must be found before it can be repaired. Familiarity with the structure of the wing will minimize the time required to locate a leak source. Four methods of leak detection are explained below. These methods are the fuel-level method, the air-bubble method, the fluid-blowback method, and the drilled screw method. The fuel level method will only determine the approximate location; however, any of the other methods will locate the exact source of the leak on the inside of the tank.

Note

If tank-to-tank leakage is suspected, it can be checked by draining tanks No. 1 and No. 4 and filling tanks No. 2 and No. 3. Any tank-to-tank leakage will then flow from the drains of tank No. 1 or No. 4.

a. APPROXIMATE LOCATION BY THE FUEL-LEVEL METHOD. The following method will determine the approximate location, and is particularly useful for locating leaks in the forward and aft beams and the inboard and outboard tank bulkheads. In most cases, the leak source will

28.2.3.3 (Cont).

be found in the approximate location of the detected leak.

1. If the tank is empty, fill the tank in 50 or 100-gallon increments until the tank begins to leak. The beginning of the leakage indicates the fuel level is just above the hole in the sealant. If the tank is full, drain it in 50 or 100-gallon increments and the leak source will be just above the point where the tank stops leaking.
2. Determine the tank fuel level by means of a dipstick.
3. Approximate the fuel level from the outside of the tank, using the dipstick as an indicator.
4. Use one of the methods described in the following paragraphs to accurately determine the location of the leak.

b. AIR-BUBBLE METHOD.

1. Defuel and purge the leaking tank. If the tank involved is one of the integral wing tanks,

28.2.3.3 (Cont).

remove the access panel in the top surface of the wing over the tank.

CAUTION

See instructions in this section for special support of the airplane prior to panel removal.

Note

Due to the forward tank access panel covering part of the aft tank, it will be necessary to drain both the forward and aft tanks if the forward panel is being removed.

2. Cover the outer tank structure at the area of the leak with a suitable leak detector fluid such as American Gas and Chemical Company, Oxy-Tec Type I, MIL-L-25567, or equivalent.

CAUTION

Do not apply the leak detector fluid inside the tank.

3. Blow over all seams, bolts, rivets, and fillets in the area of the most probable leak source with air not exceeding 100 PSI. Use an air hose without a connection on the open end to blow all possible leak sources from the inside of the tank. The hose must be held snugly against the sealant area being tested. In some cases it may be helpful to form a special nozzle of soft copper tubing as required for testing complicated tank structural points.

CAUTION

Air pressure is to be applied from inside the tank.

4. Check until the leak is located as indicated by the bubbling of the leak detector on the outside of the tank.

c. FLUID BLOWBACK METHOD. The fluid blowback method is similar to the air bubble method, except that isopropyl alcohol is blown from the outside of the tank at the suspected leak point.

1. Defuel the tank, and remove the access panel in the top surface of the wing.

CAUTION

See instructions in this section for special support of the airplane prior to panel removal.

2. Apply the alcohol from the outside leak point at 10-PSI pressure. A pressure pot, regulating valve, and gage arrangement may be used to apply the alcohol. The leak source is located when the alcohol emerges inside the tank. After the source is located, blow all alcohol out of the tank interior.

3. A variation of the above method for detecting leak sources in the lower surface of the wing consists of adding water to the tank bottom. Zinc chromate putty, such as Presstite No. 594, may be used to form a dam in order to contain the water around the area in question. Clean air is forced through the leak exit point as above, from the outside. and with a maximum air pressure of 10-PSI.

d. DRILLED SCREW METHOD. (See figure 28-10.) The following method is effective for leaks particularly difficult to locate, but it should not be used until all other methods have proved unsatisfactory:

1. Prepare a screw of an appropriate size along with the other equipment shown in the figure.

2. Defuel the tank, remove the wing access panel, and remove the rivet or bolt closest to the suspected leak point.

CAUTION

See instructions in this section for special support of the airplane prior to panel removal.

3. Install the drilled screw, gaskets, washers, and nut as shown in the figure.

4. Blow alcohol, at a maximum pressure of 10 PSI, into the screw to determine the leak source. Check the interior of the tank for the point where the alcohol emerges.

5. After the source of the leak is located, blow air through the screw to evaporate any alcohol between the structures and at the leak source. The leak can be repaired only after the alcohol has been removed.

6. Reinstall the fastener removed for the test.

28.2.3.4 FUEL TANK SEALANT APPLICATION AND LEAK REPAIR. Follow the repair directions in this section closely so a positive seal will be obtained. If the sealant is installed in a haphazard manner, the seal will leak and it will be necessary to repeat the operation. Any hours saved in attempting to perform a short-cut sealing operation will be more than used up in the necessary rework.

28.2.3.4 (Cont).

Note

On airplanes 5001 through 5045 the inside surface of all fuel tanks is coated with Buna-N (MIL-S-4383). This was changed, starting with airplane 5046 and airplanes in compliance with Service Bulletin 329-168, to a polyurethane sealant topcoat (MIL-C-83019).

The basic sealant used in the tank is MIL-S-8802. In recent years a corrosion inhibited sealant, MIL-S-81733, has been developed which provides better protection in fay surface joints, with wet installed fasteners, and in confined areas. MIL-S-8802 continues to be used for fillets, edge seals, and exposed fasteners. MIL-S-81733 should be used with new parts for fay surfaces and in injection holes, but MIL-S-8802 is used for leak and sealant repairs for parts in place. The two sealing materials are very compatible and their use together or, in an emergency, one in place of the other will cause little or no difficulty. MIL-S-8802 has better fuel resistance and, for this reason, it is best to use this on all surfaces that are exposed to fuel. The MIL-C-83019 topcoat is then applied over all new sealant to prevent degradation by the fuel.

A polyurethane (MIL-C-27725) coating is applied to the inside of wing panels and to other aluminum parts used inside the tank prior to sealing starting with airplane 5046. Because of inferior intercoat adhesion after fuel aging, repairs to damaged or deteriorated MIL-C-27725 are not made with this same material. Here a coating of MIL-S-81733 sealant is used in the exposed area with a MIL-C-83019 topcoat because more weight is given to the corrosion protection provided than to the inferior fuel resistance. Special attention should be given to assure a good topcoat when this is done.

It should be noted that MIL-C-27725 is used only to coat parts before installation. It is never applied to surfaces inside the tank after installation.

All applications of sealant should be preceded with the application of an adhesion promoter as indicated. The use of a tack free time accelerator is not recommended. Cleanliness cannot be over-emphasized.

a. Observe all of the safety precautions contained in Section 12.1 and paragraph 28.2.1.4 of this chapter.

b. CLEANING. After the source of the fuel leak is determined, clean away the old sealant before the new repair material is applied. The surfaces must be thoroughly cleaned before applying any new sealant. Oil and dirt from hands, hair, or other foreign substances will prevent the sealant from adhering to the metal. Use the following procedure for cleaning:

1. Scrape the surface around the leak with a micarta, wood, or phenolic scraper to remove the old sealant. Taper the edges of the fillet seals back from the cleaned area at about a 45-degree angle.

CAUTION

Do not use metal or plexiglass scrapers. Use only tools fabricated from micarta, wood, or phenolic materials.

2. Remove the corrosion protective coating of Buna-N or MIL-C-83019, as necessary, so reapplication of sealant will not be made over the old coating. Buna-N may be removed with cloths dampened with TT-M-268 Methylisobutylketone (MIBK). The softened Buna-N must be scraped off with a phenolic tool. Remove residual Buna-N with clean rags dampened with MIBK and wipe dry. MIL-C-83019 topcoat is removed by scraping and abrading with care not to damage any tank structure. Never apply sealant on Buna-N or MIL-C-83019 topcoat. Sealant must be applied only to adhesion promoter covered, solvent cleaned bare or MIL-C-27725 coated metal.

WARNING

Exercise extreme care; some of the following solvents are flammable. Provide adequate ventilation. Avoid skin contact and prolonged breathing of vapors.

3. Thoroughly clean the area to be sealed using one of the following solvents:

   (a) Aliphatic naphtha TT-N-95
   (b) Trichloroethane O-T-620
   (Alt. Degreaser MIL-T-81533)
   (c) Methylethylketone (MEK) TT-M-00261
   (d) Methylisobutylkeytone (MIBK) TT-M-268
   (e) Dry cleaning solvent P-D-680, Type II
   (f) High flash Stoddard solvent P-S-661

Apply the fluid with clean cellulose sponges and wipe dry immediately with clean, dry cellulose sponge. On airplanes 5046 and up and those airplanes that have Service Bulletin 329-168 incorporated, areas to be sealed after thorough

28.2.3.4 (Cont).

cleaning shall be coated with an adhesion promoter (See figure 28-12). Allow promoter to dry for 1/2 hour prior to application of sealing compound.

Note

The adhesion promoter is supplied in an aerosol can with an adjustable spray nozzle for spray application or it may be supplied in pint containers with a screw cap. Care shall be taken not to contaminate the material and to keep the container tightly closed when not actually withdrawing material from the bottle. Once the container is opened, it shall be used within 7 days. Any material remaining in a container, with screw type cap, longer than 7 days after opening shall be discarded. Any material that appears cloudy shall be discarded. Always keep the container tightly capped while not removing material.

CAUTION

Do not use plastic sponges or rayon or nylon cloths since they quickly develop static charges of electricity when rubbed and may ignite the vapor.

c. MIXING SEALANTS. Two different types of sealant materials are recommended for repairing leaks: for fillet repair, MIL-S-8802, Class B is recommended; for fastener repair, MIL-S-8802, Class A is recommended. Both types come with different application lives and cure times (See figure 28-12) and are procured in two parts: a base and an accelerator. For service repairs, this sealant should be procured in 1/2-pint kits. The kits are supplied in correct proportions of base and accelerator. Mix the two parts in the base can. Uses the following procedures when mixing the sealant:

Note

Do not mix the sealant at temperatures below 65°F or above 90°F. The sealant may be heated or cooled prior to mixing by placing the container in hot or cold water.

1. Mix the accelerator thoroughly to a smooth paste before adding it to the base. Discard the entire batch if it is lumpy or caked and cannot be made into a smooth paste by stirring.
2. Mix the entire kit regardless of how little sealant will be required. All the accelerator must be scraped into the can of base compound, and the base compound remaining on the lid also must be scraped into the mixture. Adding accelerator gradually during mixing aids in complete mixing of the ingredients.
3. Stir the mixture slowly by hand using a flat mixing paddle or a medical tongue depressor until a completely uniform color, free from marbled appearance, results. Stir slowly for at least 4 minutes. Stirring the mixture rapidly will introduce air bubbles which might later burst and cause pinhole leaks in the tank. Rapid stirring of the mixture also may affect the set time due to increased heat. Remove the paddle at lest once during mixing, and scrape the sealant from it back into the container. Do not try to thin the compound, and do not mix different manufacturers' compounds or different batches of the same manufacturer's compound.

d. SEALANT APPLICATION. Consult figure 28-12 to select the material to be used, and then mix in accordance with the instructions above. Apply the sealant in the order listed below in accordance with the instructions given for each type of seal:

1. Make all injection seals. All voids in the tank structure should be sealed by injection sealing of sealant with a pressure sealing gun. MIL-S-8802, Class B sealant should be used for leak and sealant repairs. MIL-S-81733, Type II sealant should be used for fay sealing of newly installed parts. Such voids normally occur at corners and under joggles and overlapping structural members. Since the old sealant may not be completely removed from the voids, some will not take any sealant. Injections should be made through the normal openings in the structure and through the No. 40 injection holes located in doublers, fittings and in other parts covering voids. (See figure 28-11.)

2. Coat surfaces to receive fillet seals with an adhesion promoter. (See figure 28-12.)Apply a light coat of adhesion promoter and allow to dry for a minimum of 30 minutes.

3. Replace fillet seals with MIL-S-8802, Class B sealant. Apply the fillet with a pressure sealing gun. Hold the gun approximately perpendicular to the edge being sealed to force the compound into place and eliminate air pockets. A bead of sealant should precede the tip as shown in figure 28-11. All fillets should be continuous, without any voids or breaks. Fair fillets smoothly into adjoining fillets. After the fillets have been applied, smooth them out with a filleting spatula or with a brush which may be lubricated with Class A, Brush-Type sealant. Use a forced smearing motion to press the fillets into place. Work out any evidence of air bubbles. All fillets should conform generally to the original dimensions. When the edge thickness is over 0.64 inches, it may be necessary to use two applications of compound to obtain the desired dimensions. If the second application is necessary, it should be made before the first has cured. If the first has cured, cleaning is required prior to the second application. Dress the first fillet with a brush or spatula before the second application.

28.2.3.4 (Cont).

4. Coat surfaces to receive brush coats or fillets with an adhesion promoter. (See figure 28-12.) Apply a light coat of adhesion promoter and allow to dry for a minimum of 15 minutes.

5. Two brush coats of sealing compound are required over all fasteners, such as rivets, nuts, and bolts passing through the tank barrier. The first coat should be tack-free before the second coat is applied. The sealing compound should completely surround and cover the fasteners as shown in figure 28-11. The sealant must be free of pinholes at the critical area designated in figure 28-11. Fillets may be dressed by the application of a thin brush coat within one hour after the application of the fillet.

e. Allow sealant to cure until tack-free. This condition exists when the sealant will not stick to fingers or a sheet of polyethylene held against it. See figure 28-12 for the approximate time required for the sealant to cure to a tack-free condition. The cure may be accelerated by applying heated filtered air to the tank. Apply heat not hotter than 140 degrees Fahrenheit until the sealant is tack-free. On all airplanes sealant must be overcoated with MIL-C-83019 sealant topcoat to protect the sealant from degradation by the fuel. Apply the coating to produce a coating thickness of approximately 0.001 to 0.003 inches. The coating should overlap adjacent coated surfaces by 1/2 inch maximum. The coating will cure to a tack-free state in approximately 2 to 3 hours. Tanks may be refueled after coating reaches a tack-free state.

f. Tanks having the Buna-N system and not modified per Service Bulletin 329-168 should have damaged Buna-N repaired with MIL-C-83019. Clean the damaged area with clean cotton DDD-R-30 rags and approved solvents; and apply MIL-C-83019 sealant topcoat. Tanks may be refueled after the coating reaches a tack-free state.

g. On airplanes 5046 and up these surfaces have been coated with MIL-C-27725 polyurethane, plus on airplanes 5104 and up the lower surface of the tank has been coated with a green biocidal dye, to protect the milled skins from corrosion. Damaged, loose or blistered coating should be removed mechanically and repaired as indicated in Paragraph 51.7.3.1.

h. Install the upper access panels in accordance with the separate instructions in this section. The tanks may be fueled immediately thereafter.

28.2.3.5 TEMPORARY EXTERNAL REPAIRS FOR FUEL TANK LEAKAGE. The following information specifies the materials and procedures for the application of external patches as a possible method of stopping fuel tank leakage instead of removing wing panels to effect an internal seal repair:

a. Leak Detection - Locate the exact point of the leak, and determine the size of the external patch needed to repair it (maximum patch size: 3-inch diameter or equivalent area).

b. **Preparation of Wing - Defuel as necessary to remove fuel from behind the leak source.**

c. **Fabrication of Patch - Fabricate a patch from .003- to .016-inch thick 5052 or equivalent soft aluminum foil, size as determined in step (a).**

d. **Mixing of Sealant - Mixing of sealant should be done with extreme care. The accelerator and base compound should be pre-mixed in their respective containers. The base compound should then be thoroughly mixed with the accelerator. The resultant mixture must be of uniform color and consistency. Extreme precautions must be taken to avoid trapping air.**

e. **Cleaning - Thoroughly clean patch and area to which patch is to be applied by wiping with clean cotton rags moistened with methylethylketone (MEK). Wipe surfaces dry; do not allow solvent to air dry. Cleaning must remove all loose paints, grease and residual fuel. Surfaces must be clean and dry before sealant application.**

f. **Application of Patch - Apply approximately 1/32-inch film of Specification MIL-S-8802 Class B-1/2 sealant to both the cleaned patch and the cleaned surface of the wing. Apply the aluminum foil patch to the repair area. Masking tape should be applied to the area surrounding the patch to prevent sealant adhesion to this area. Then cover the entire area with a thin sheet of polyethylene.** Press firmly and smooth out the patch with fingers or roller. Remove the polyethylene sheet and the masking tape; clean up the surface with a plastic tool and a cotton rag dampened lightly with MEK. The excess sealant squeezed out at the edges of the patch must be removed while sealant is still uncured. Use masking tape to hold the patch in place until the sealant has cured.

g. Curing - Allow the sealant to cure approximately 16 hours at 75° F and 50 percent relative humidity before refueling tank. Cure may be accelerated by use of heat, 140° F maximum. Accelerated cure may be accomplished in 3 to 4 hours at 130° to 140° F.

h. Remove the masking tape and touch up the paint as required.

28.2.3.5 (Cont).

Note

The above procedure is only a temporary fix. All external fuel leak patches are to be removed and repair made inside the fuel tank at the next major overhaul or removal of wing upper surface panels.

Note

Semco Model 400A-1 Field Repair Kit contains all materials and tools to make temporary fuel tank repairs. Information on this is available from Semco Division, Products Research Corp; 2919 Empire Ave., Burbank, CA 91504

28.2.3.6 FUEL SOAK AND STAND PIPE TEST FOR EXTERNAL LEAKAGE. See Section 28.5.

28.2.3.7 TANK TO TANK LEAKAGE. See Section 28.5.

28.2.4 PRESERVATION. When the airplane is inactive for a prolonged period, the sealant in the fuel tanks and the rubber seals throughout the fuel system will deteriorate if not given the proper treatment. Observe the precautions in steps (a) through (c) to prevent this deterioration. If the period of inactivity will be 45 days or less, the procedure in paragraph 28.2.4.1 may be followed as an alternate procedure.

a. Fill all of the fuel tanks completely with fuel, allowing only about 2 inches of space from the tank upper surface for fuel expansion on hot days, and fill the fuel system completely with fuel (with the exception of the fuel jettison line which should be empty). This will protect the fuel tank sealant and the fuel system rubber seals indefinitely. Check the fuel tanks condensate drain valves and drain condensate as necessary every 10 days.

b. If the fuel system is drained, it should be refilled within 3 days to prevent damage to the rubber seals.

c. If the fuel tanks are drained, they should be refilled within 14 days to prevent damage to the sealant.

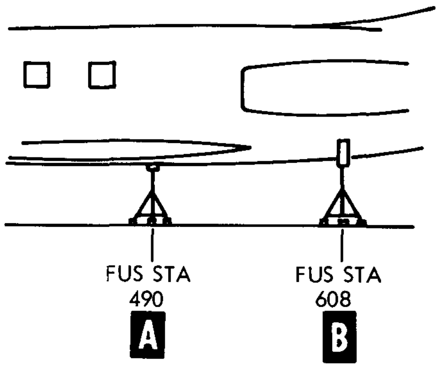
28.2.4.1 ALTERNATE PRESERVATION METHOD. The following alternate method is recommended on (1) tanks to be inactive 45 days or less or (2) thoroughly cleaned tanks to be inactive 6 months or less.

a. Maintain at least 75 pounds of fuel in each external fuel tank and at leat 1,500 pounds of fuel uniformly distributed in the integral fuel tanks. The vapor from this fuel will prevent deterioration of the tank sealant.

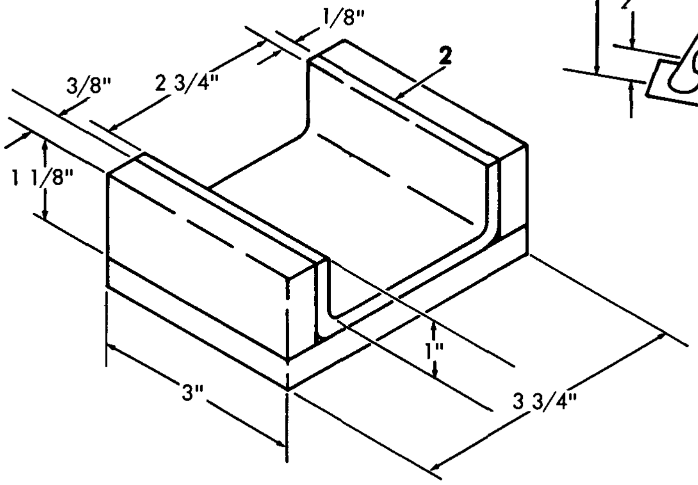
b. Maintain fuel in all parts of the fuel system except the fuel jettison line. This will preserve the rubber seals.

c. Check fuel tank condensate drain valves every 10 days and drain any water that appears. If bacteria or fungus (slime) is detected, the tanks should be opened, cleaned thoroughly, and inspected to insure the contamination is completely removed. Close the tanks and refuel according to quantities in step (a) above.

28.2.4.2 PRESERVATION OF OPEN FUEL TANKS. When tanks are open for an extended period of time (refer to paragraph 28.2.2), the tank interior is to be coated with MIL-L-6081, Grade 1010 oil in order that the sealant will not dry out. Open tanks, tank covers, removed access doors, etc., are to be covered with wrapping paper or plastic film to prevent contamination and ensure cleanliness. Pumps, valves, and similar equipment should be removed, oiled, placed in protective bags, and stored in a protected area. The inside tank surface will be checked periodically and cleaned and/or reoiled as required to maintain a clean, oily tank interior.



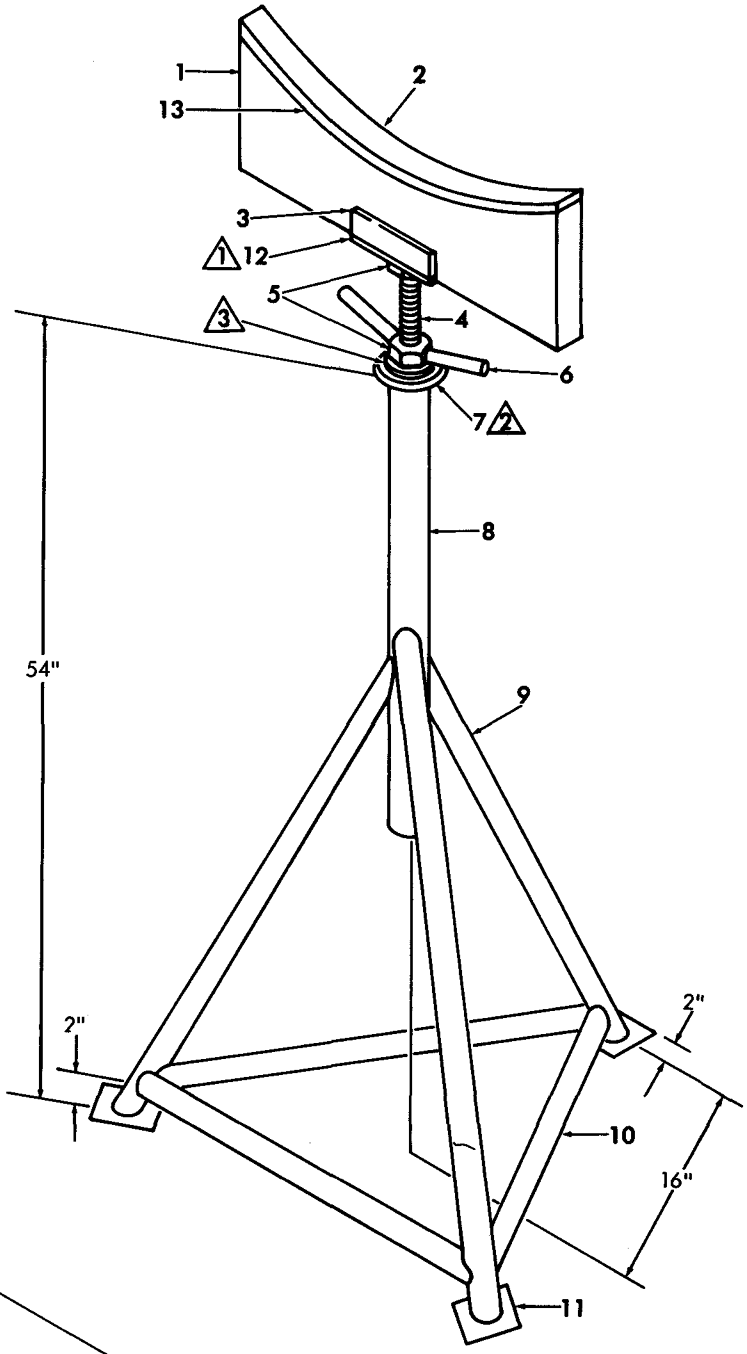
INDEX NO.	MATERIAL	QTY.
1	PLYWOOD 3/4 X 9 X 26 INCHES	3
2	1/8 RUBBER	1
3	STEEL PLATE 1/4 X 2 X 6	2
4	STEEL BAR THD 1 - 8	1
5	STEEL HEX. NUT 1 - 8	2
6	STEEL BAR 1/2 X 4	2
7	STEEL BAR 3 1/2 DIA. X 1	1
8	STEEL PIPE 2 I.D. X 31	1
9	STEEL PIPE 1 1/2 I.D. X 37	3
10	STEEL PIPE 1 1/2 I.D. X 26 1/2	3
11	STEEL PLATE 1/4 X 3 X 3	3
12	STEEL PLATE 1/4 X 2 3/4 X 6	1
13	STEEL PLATE 1/4 X 2 3/8 X 29	1



FUS. STA. 490 CRADLE

A

MAKE FROM 3/8 STEEL PLATE.
CEMENT RUBBER PAD TO CRADLE.
(LOCKHEED PART NO. 2600-U-MEJ-102)



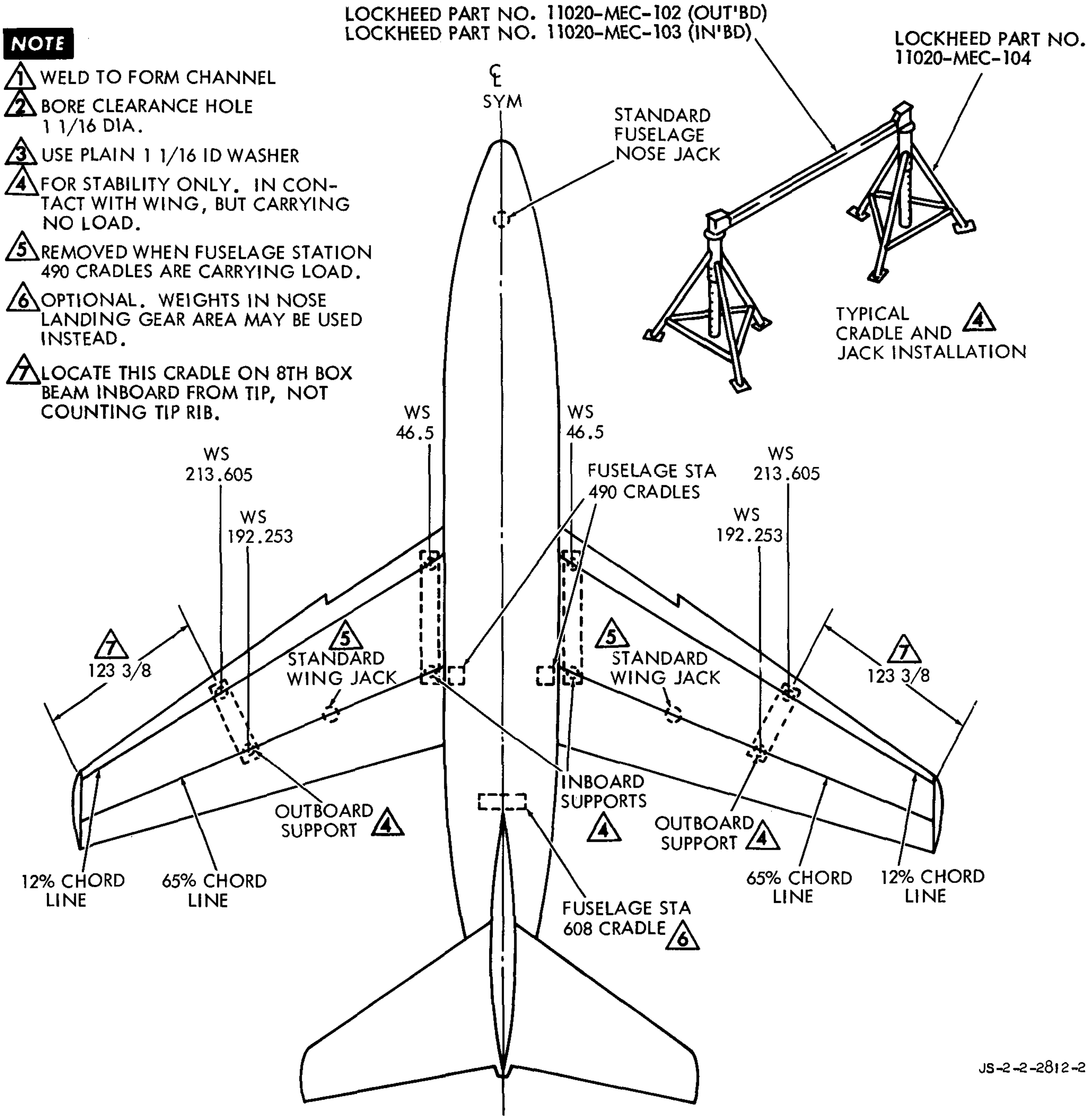
FUS. STA. 608 CRADLE

B

MAKE FROM MATERIALS LISTED.
WELD IN ACCORDANCE WITH
STANDARD SHOP PRACTICES.
(LOCKHEED PART NO. 1000-MEC-2)

JS-2-2-2812-1

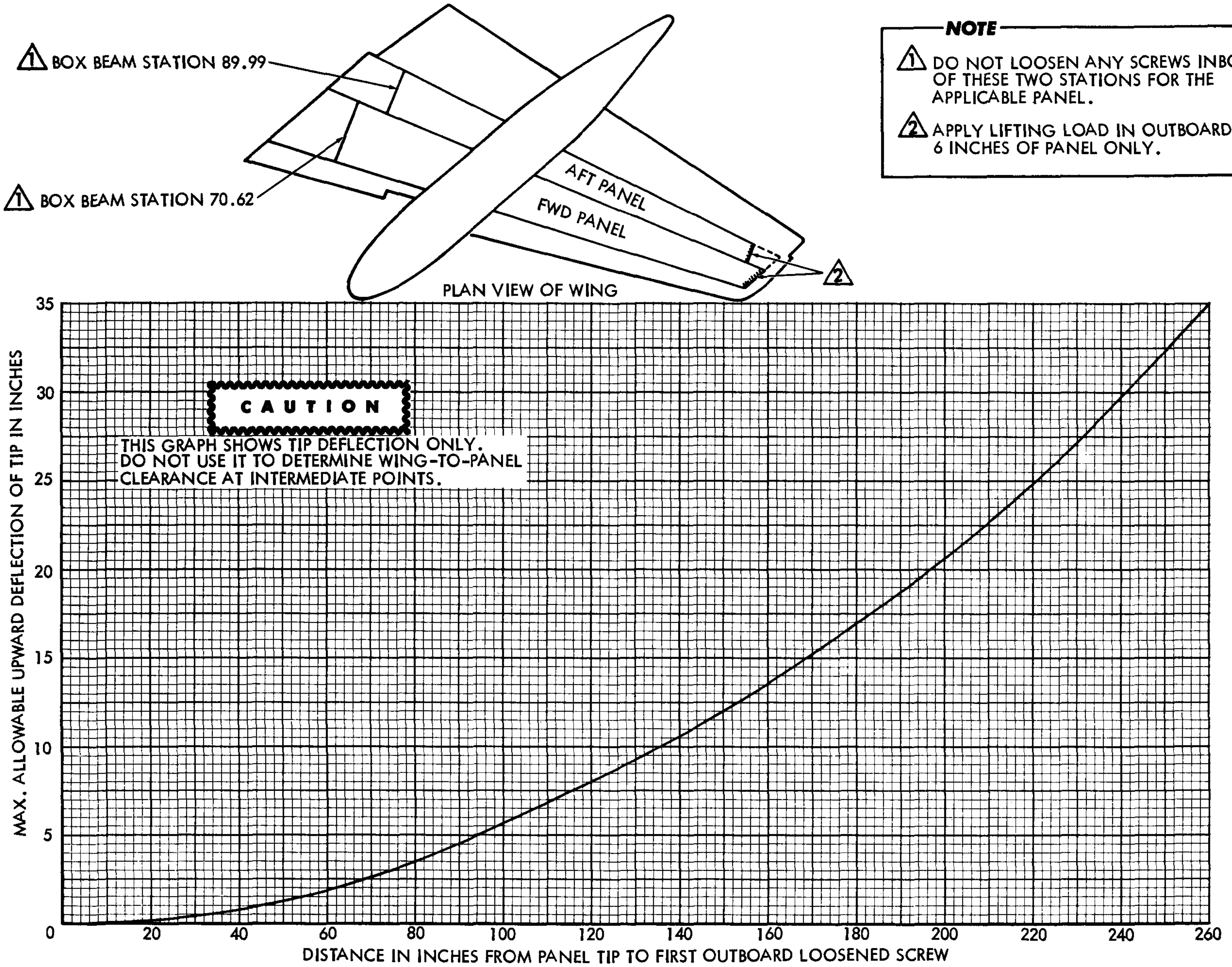
Figure 28-6. Wing Support For Tank Cover Removal (Sheet 1 of 2)



NOTE

1 WELD TO FORM CHANNEL
2 BORE CLEARANCE HOLE 1 1/16 DIA.
3 USE PLAIN 1 1/16 ID WASHER
4 FOR STABILITY ONLY. IN CONTACT WITH WING, BUT CARRYING NO LOAD.
5 REMOVED WHEN FUSELAGE STATION 490 CRADLES ARE CARRYING LOAD.
6 OPTIONAL. WEIGHTS IN NOSE LANDING GEAR AREA MAY BE USED INSTEAD.
7 LOCATE THIS CRADLE ON 8TH BOX BEAM INBOARD FROM TIP, NOT COUNTING TIP RIB.

Figure 28-6. Wing Support For Tank Cover Removal (Sheet 2 of 2)



NOTE

1 DO NOT LOOSEN ANY SCREWS INBOARD OF THESE TWO STATIONS FOR THE APPLICABLE PANEL.

2 APPLY LIFTING LOAD IN OUTBOARD 6 INCHES OF PANEL ONLY.

CAUTION

THIS GRAPH SHOWS TIP DEFLECTION ONLY. DO NOT USE IT TO DETERMINE WING-TO-PANEL CLEARANCE AT INTERMEDIATE POINTS.

JS-2-2-2813

Figure 28-7. Integral Fuel Tank Upper Access Panel Permissible Deflection

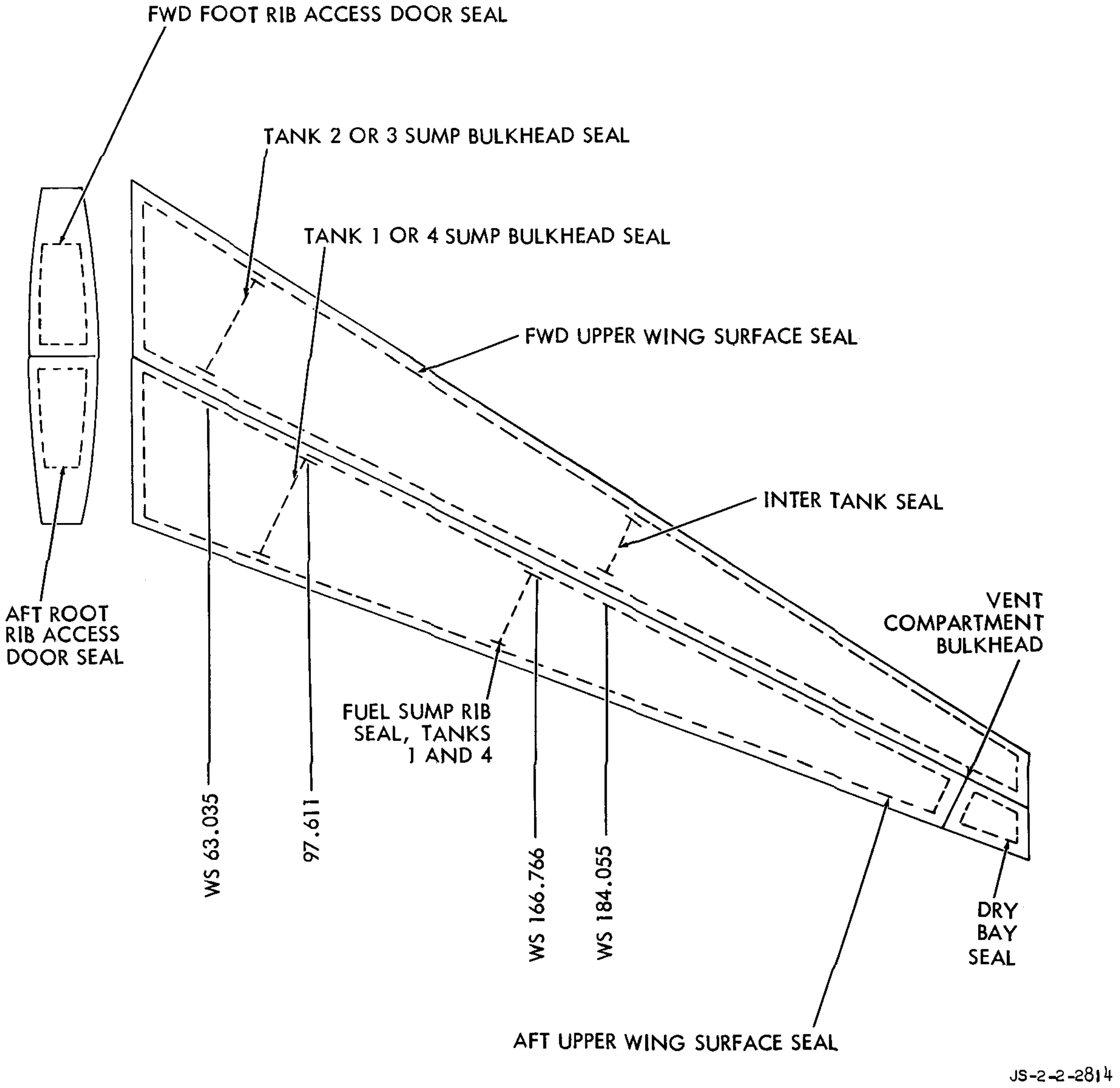
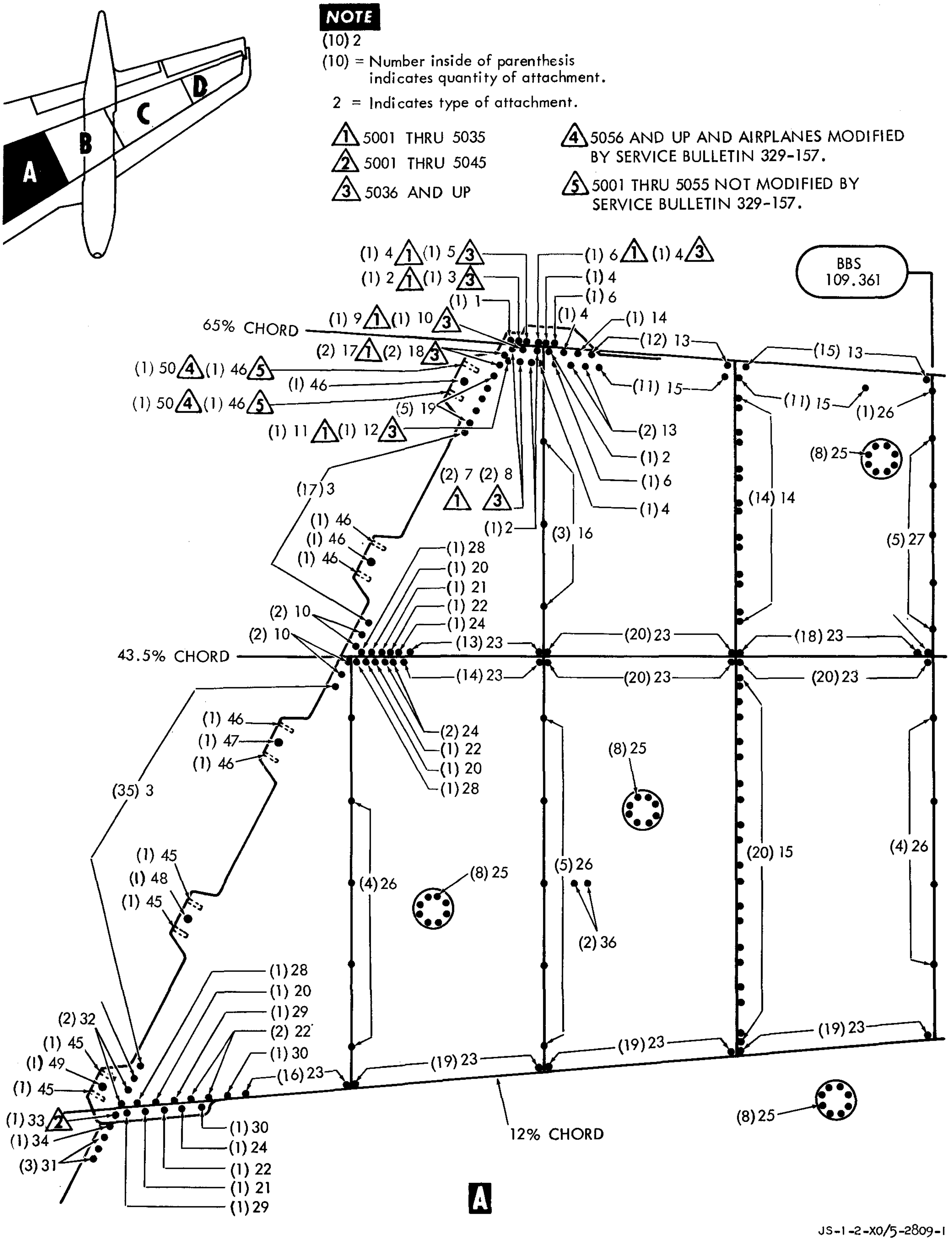

FWD FOOT RIB ACCESS DOOR SEAL
TANK 2 OR 3 SUMP BULKHEAD SEAL
TANK 1 OR 4 SUMP BULKHEAD SEAL
FWD UPPER WING SURFACE SEAL
INTER TANK SEAL
VENT COMPARTMENT BULKHEAD
AFT ROOT RIB ACCESS DOOR SEAL
FUEL SUMP RIB SEAL, TANKS 1 AND 4
WS 63.035
97.611
WS 166.766
WS 184.055
DRY BAY SEAL
AFT UPPER WING SURFACE SEAL
JS-2-2-2814

Figure 28-8. Location of Fuel Tank Mechanical Seals



NOTE

(10) 2

(10) = Number inside of parenthesis indicates quantity of attachment.

2 = Indicates type of attachment.

△1 5001 THRU 5035

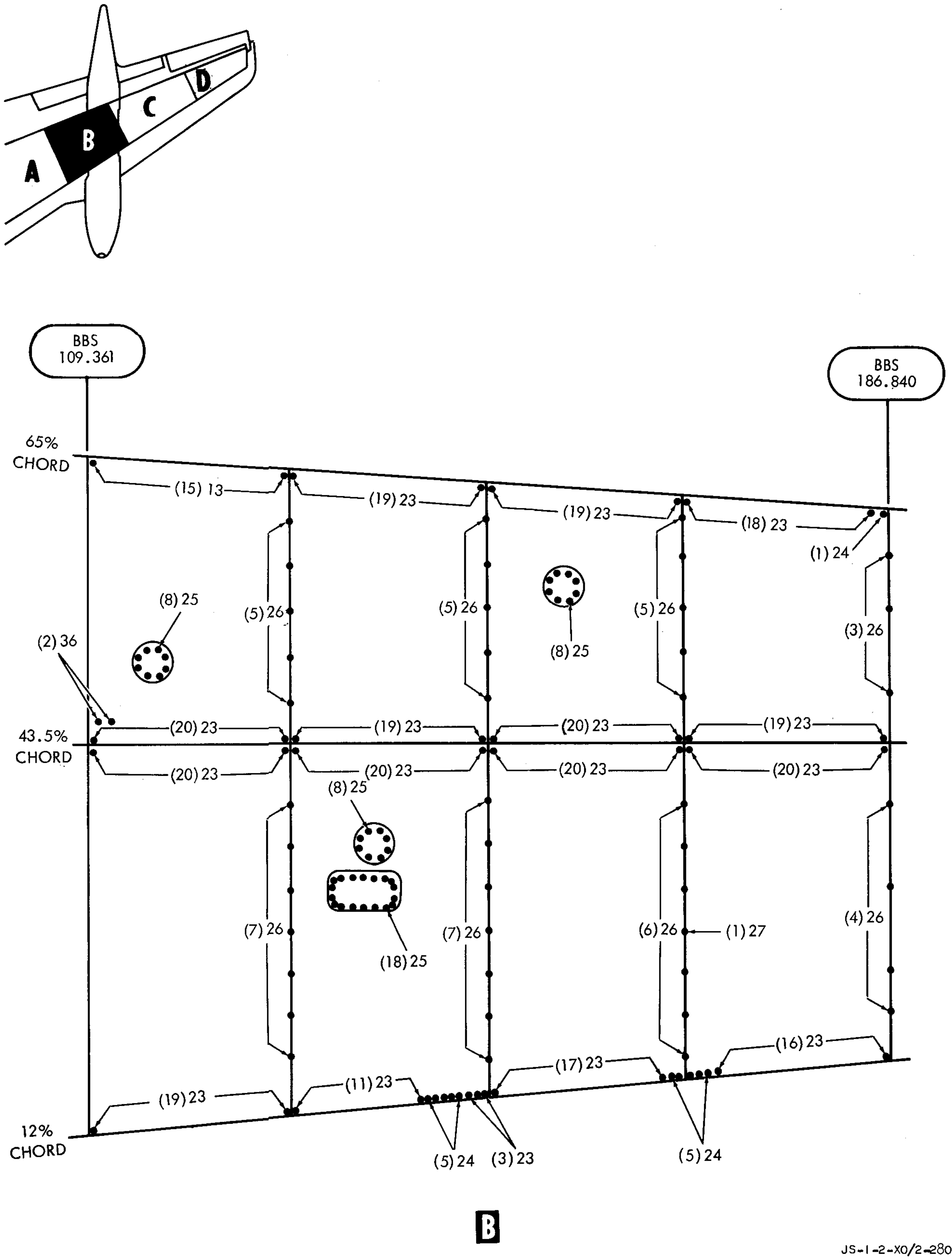
△2 5001 THRU 5045

△3 5036 AND UP

△4 5056 AND UP AND AIRPLANES MODIFIED BY SERVICE BULLETIN 329-157.

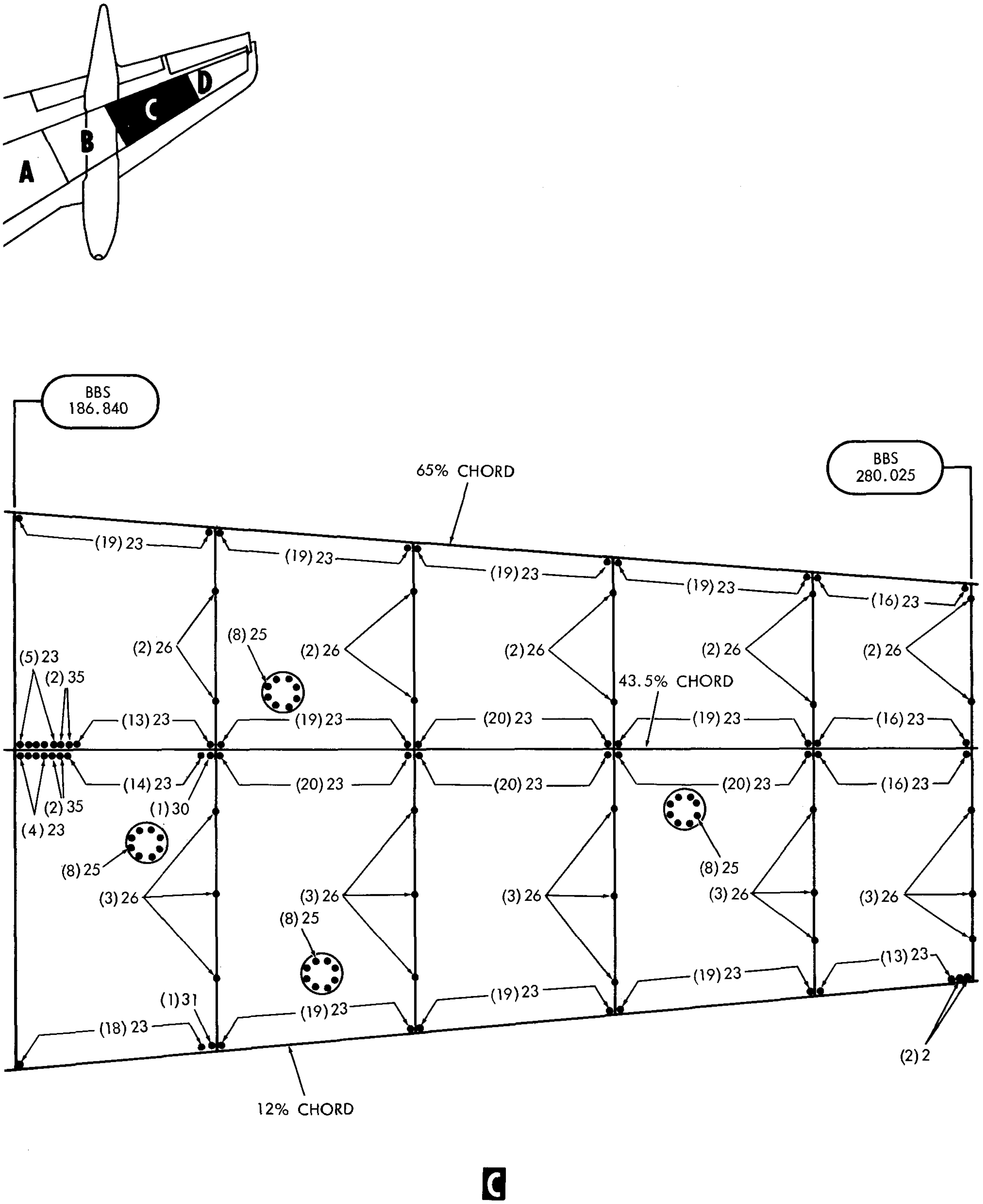
△5 5001 THRU 5055 NOT MODIFIED BY SERVICE BULLETIN 329-157.

Figure 28-9. Upper Wing Panel Attachments (Sheet 1 of 4)



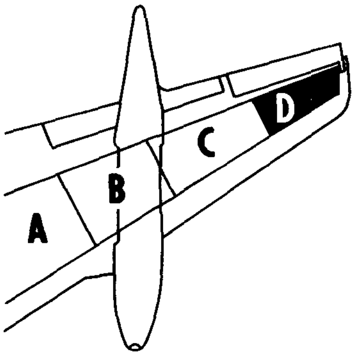
JS-1-2-X0/2-2809-2

Figure 28-9. Upper Wing Panel Attachments (Sheet 2 of 4)



JS-1-2-X0/2-2809-3

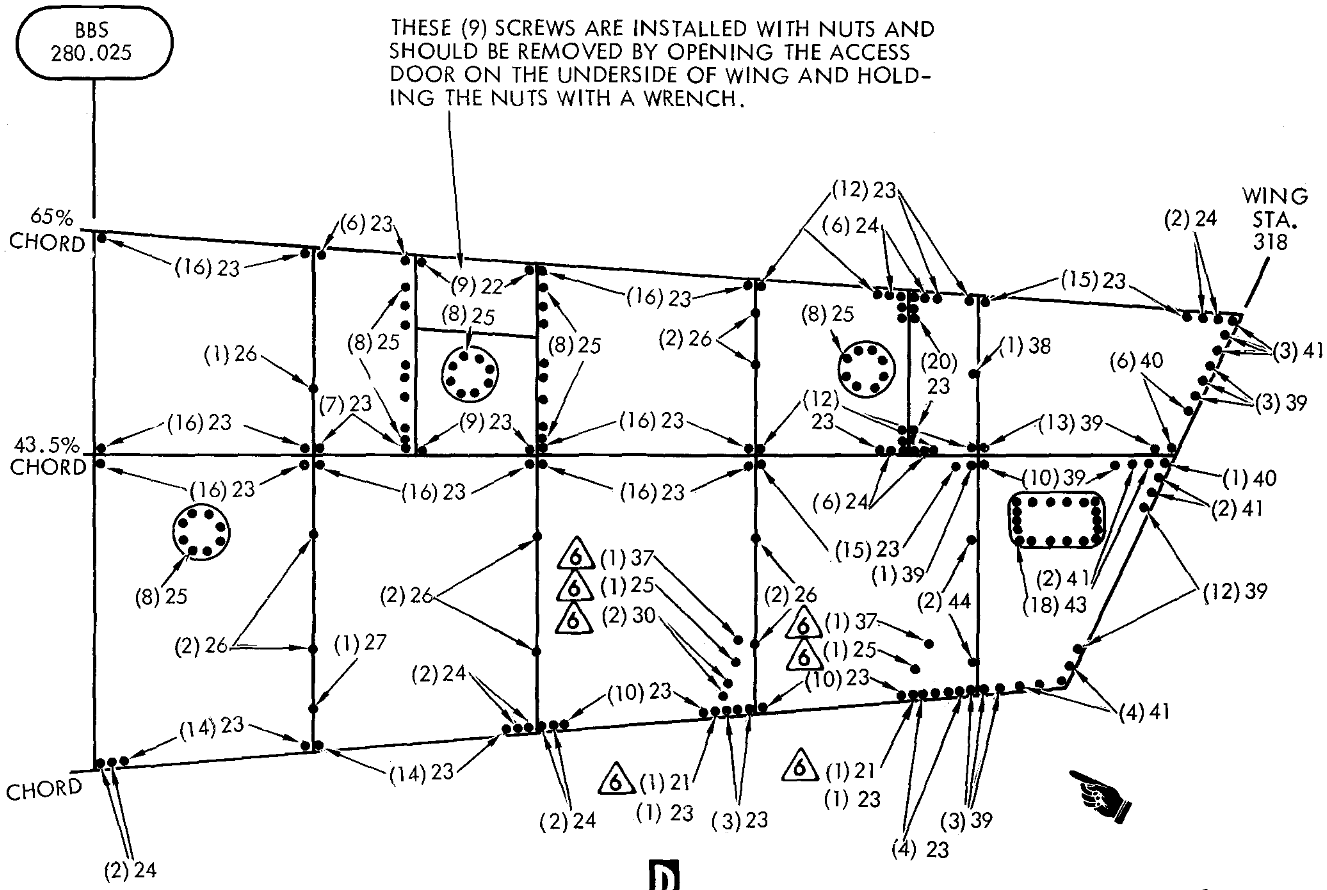
Figure 28-9. Upper Wing Panel Attachments (Sheet 3 of 4)



1. NAS 1633-13
2. NAS 1624-11
3. NAS 1634-12
4. NAS 1624-10
5. NAS 1634-11
6. NAS 1624-9
7. NAS 1624-14
8. NAS 1634-16
9. NAS 1624-12
10. NAS 1634-14
11. NAS 1624-16
12. NAS 1634-18
13. NAS 1624-7
14. NAS 1624-8
15. NAS 1624-6
16. NAS 1624-19
17. NAS 1105-15D
18. NAS 1105-16D
19. NAS 1105-13D
20. NAS 1623-12
21. NAS 1623-9
22. NAS 1623-8
23. NAS 1623-5
24. NAS 1623-7
25. NAS 1623-3
26. NAS 1624-18
27. NAS 1624-17
28. NAS 1623-13
29. NAS 1623-10
30. NAS 1623-6
31. NAS 1634-13
32. NAS 1634-15·
33. NAS 1633-11
34. NAS 1633-12
35. NAS 1123-5
36. NAS 623-4-7
    MS29513-9 O-RING
37. NAS 1623-2
38. NAS 1624-29
39. LS 5915-3-5
40. LS 5915-3-6
41. LS 5915-3-7
42. LS 5915-4-24
43. LS 5915-3-3
44. LS 5915-4-29
45. MS20008-20
46. MS20010-22
47. MS20006-16
48. MS20007-14
49. MS20010-24
50. MS21250-10022

NOTE

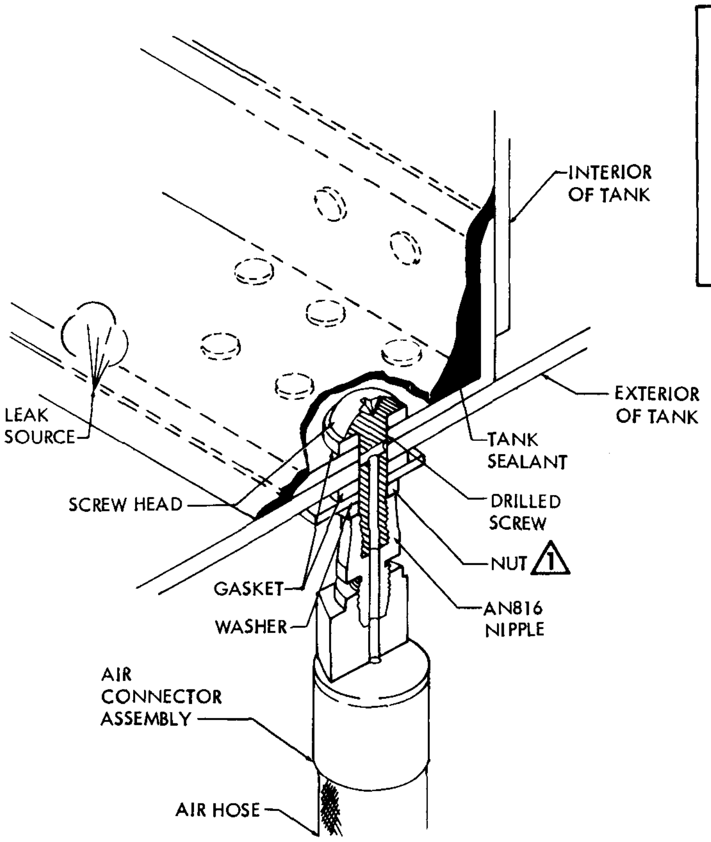
⚠6 USED ONLY ON AIRCRAFT WITH HONEYCOMB LEADING EDGE FLAPS.


THESE (9) SCREWS ARE INSTALLED WITH NUTS AND SHOULD BE REMOVED BY OPENING THE ACCESS DOOR ON THE UNDERSIDE OF WING AND HOLDING THE NUTS WITH A WRENCH.

D

JS-28-X0/5-009-4

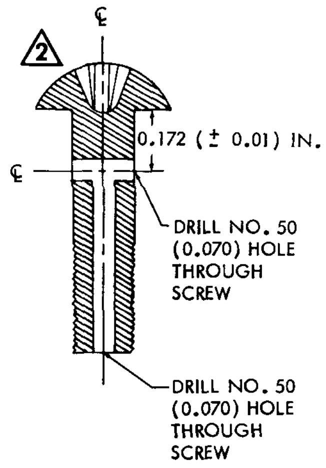
Figure 28-9. Upper Wing Panel Attachments (Sheet 4 of 4)



NOTE

△1 BRAZE NUT TO TOP END OF NIPPLE AS SHOWN. TAP INSIDE DIAMETER OF NIPPLE FITTING WITH SAME THREAD AS NUT.

△2 USE SCREW OF EQUAL OR SMALLER DIAMETER THAN RIVET OR BOLT REMOVED FROM WING.


℄
△2
0.172 (± 0.01) IN.
℄
DRILL NO. 50 (0.070) HOLE THROUGH SCREW
DRILL NO. 50 (0.070) HOLE THROUGH SCREW

DRILLED SCREW DETAILS

Figure 28-10. Drilled Screw Method for Tank Leak Detection

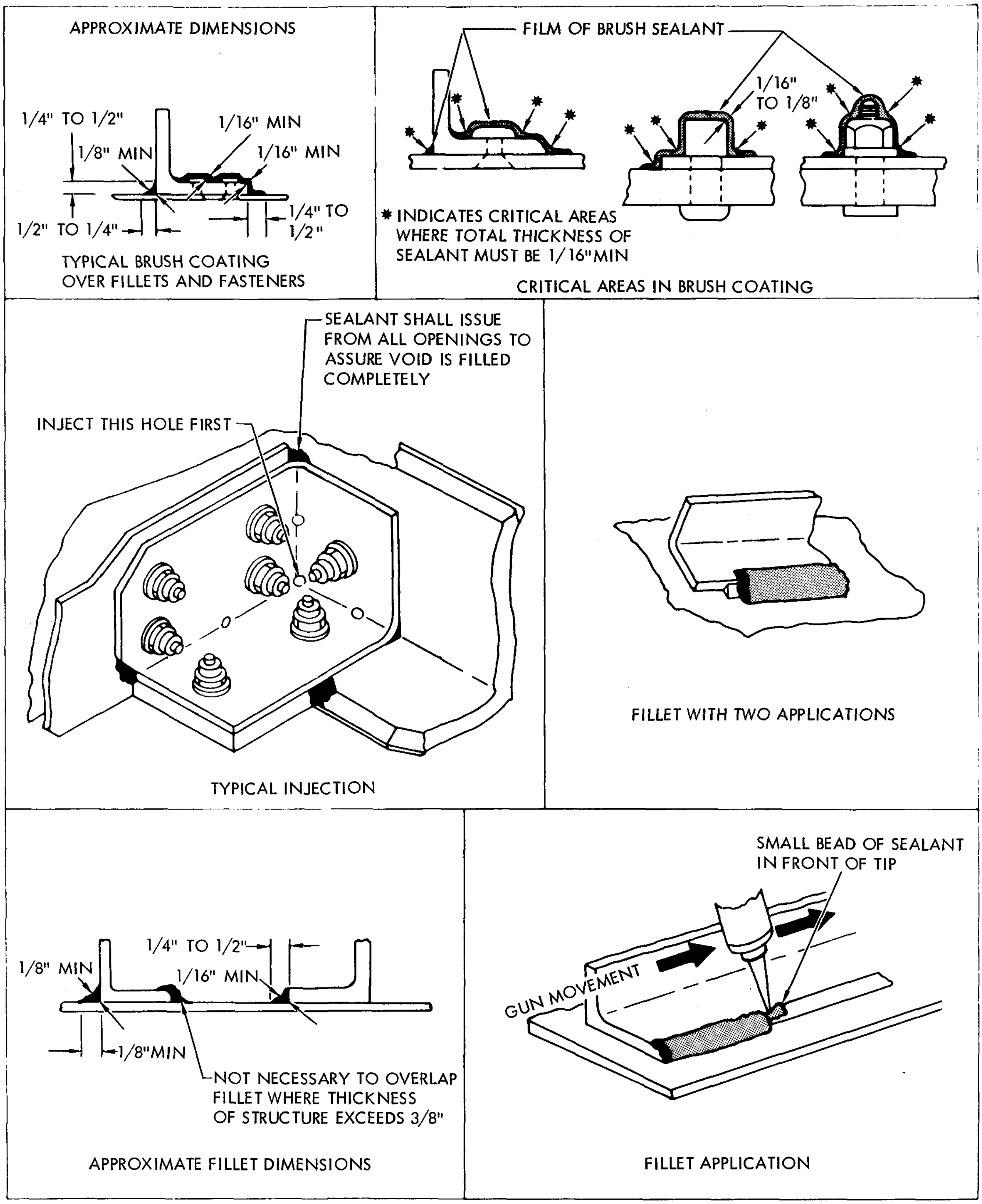

APPROXIMATE DIMENSIONS
1/4" TO 1/2"
1/8" MIN
1/16" MIN
1/16" MIN
1/2" TO 1/4"
1/4" TO 1/2"
TYPICAL BRUSH COATING OVER FILLETS AND FASTENERS
FILM OF BRUSH SEALANT
1/16" TO 1/8"
* INDICATES CRITICAL AREAS WHERE TOTAL THICKNESS OF SEALANT MUST BE 1/16" MIN
CRITICAL AREAS IN BRUSH COATING
SEALANT SHALL ISSUE FROM ALL OPENINGS TO ASSURE VOID IS FILLED COMPLETELY
INJECT THIS HOLE FIRST
TYPICAL INJECTION
FILLET WITH TWO APPLICATIONS
1/8" MIN
1/4" TO 1/2"
1/16" MIN
1/8" MIN
NOT NECESSARY TO OVERLAP FILLET WHERE THICKNESS OF STRUCTURE EXCEEDS 3/8"
APPROXIMATE FILLET DIMENSIONS
SMALL BEAD OF SEALANT IN FRONT OF TIP
GUN MOVEMENT
FILLET APPLICATION
JS-1-2-X0/2-2811

Figure 28-11. Typical Sealant Application

Material Description	Min. Application Life at 77°F and 50% Relative Humidity	Max Tack-free Time at 77°F and 50% Relative Humidity	Manufacturer	Government Specification
Quick-Repair Brush Sealant Used in service for all minor repairs of rivets and fasteners	30 Minutes	10 Hours	Chemical Seal Corp △2 CS-3204, A1/2; Goal Chem Corp △5 GC-408, A1/2; Products Research △7 PR1422, A1/2 or Pro Seal 890, A1/2	MIL-S-8802 Class A-1/2
Standard Brush Sealant Used for sealing rivets, bolts and all fasteners	2 Hours	40 Hours	Chem Seal Corp △2 CS-3204, A-2; Goal Chem Corp △5 GC-408, A-2; Products Research △7 PR-1422, A-2 or Pro Seal 890, A-2	MIL-S-8802 Class A-2
Quick-Repair Fillet and Injection Sealant Used in service repairs for filling gaps and voids or for filleting along seams and joints	30 Minutes	10 Hours	Chem Seal Corp △2 CS-3204, B1/2 Products Research △7 PR 1422, B1/2 or Pro Seal 890, B1/2	MIL-S-8802 Class B-1/2
Standard Fillet and Injection Sealant Used for filleting along seams and joints and for filling gaps and voids.	2 Hours	40 Hours	Chem Seal Corp △2 CS-3204, B-2; Goal Chem Corp △5 GC-408, B-2; Products Research △7 PR-1422 B-2 or Pro Seal 890, B-2	MIL-S-8802 Class B-2

Figure 28-12. Sealing Materials (Sheet 1 of 2)

Material Description	Min. Application Life at 77°F and 50% Relative Humidity	Max Tack-free Time at 77°F and 50% Relative Humidity	Manufacturer	Government Specification
Sealant Corrosion Inhibited, High Adhesion Use for Fay Surface Sealing, Wet-Installation of Fasteners and Repair of MIL-C-27725 Polyurethane Coating	30 Minutes	16 hours	Products Research △7 PR-1422G, A-1/2 or Pro Seal 870, A-1/2	△4 MIL-S-81733 Type I, Class A-1/2
	2 Hours	24 Hours	Products Research △7 PR 1422G, A-2 or Pro Seal 870, A-2	MIL-S-81733 Type I, Class A-2
	30 Minutes	16 Hours	Product Research △7 PR 1422G, B-1/2 or Pro Seal 870, B-1/2	△4 MIL-S-81733, Type II, Class B-1/2
	2 Hours	24 Hours	Products Research △7 PR 1422G, B-2 or Pro Seal 870, B-2	△4 MIL-S-81733 Type II, Class B-2
	4 Hours	32 Hours	Products Research △7 PR 1422G, B-4 or Pro Seal 870, B-4	△4 MIL-S-81733 Type II, Class B-4
Sealant Topcoat, Polyurethane	2 Hours (See Note in Paragraph 28.2. 3.4.b)	4 Hours	Advance Coatings, △1 2-1C-2 Dexter Corp △3 Magna Flex Clear 7-C-27	MIL-C-83019
Adhesion Promoter	–	Drying Time 15 minutes minimum.	Products Research △7 PR 147, PR 148 or Pro Seal 151	–
CS-3100, Potting Compound	60 Minutes	48 Hours cure	Chemseal Division., Flamemaster Co.	MIL-S-8516 Type II, Class 2

NOTE

△1 Advance Coatings and Chem. Co., 4343 Temple City Blvd., Temple City, CA 91780

△2 Chem Seal Corp., 11120 Sherman Way, Sun Valley, CA 91352

△3 Dexter Corp., Midland Division, 7 E. Water St., Waukegan, IL 60085

△4 LAC STM40-111 is equivalent to MIL-S-81733

△5 Goal Chem. Corp., 1337 E. 26th St., Los Angeles, CA 90023

△6 Minnesota Mining and Mfg. Co., Adhesives, Coatings and Sealers Division, 3M Center, St. Paul, MN 55101

△7 Products Research Corp., 2919 Empire Ave., Burbank, CA 91504

Figure 28-12. Sealing Materials (Sheet 2 of 2)

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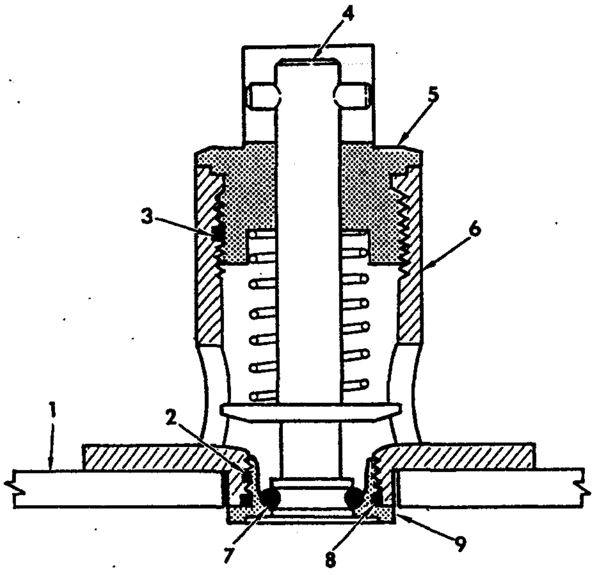

1. LOWER SURFACE OF TANK
2. NYLOK
3. NYLOK
4. POPPET
5. CAP
6. BODY
7. MS29513-9 O-RING
8. MS29513-13 O-RING
9. VALVE SEAT

REPLACEMENT OF DRAIN VALVE SEAL

REPLACEMENT OF SEAL (7) CAN BE ACCOMPLISHED FROM OUTSIDE THE TANK WITHOUT DRAINING FUEL. REMOVE SEAT (9) WITH A BROAD BLADE SCREWDRIVER. THIS ALLOWS THE POPPET (4) TO DROP, CLOSING DRAIN AND EXPOSING SEAL (7) WHICH MAY BE REPLACED. BEFORE REPLACING SEAT (9) CHECK TO SEE THAT NYLOK (2) IS IN PLACE. WITH A NEW O-RING (8), REINSTALL SEAT (9) USING SAME BROAD BLADE SCREWDRIVER. UPON COMPLETION, THE SHOULDER ON THE SEAT (9) SHOULD BE TIGHT AGAINST THE BODY (6) ALL AROUND.

CAUTION

SOME FUEL SPILLAGE WILL RESULT FROM THIS PROCEDURE, AND THE FUEL SERVICING PRECAUTIONS OF SECTION 12.1 SHOULD BE OBSERVED. ALSO, THE CLOSED DRAIN DOES NOT PROVIDE A ZERO LEAKAGE SEAL SO THE SEAL (7) AND SEAT (9) SHOULD BE REPLACED PROMPTLY AFTER REMOVAL.

JS-1-2-XO/2-2812A

Figure 28-12A. Fuel Tank Drain Valve (Airplanes 5046 and Up)

SECTION 28.3
FUEL QUANTITY INDICATING SYSTEM

Contents Page

DESCRIPTION AND OPERATION 28-45
General 28-45
Tank Units 28-45
Quantity Indicators 28-45

TROUBLESHOOTING 28-46

REMOVAL AND REPLACEMENT 28-46
Installation of Fuel Quantity Indicating System Electrical Connectors . . . . 28-46 ,28-46A
Internally-Mounted Probes 28-46A, 28-47
Externally-Mounted Probes 28-47

ADJUSTMENT 28-48
Fuel Quantity Indicating Circuit Calibration 28-48

CARE OF FUEL QUANTITY PROBES 28-50A
Insulation Resistance Test-Tank Units 28-50A
Cleaning Normal Fiberglass Probes 28-50A
Cleaning Alternate Aluminum Probes 28-50B

FIGURES

28-13. Fuel Quantity System Probe Locations 28-51
28-14. Probe Installation and Capacitance 28-52
28-15. Fuel Quantity Indicators and Indicator Calibration 28-53
28-16. Tank Probes Schematic Diagram 28-54

28.3.1 DESCRIPTION AND OPERATION.

28.3.1.1 GENERAL. The fuel quantity indicating system consists of capacitor-type tank units, fuel quantity indicators, compensators, and a fuel quantity test relay. Five fuel quantity indicators, one for each main tank and one with dual concentric pointers for both auxiliary tanks, are on the fuel management panel. Ten tank units and one compensator are in each aft main tank. Six tank units and one compensator are in each forward main tank. Each external tank has five tank units and one compensator. The fuel quantity test relay is energized by a switch on the copilot's control column. When energized, the test relay breaks the circuit that supplies DC power to the fuel quantity indicators, and the indicator pointers move counterclockwise. When the test switch is released, the pointers move clockwise to a position which indicates the correct fuel quantity. AC power is supplied from the main buses on aircraft 5001 thru 5050, and from the essential buses on aircraft 5051 and up. DC power is supplied from the main buses on aircraft 5001 thru 5050 and 5052. On aircraft 5051, 5053 and up, DC power is supplied from the essential buses.

28.3.1.2 TANK UNITS. The tank units (probes) (see figure 28-13) are the sensing elements for the fuel quantity indicating system. The capacitance of the probes changes with the fuel level. The compensator attached to the bottom end of the probe in each tank corrects the fuel quantity indication for changes in the temperature, density, and type of the fuel in the tanks. The probes are arranged in the tanks to minimize indication error due to changes in the attitude of the airplane. Most probes are mounted from the top of tanks so they can be replaced without draining the tank. Two probes in each wing are internally mounted.

28.3.1.3 QUANTITY INDICATORS. The indicator consists of a two-phase motor, gear train, rebalance potentiometer, fuel reserve switch, power unit, and amplifier. Changes in the capacitance of the tank units cause an unbalance of the capacitance bridge circuit in the power unit of the indicator. A phase-sensitive amplifier energizes the two-phase induction motor, which rebalances the bridge circuit by positioning the rebalance potentiometer. The position of the indicator pointer attached to the rebalance potentiometer shaft indicates the fuel quantity in the tank on a graduated dial. Provisions for adjusting the indicator are on top of the indicator case. When the quantity of fuel remaining in a main tank being drained decreases to 1,000 pounds, the fuel reserve switch in the fuel quantity indicator will close and energize the fuel reserve relay.

28.3.2 TROUBLESHOOTING.

WARNING

A malfunction in the fuel quantity gaging system could be a contributing factor in producing an unsafe condition in the fuel tank. Should any circuit breaker associated with the fuel quantity gaging system trip, it must not be reset until the cause of the malfunction has been located and corrected. Additionally, should any electrical malfunction occur in the fuel quantity gaging system, the circuit breaker associated with that system must be opened and must not be reset until the cause of the malfunction has been located and corrected.

PROBABLE CAUSE	ISOLATION PROCEDURE
INDICATOR READING OBVIOUSLY INCORRECT.	
System out of adjustment.	Perform adjustment procedure.
Open 400 cycle or coaxial lead.	Inspect probes for loose captive screw. Check wiring, or capacitance of tank unit circuit.
Defective tank unit.	Replace tank unit.
No power.	Check 115V, 400 cycle power.
Defective amplifier section of indicator.	Replace indicator.
Coaxial lead grounded.	Check wiring (prolonged existence of this condition will burn out fire hazard resistor in indicator).
Foreign matter surrounding compensator.	Open the condensate drain valves - check for water - drain.
Defective test switch.	Continuity check the test switch, replace.
MOVEMENT OF ONE DUAL INDICATOR AFFECTS ANOTHER POINTER.	
Loss of continuity in coaxial shield.	Check and repair wiring.
FUEL RESERVE INCORRECT DURING JETTISON.	
Fuel quantity indicator reserve switch not functioning correctly.	Perform continuity check of indicator switch and associated circuits. Replace indicator or repair wiring.
QUANTITY INDICATORS DO NOT RESPOND TO TEST SWITCH OPERATION.	
Defective FUEL QUANTITY TEST switch.	Perform continuity check of switch. Replace defective switch.
Defective indicator.	Substitute new indicator.

28.3.3 REMOVAL AND REPLACEMENT.

28.3.3.1 INSTALLATION OF FUEL QUANTITY INDICATING SYSTEM ELECTRICAL CONNECTORS (AIRCRAFT 5001 thru 5037). These connectors, two in each wing root bulkhead and one in the bulkhead at the rear of each external tank, are attached to the respective bulkheads with jamnuts. There is no means of preventing rotation of the connector body other than the friction of the jamnut. If the connector turns, the tank seal is broken, and a fuel leak may result. Care should therefore be taken not to use excessive torque when installing the mating plug on the connector. Hand-tightening is sufficient.

28-3.3.1 (Cont).

Note

Any time the tanks are opened, check-tighten the jamnuts to a torque value of 200 to 250 inch-pounds, and reseal the connector inside the tank.

28.3.3.2 INSTALLATION OF FUEL QUANTITY INDICATING SYSTEM ELECTRICAL CONNECTORS (AIRPLANES 5038 and UP). The connectors are located the same as on aircraft 5001 thru 5037, but the connector body is prevented from turning by a doubler. cut out to engage the wrenching flats and secured permanently to structure. The tightening torque for the connector jamnut is 200 to 250 inch-pounds. The connector is sealed inside the tank, but it should not require resealing due to the anti-rotation feature. Install the mating plug hand-tight.

28.3.3.3 REMOVAL OF INTERNALLY-MOUNTED PROBES.

Note

All the probes in the external tanks, and the furthest inboard probe in each integral tank are internally-mounted.

a. Defuel and drain the affected fuel tank in accordance with the procedures in Chapter 12. Disconnect electrical power from the airplane.

b. If the probe is in one of the integral tanks, remove the section of wing fillet beneath the appropriate fuel boost pump access door, and remove the access door in accordance with the instructions in Section 28.0.

c. Disconnect the electrical wiring from the affected probe. *Disassemble the clamps securing* the probe to the structure, and remove the probe.

d. If the probe is in the external tank, gain access to the probes as follows: then disassemble the clamps securing the probes to the structure. Disconnect the electrical wiring and remove.

   (1) Probe in the FWD compartment; gain access through the forward end of the tank by removing the landing and taxi lights and then through the forward bulkhead (item 2, figure 52-5.)

   (2) Probe in the center compartment; gain access through the forward compartment as in step (1) and then into center compartment through the dome bulkhead (item 17, figure 57-9).

   (3) Probe in the upper aft compartment; gain access through the door in the bottom of this compartment (item 41, figure 52-5) which is accessible with the trailing edge flap lowered.

   (4) Probe in the compartment below the wing; gain access through the tank boost pump mounting hole with the boost pump removed (item 45, figure 52-5) or, when the external tank is off the wing, through the access door in the top of the compartment (item 41, figure 52-5).

   (5) Probe in the compartment above the wing; through the access door (item 41, figure 52-5) at the bottom of the compartment which is removable when the external tank is off the wing.

28.3.3.4 REPLACEMENT OF INTERNALLY-MOUNTED PROBES. The replacement procedure is, in general, the reverse of the removal procedure. Check system operation in accordance with the instructions in 28.3.4. Check the replaced access cover for leakage.

28.3.3.5 REMOVAL OF EXTERNALLY-MOUNTED PROBES.

Note

Each integral tank contains one internally-mounted probe at the extreme inboard end. All other probes are externally-mounted, so that it is not necessary to completely defuel the tanks unless an internally-mounted probe is to be changed. (Fuel level should not be more than approximately 2/3 full.) Probes are not located in the areas covered by the external tanks.

a. Disconnect electrical power from the airplane.

b. Remove eight screws securing the affected probe to the tank cover.

c. Loosen the probe mounting flange from the gasket, and raise the probe enough to expose the electrical wiring.

d. Disconnect the electrical wiring from the probe, *taking care not to let the wiring drop into the* tank. Tape the wiring to the wing surface. Remove the probe.

e. If the replacement probe is not to be installed immediately, protect the tank opening from entry of foreign matter with a suitable cover.

28.3.3.6 REPLACEMENT OF EXTERNALLY-MOUNTED PROBES.

a. Remove the temporary cover from the wing opening.

b. Check the condition of the gasket in the wing recess, and replace it if necessary.

Note

On aircraft 5001 thru 5027, the gasket used at the furthest inboard externally-mounted probe location in each forward tank is a special, tapered gasket. If this gasket has to be changed, install the new one with the thicker side in the forward direction.

c. If a gasket has to be changed, remove the old gasket, taking care to avoid scratching the wing skin. Thoroughly clean the sealing surface of the recess with a suitable solvent, such as aliphatic naphtha or Stoddard solvent using a clean cellulose sponge. Wipe the area dry immediately with a clean, dry cellulose sponge.

CAUTION

During this operation, the tank opening is open to atmosphere. Do not allow solvent to enter the tank. Do not use rayon or nylon plastic sponges, since they quickly develop static charges of electricity when rubbed and may ignite either the solvent or the fuel vapor from the tank.

d. Apply a thin coating of sealant to one side of the replacement gasket, and install the gasket in the wing recess with the wet side down. Keep sealant off the upper surface of the gasket.

e. Clean the finish from the countersinks of the attaching screw holes in the probe mounting flange to obtain an electrical bond between the probe and the wing through the attaching screws. Apply Johnson Paste Wax or Simoniz Car Wax (Paste) to the lower surface of the flange.

f. Connect the electrical wiring to the probe, lower the probe into the tank, and secure it with eight screws.

g. Check system operation in accordance with Adjustment Procedures.

28.3.4 ADJUSTMENT.

28.3.4.1 FUEL QUANTITY INDICATING CIRCUIT CALIBRATION.

Test Equipment Required

Figure Number	Nomenclature	Description	Use and Application	Remarks
28-15	Fuel Quantity Indication System Tester (two required)	HT-109 (General Radio) Variable Capacitor or equal.	Two testers are used to simulate the tank unit and compensator during system calibration.	A QUAN-TEST TF20-( )A or GTF-6 variable capacitor, automatic capacitance megger will replace 2 HT-109 and MD-2 testers.
28-15	Patch Cables for HT-109 Tester (two sets required)	Lockheed 6045-U-TSK-P and 6046-U-TSK-P.	To connect the HT-109 tester to airplane wiring.	
28-15	Capacitance Tester.	MD-2A Capacitance Bridge. General Radio or equal.	To calibrate the quantity system tank units.	
28-15	Fuel Gage Patch Cables	Patch Cables (MD-2A).	To connect the Type MD-2A capacitance tester to airplane wiring.	

Note

MD-2 test sets are not approved due to the possibility of hazardous arcing occuring in the fuel tanks if leads are connected incorrectly.

Modify test sets as follows to prevent high voltage from entering the tank wiring circuit.

a. Eliminate the equipment ground wire from Pin B of the test set power to preclude the possibility of high voltage being inadvertently connected to the shield circuit through the equipment ground wire.

b. Add a static ground wire from the ground terminal of the test set to earth ground.

c. Replace the existing wire that grounds the test set to the aircraft with a No. 16 copper wire. This larger wire will provide better protection in case of a system malfunction.

d. On MD-2A and TF-20-( ) test sets remove the connection between the ground wire and Pin B.

28.3.4.2 When any part of the fuel quantity indicating system is repaired or replaced, the following adjustments, or the appropriate portion of them, should be made.

Note

Perform steps (a) through (s) to calibrate an indicator when one has been replaced or is suspected of being out of adjustment, and the trouble does not affect the tank probes or compensators. Accomplish the entire procedure if extensive work has been done on the system which has affected both the units in the tanks and the indicators or electrical wiring between the tanks and the indicators. Perform steps (k) through (ae) if trouble has been isolated to the tank probes or compensators, or to wiring in the tanks.

a. **Energize the main DC bus with 28 (±1) volts on aircraft 5001 thru 5050 and 5052. On aircraft 5051, 5053 and up, energize the essential DC bus with 28 (±1) volts. Energize the main AC bus with 115 (±5) volts on aircraft 5001 thru 5050. On aircraft 5051 and up, energize the essential AC bus with 115 (±5) volts.**

28.3.4.2 (Cont).

Note

The above tolerances must be maintained to calibrate the indicator correctly.

b. **Close the following fuel quantity gage circuit breakers on the copilot's side circuit breaker panel:**

Note

Energize only the indicating system for the indicator to be calibrated.

Tanks 1 and 4

Tanks 2 and 3

Left and Right Auxiliary Tanks

Low Level Warning Lights

c. Disconnect the fuel quantity indication electrical connector at the tank corresponding to the indicator to be calibrated. Connect two HT-109 testers to the connector of the harness leading to the affected indicator, using the special patch cables as shown in figure 28-16. The tank system probes and compensators are to be disregarded during indicator calibration.

Note

One HT-109 tester simulates the tank units. and the other simulates the tank unit compensator. The patch cables should be calibrated to the equipment.

d. If TF-20-( )A or GTF-6 test sets are to be used for indicator calibration, perform the following voltage checks steps e through k.

WARNING

Test leads to the tank and indicator circuits must not be connected before making voltage checks.

e. Ensure that aircraft is properly grounded.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

f. If a maintenance stand is used for tank testing, connect a static ground.

CAUTION

No other power cables are permitted on stand during test.

g. Connect an approved static ground wire from earth ground to GND terminal of test set.

h. Connect the ground wire from aircraft structure to test set GND terminal. Do not connect this wire near any exposed power terminations.

i. Connect TF-20-( )A test set to a source of 115V single phase, 400 Hz power. GTF-6 test set can be connected to a 115V single phase, 60 or 400 Hz power source. Turn test switch on. Position function switch to "A/C TEST-UNSH" position for GTF-6 only.

j. Using a multimeter set on lowest scale, check AC and DC voltage between "tank unit" shell of "COAX" connector on the test set and aircraft structure. If any voltage is indicated, do not proceed with test until the fault has been corrected.

k. Turn test switch to "OFF."

l. Connect test leads to the connector of the harness leading to the affected indicator.

WARNING

Do not connect test leads to the tank circuits when calibrating indicators.

m. Set and lock the simulated compensator tester to 50.7 micro-microfarads (mmfd).

n. Set and lock the simulated tank unit tester to the value listed below (empty) for the indicator being checked, and apply electrical power:

Tank	Value (MMFD) Aircraft 5003, 5041 and Up	Value (MMFD) Aircraft 5001, 5002, 5004 thru 5040
No. 1 and No. 4	202.1	206.6
No. 2 and No. 3	172.6	174.4
LH and RH Aux.	226.2	226.2

o. Turn the EMPTY adjustment control on the indicator until the pointer reads zero.

p. Set and lock the simulated tank unit tester to the value listed below (full), and then turn the full adjustment control on the indicator until

28.3.4.2 (Cont).

the pointer moves to the last dial graduation at the value shown below.

Tank	Tester Setting Air-planes*	Tester Setting Air-planes**	Indicator Reading (Pounds)
No. 1 and No. 4	420.5	425.0	2,700
No. 2 and No. 3	365.0	366.8	2,900
LH and RH Aux.	487.9	487.9	4.100

*Airplanes 5003, 5041 and Up.
**Airplanes 5001, 5002 and 5004 thru 5040.

q. Repeat step (n). If the indicator does not show zero, repeat steps (o) and (p) until both the empty and full indications are obtained without further adjustment.

r. Disconnect the test equipment. Connect the airplane harness to the tank connector.

s. Operate the fuel quantity test switch. The indicator pointer should move counterclockwise, and it should return to the normal position when the switch is released.

Note

If the tanks are completely dry, the indicator will register below zero before the test switch is depressed. In this case, depressing the test switch may not cause any movement of the indicator needle.

t. The following procedure checks the tank probes, compensator and tank wiring. A Type MD-2A, TF-20-( )A or GTF-6 tester and an adapter harness will be required.

CAUTION

The center selector switch on the MD-2A tester should be locked from within the capacitance bridge to prevent movement from the capacitance position. Do not attempt to move the center selector switch from the "CAPACITANCE MMFD" position, as this would cause damage to the tank probe.

WARNING

The Model GTF-6 (GULL) test set is the only set authorized for live tank calibration and maintenance troubleshooting. A live tank is one which contains or has contained fuel and which has not been purged to 20 percent or less LEL (Lower Explosive Limit).

WARNING

Model MD-2A and TF-20 capacitance type fuel quantity gaging system test sets are not to be used for testing aircraft fuel tanks that contain or have contained fuel unless the tanks have been inerted or purged to a fire-safe level of 20% LEL or below.

WARNING

Test leads to the tank circuits must not be connected before making voltage checks.

u. Ensure that aircraft is properly grounded.

v. If a maintenance stand is used for external tank testing, connect a static ground wire from stand to earth ground.

CAUTION

No other power cables are permitted on stand during test.

w. Connect an approved static ground wire from earth ground to GND terminal of test set.

x. Connect the ground wire from aircraft structure to test set GND terminal. Do not connect this wire near any exposed power terminations.

y. Connect MD-2A or TF-20-( )A test set to a source of 115V single phase, 400 Hz power. GTF-6 test set can be connected to a 115V single phase, 60 or 400 Hz power source. Turn test switch on. Position function switch to "A/C TEST-UNSH" position for GTF-6 only.

28.3.4.2 (Cont).

z. Using a multimeter set on lowest scale, check AC and DC voltage between "tank unit" shell of "COAX" connector on the test set and aircraft structure. If any voltage is indicated, do not proceed with test until the fault has been corrected.

aa. Turn test switch to "OFF."

ab. Disconnect the fuel quantity indicating system harness electrical connector at the affected tank, and connect the test harness between the tank socket and the tester.

ac. Place the tester power switch in the "ON" position. Allow the tester to stabilize for 5 minutes.

ad. Read the capacitance on the tester. The capacitance should be within the limits specified below, and never lower than the applicable lower tolerance.

ae. Disconnect the airplane system from the test harness, and connect the airplane harness to the socket at the tank. Remove the test equipment.

Note

If the following limits are not obtained, perform checks on the individual probes and compensators to isolate the cause of the trouble. See figure 28-14 for individual probe readings.

AIRCRAFT 5003, 5041 and UP		AIRCRAFT 5001, 5002, 5004 thru 5040	
FUEL TANK	CAPACITANCE (MMFD) TANK CONTAINING UNUSABLE FUEL OR COMPLETELY DRY AND PURGED	CAPACITANCE (MMFD) TANK CONTAINING UNUSABLE FUEL	CAPACITANCE (MMFD) TANK COMPLETELY DRY AND PURGED
No. 1 and No. 4	202.1 (±3.0)	206.6 (±2.0)	202.1 (±3.0)
No. 2 and No. 3	172.6 (±2.7)	174.4 (±1.7)	172.6 (±2.7)
LH and RH Aux.	226.2 (±3.2)	226.2 (±2.2)	225.6 (±3.4)

28.3.5 CARE OF FUEL QUANTITY PROBES. A normal fiberglass fuel quantity probe which has been immersed in fuel contaminated by water or other foreign matter will reflect a lower value of resistance. Water absorption on the probe is due to the coating being damaged or completely void in some areas. The following resistance values are given for new probes and also apply to used probes that have been thoroughly cleaned, dried, and coated with polyurethane corrosion protective compound. (Products Research #PR1560 or Magna Corporation LAMINAR X500, P/N 7C-27-40). Alternate aluminum tank probes require no renewal of outside coatings.

28.3.5.1. INSULATION RESISTANCE TEST-TANK UNITS. Measure the resistance between the following — external to the aircraft and with the aircraft wiring disconnected.

a. Coax term to ground (wire clamp or ground screw) - not less than 100 megohms.
b. 400 cycle term to ground - not less than 100 megohms.
c. Compensator term to ground - not less than 100 megohms.
d. Coax term to 400 cycle term - not less than 5000 megohms.
e. Coax term to compensator term - not less than 5000 megohms.
f. 400 cycle term to compensator term - not less than 5000 megohms.

28.3.5.2 CLEANING NORMAL FIBERGLASS PROBES.

a. Wash thoroughly using Joy detergent or equal in water at a temperature about 120°F.

b. Rinse thoroughly in clean water - preferably distilled.

c. Dry for 2 to 4 hours at 160°F oven temperature.

d. Dip coat in polyurethane - (protect terminals and mounting surfaces).

e. Allow to cure for 24 hours at 160°F oven temperature.

f. Check capacitance (see figure 28-14) and insulation resistance per paragraph 28.3.5.1 before installing in aircraft.

WARNING

O-T-620 trichloroethane is toxic to skin, eyes and respiratory tract. Skin and eye protection required. Avoid repeated or prolonged contact. Good general ventilation is normally adequate.

28.3.5.3 CLEANING ALTERNATE ALUMINUM PROBES.

a. Aluminum probes are cleaned with a trichloroethane O-T-620 degreaser. No renewal of the coating is required. MIL-S-81533 is alternate for O-T-620.

b. Check capacitance (see figure 28-14) and insulation resistance per paragraph 28.3.5.1 before installing in aircraft.

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NOTE

NUMERALS INDICATE DASH NO. OF LOCKHEED PART NUMBER
NO. 4 TANK OPPOSITE OF NO. 1 TANK.
NO. 3 TANK OPPOSITE OF NO. 2 TANK.

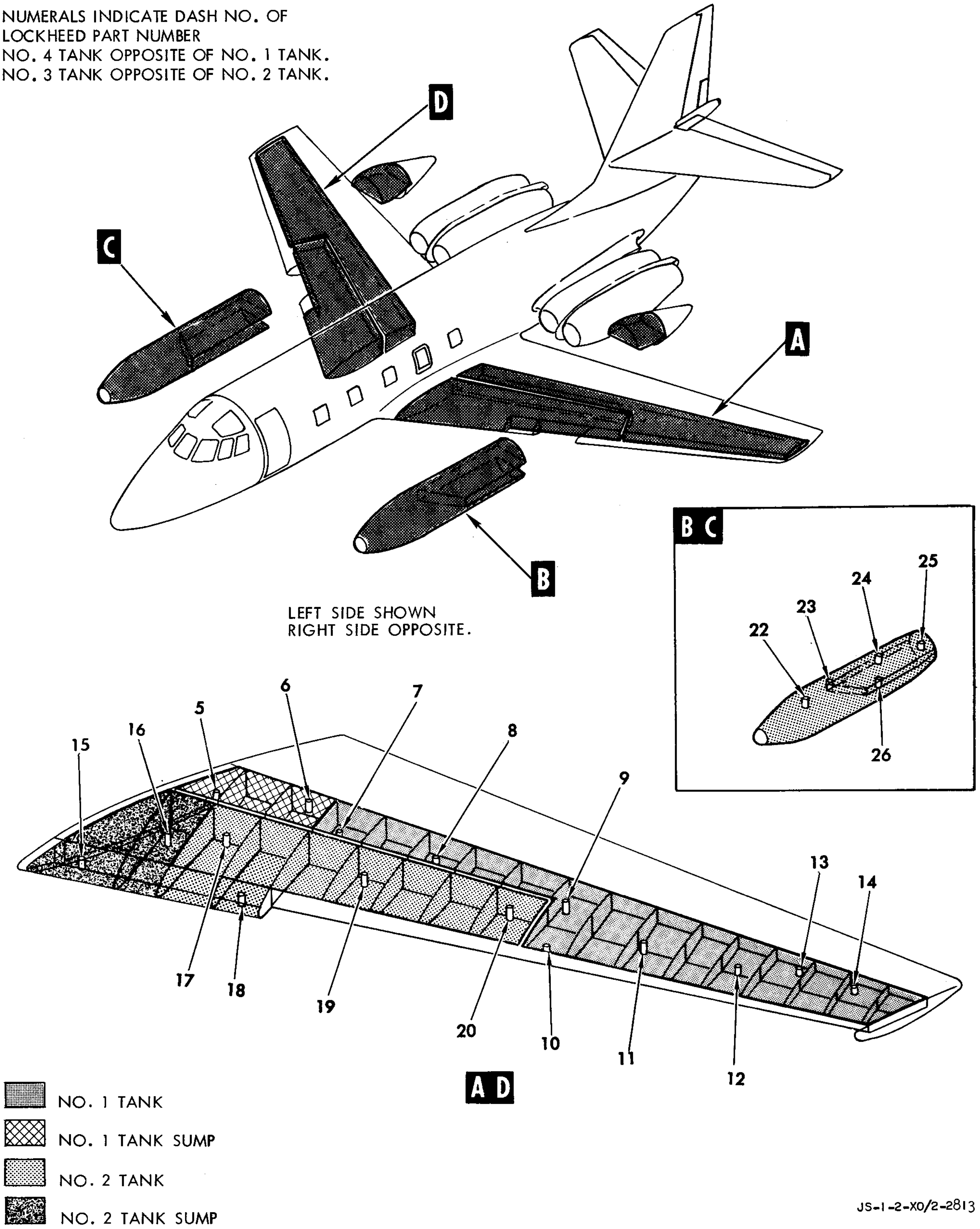


Figure 28-13. Fuel Quantity System Probe Location

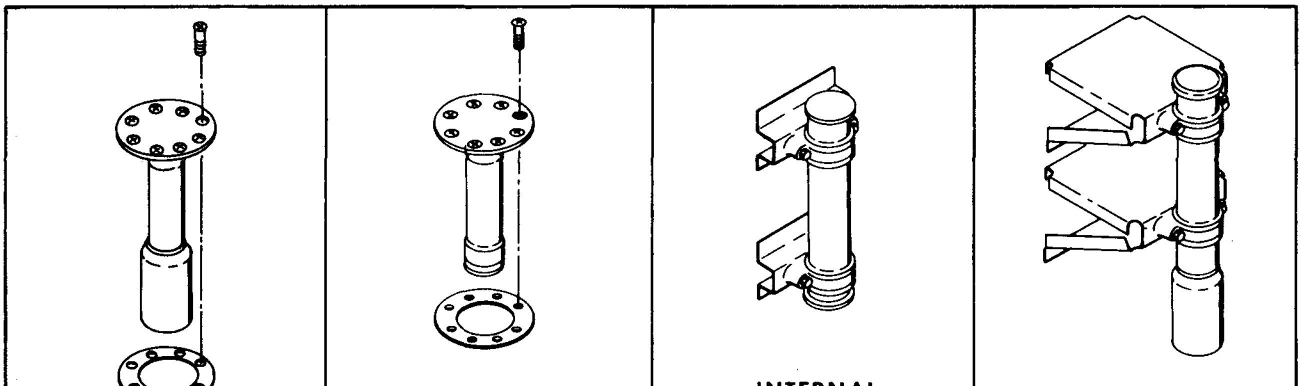
PROBE PART LIST

IDENT. NO.	PART NAME	CAPACITANCE (MMFD.)	QTY.	PART NUMBER	
				LOCKHEED	SIMMONDS
1	Not Used	-	0	-	-
2	Not Used	-	0	-	-
3	Not Used	-	0	-	-
4	Not Used	-	0	-	-
5	Probe*	28.2 ± 0.5	2	JR 1018-5 (-33) ***	381093-02661
6	Probe	20.2 ± 0.5	2	JR 1018-6 (-34)	381096-18660
7	Probe	18.5 ± 0.5	2	JR 1018-7 (-35)	381096-18648
8	Probe	33.3 ± 0.5	2	JR 1018-8 (-36)	381096-18647
9	Probe	23.7 ± 0.5	2	JR 1018-9 (-37)	381096-18130
10	Probe	14.0 ± 0.5	2	JR 1018-10 (-38)	381096-18129
11	Probe	24.3 ± 0.5	2	JR 1018-11 (-39)	381096-18646
12	Probe	21.2 ± 0.5	2	JR 1018-12 (-40)	381096-18645
13	Probe	11.5 ± 0.5	2	JR 1018-13 (-41)	381096-18644
14	Probe	7.2 ± 0.5	2	JR 1018-14 (-42)	381096-17643
15	Probe	17.0 ± 0.5	2	JR 1018-15 (-43)	381056-02665
16	Probe *	31.9 ± 0.5	2	JR 1018-16 (-44)	391067-04679
17	Probe	48.4 ± 0.5	2	JR 1018-17 (-45)	381096-18664
18	Probe	13.7 ± 0.5	2	JR 1018-18 (-46)	381096-18131
19	Probe	41.7 ± 0.5	2	JR 1018-19 (-47)	381096-18663
20	Probe	19.9 ± 0.5	2	JR 1018-20 (-48)***	381096-17662
21	Not Used		0	-	-
22	Probe	76.4 ± 0.8	2	JR 1018-22	381056-02905
23	Probe	68.5 ± 0.7	2	JR 1018-23	381056-02906
24	Probe	29.3 ± 0.5	2	JR 1018-24	381056-02907
25	Probe	28.2 ± 0.5	2	JR 1018-25	381056-01908
26	Probe*	23.2 ± 0.5	2	JR 1018-26	381093-02909
27	Dual Indicator	-	1	JR 1018-27	393003-07206
28	Indicator	-	2	JR 1018-28	393004-06239
29	Indicator	-	2	JR 1018-29	393004-06240

* WITH COMPENSATOR (COMPENSATOR VALUE = 25.0 ± 0.5 MMFD.)

*** INTERCHANGEABLE ALUMINUM PROBE (EXCEPT AS NOTED IN FIGURE 28-16)

PROBE INSTALLATIONS



FLANGED AND GASKET MOUNTING (WITH COMPENSATOR)

FLANGED AND GASKET MOUNTING (WITHOUT COMPENSATOR)

INTERNAL MOUNTED PROBE (WITHOUT COMPENSATOR)

INTERNAL MOUNTED PROBE (WITH COMPENSATOR)

JS-1-2-X0/2-2814

Figure 28-14. Probe Installation and Capacitance

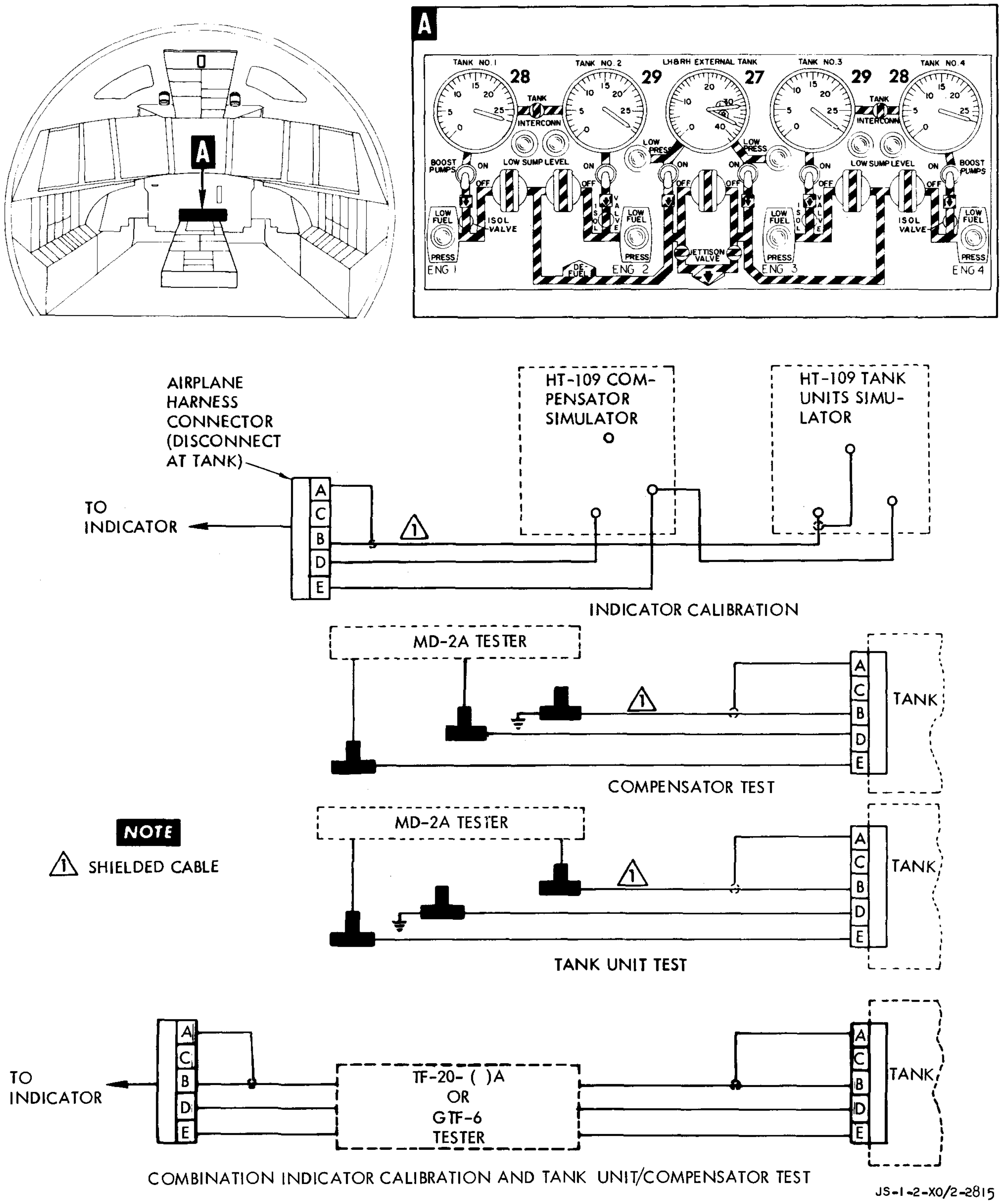
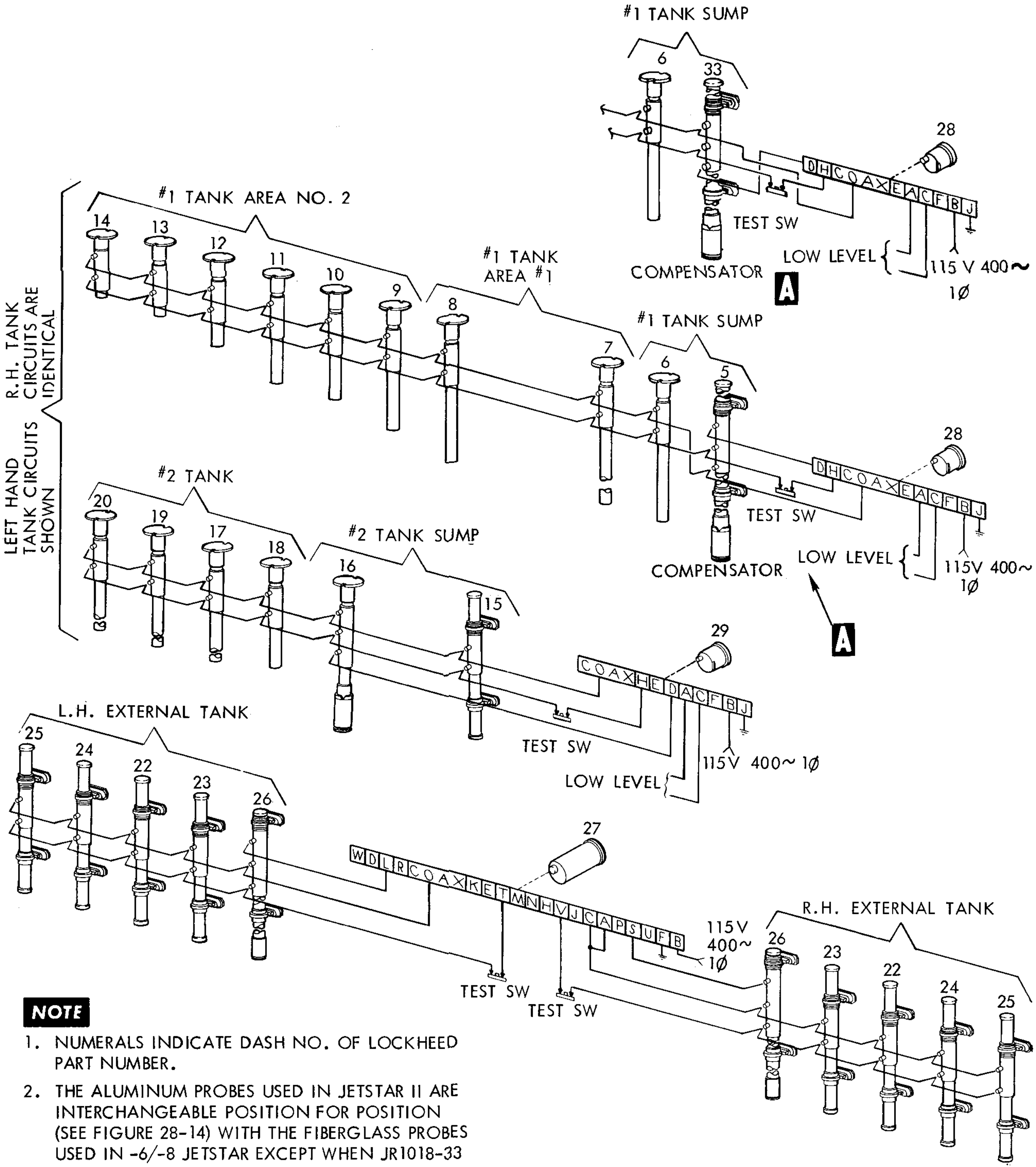


Figure 28-15. Fuel Quantity Indicators and Indicator Calibration



NOTE

1. NUMERALS INDICATE DASH NO. OF LOCKHEED PART NUMBER.
2. THE ALUMINUM PROBES USED IN JETSTAR II ARE INTERCHANGEABLE POSITION FOR POSITION (SEE FIGURE 28-14) WITH THE FIBERGLASS PROBES USED IN -6/-8 JETSTAR EXCEPT WHEN JR1018-33 (ALUMINUM)IS USED IN LIEU OF JR1018-5 (FIBERGLASS). THE WIRING AT THIS PROBE IS SHOWN IN **A**. ALL OTHER ALUMINUM PROBES ARE CONNECTED THE SAME AS FIBERGLASS PROBES.

Figure 28-16. Tank Probes Schematic Diagram

SECTION 28.4
DEFUELING AND JETTISONING SYSTEMS

Contents Page

DESCRIPTION AND OPERATION 28-55
  Defueling System 28-55
  Fuel Jettison System 28-55

OPERATIONAL CHECKOUT 28-56

REMOVAL AND REPLACEMENT 28-56
  Jettison Line Coupling Shroud 28-56
  Jettison Line Flexible Hose 28-56

FIGURES
  28-17. Defueling Provisions 28-57

28.4.1 DESCRIPTION AND OPERATION.

28.4.1.1 DEFUELING SYSTEM. (See figures 28-3 and 28-17.) The manually operated defueling valve and defueling coupling in the aft compartment can be used to defuel all the tanks in the airplane. An access panel in the fuselage outer skin under the fuel valve compartment must be removed to connect a hose to the defueling coupling. Any combination of tanks can be defueled by controlling the fuel flow with the crossfeed valves. The crossfeed separation valve and the crossfeed valves must be opened to defuel all the tanks. The tank boost pumps are used to defuel the tanks.

a. The defueling outlet incorporates an orifice which limits fuel flow through the outlet, thus preventing the fuel boost pumps from overloading at extremely high flows. The orifice, which consists of either an internal machined fitting screwed onto the end of the valve, or an orifice plate mounted in series with the valve, must always be reinstalled if the outlet fitting is dismantled.

b. After fuel has ceased to flow from the defueling valve, there will still be a certain amount of residual fuel left in the tanks. Most of this fuel may be drained through flush-mounted poppet valves in the bottom skin of each tank. Refer to Chapter 12 for information on residual fuel quantities and defueling procedures.

28.4.1.2 FUEL JETTISON SYSTEM. (See figure 28-2.) The fuel lines from the fuel tanks to the crossfeed manifold and the crossfeed system are also used for fuel jettisoning. The only additional plumbing for the jettison system is a fuel line in the aft fuselage and empennage, and a shutoff valve connected to the crossfeed manifold on each side of the crossfeed separation valve. The outlets of these valves are connected to the jettison line. The jettison line is routed aft through the unpressurized compartment and up through the vertical stabilizer to the aft end of the fairing between the horizontal and vertical stabilizers. A flexible hose in the line permits the empennage portion of the line to move with empennage trim movement. A 20-mesh brass wire cloth flame arrestor is fitted to the end of the jettison line. The jettison line exit is aerodynamically designed to prevent splashing or reverse flow.

a. Each joint of the jettison line is enclosed in a shroud which collects fuel coming from an improperly tightened joint, and all shrouds are connected by a small drain line which empties into the valve compartment.

b. A manually operated drain valve is provided for the jettison line. This valve is at the lowest point in the fuel jettison manifold and is used by ground crews to drain fuel or condensate that collects in the fuel jettison line. The stem of the spring loaded valve protrudes approximately 1 inch through the lower fuselage forward of the speed brake. The valve is opened by depressing the valve stem approximately 5/32 inch. Valve malfunctioning or binding of the valve stem in its grommet can prevent the valve from closing; this will result in fuel spilling on the aft fuselage during subsequent fuel jettisoning operations. Operate this drain valve repeatedly after every jettison procedure and at intervals prescribed in Chapter 5.

WARNING

Each time the jettison drain valve is used, check to see there is not binding in the grom-

28.4.1.2 (Cont).

met where it passes through the fuselage skin. Binding at this point can keep the valve spring from closing the valve. This operation also reveals any fuel or condensate entrapped in the jettison manifold. The manifold should be drained completely if fluids are discovered.

c. The airplane fuel boost pumps are used to pump the fuel out of the tanks and overboard through the jettison line (figure 28-4). The external tank fuel can be jettisoned by opening the jettison valves and operating the external tank boost pumps. To stop fuel jettisoning from the external tanks, the jettison valves must be closed with the control switches; otherwise, the tanks will drain completely.

d. The crossfeed valve corresponding to each main tank must be opened to jettison fuel from the main tanks. If a jettison valve fails to open, the crossfeed separation valve can be opened and fuel jettisoned through the other jettison valve. When the quantity of fuel remaining in a main tank from which fuel is being jettisoned decreases to 1,000 pounds, a switch in the fuel quantity indicator closes and energizes the fuel reserve relay (see figure 28-4). This energizes the closing contacts of the crossfeed valve motor corresponding to that tank, and stops fuel being jettisoned from the tank concerned. The crossfeed valve remains closed until the FUEL JETTISON switch is placed in the "OFF" position.

WARNING

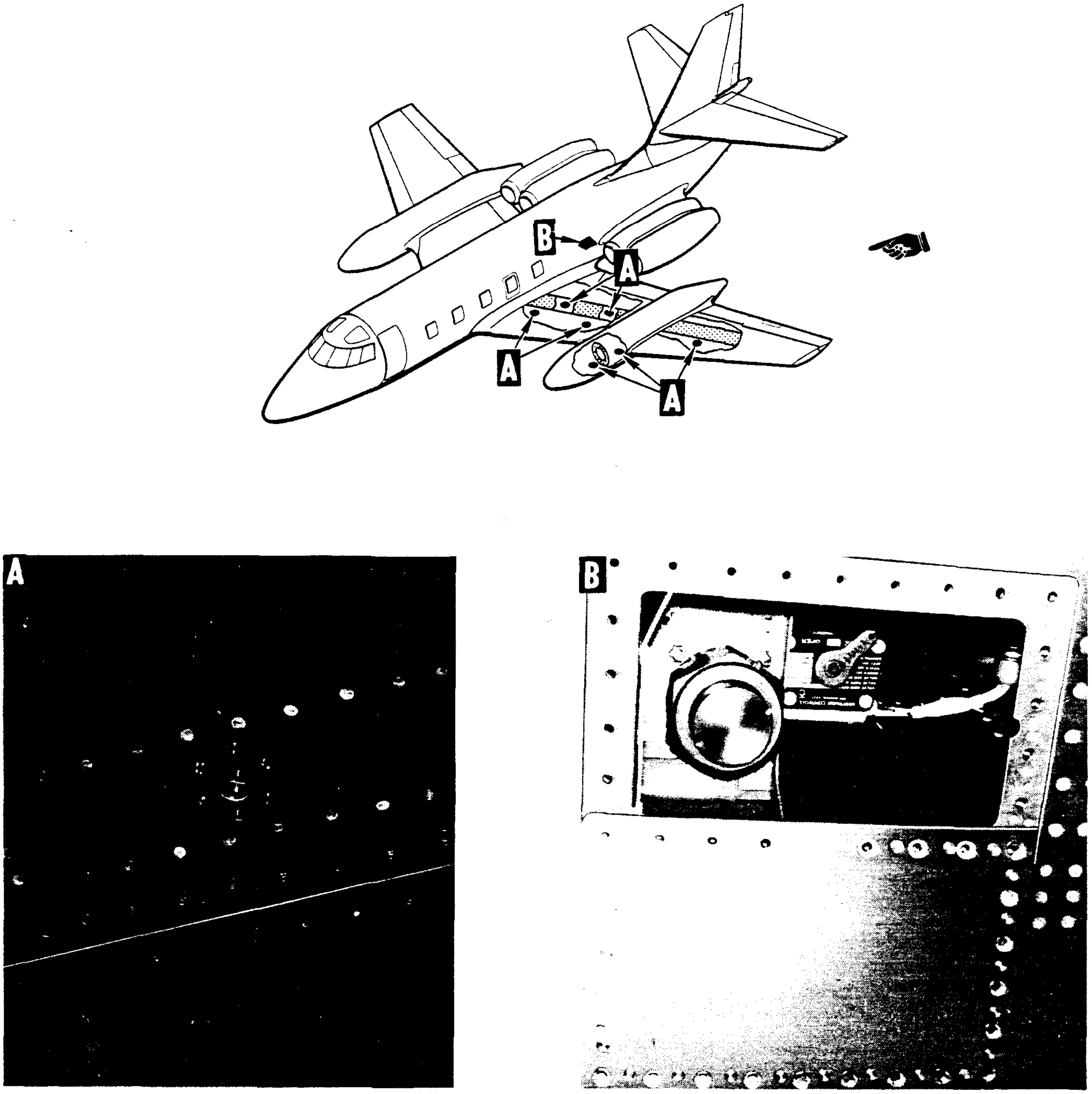
Defects in the empennage hose connections or corrosion in the aluminum lines can cause fuel leakage into the aft equipment area.

28.4.2 OPERATIONAL CHECKOUT. Refer to Section 28.5 for operational checkout procedures of the defueling and jettisoning systems.

28.4.3 REMOVAL AND REPLACEMENT.

28.4.3.1 JETTISON LINE COUPLING SHROUD. (See figure 28-1, detail D.) The metal shroud half is installed over the tube end with a rubber fabric cushion between it and the jettison tube. It is held in place by a hose clamp. The ends of the metal shrouds are sealed with Specification MIL-S-8802, Class A-1/2, sealant (see figure 28-12). A thin coat of this sealant is also applied to the outside of the metal shroud halves at the clamp positions when they are connected with the flexible rubber fabric coupling. The small drain tube located on the forward side should be at the shroud low point. On the aft side, the location of the drain tube is not critical. The small drain line should, however, have a downward slope forward toward the fuel valve compartment for positive drainage. Connections between the drain tubes and the shrouds are made with flexible rubber fabric couplings and clamps. Here too, a thin coat of sealant should be used on the tube at the clamp position when the rubber coupling is installed. All bonding jumpers disturbed should be reinstalled in their original locations with finish removed to assure good electrical contact.

28.4.3.2 JETTISON LINE FLEXIBLE HOSE. (See figure 28-1, detail E.) Removal of the jettison line flexible hose is accomplished after removal of the aft fuselage tail cone as described in Chapter 53. This hose should be installed with a thin coat of Specification MIL-S-8802, Class A 1/2, sealant (see figure 28-12) applied to the outside of the tube at the clamp positions.



POPPET DRAIN VALVE

DEFUELING VALVE

(VIEW LOOKING UP)

JS-2-4-2310

Figure 28-17. Defueling Provisions

SECTION 28.5
FUEL SYSTEM TESTING

Contents Page

PREPARATION 28-59

FUEL FLOW AND CROSSFEED OPERATION 28-59

FUEL JETTISON SYSTEM 28-60

CENTRAL DEFUELING. 28-61

TRANSFER EJECTORS, TANK INTERCONNECT, FLOAT SWITCH AND SUMP LEAKAGE 28-62

TANK-TO-TANK LEAKAGE 28-63

28.5.1 PREPARATION.

a. The airplane static ground line must be connected during these tests. Place the power cart a safe distance from the airplane.

CAUTION

Do not use mercury-filled measuring equipment when testing any part of the fuel system. Mercury is highly corrosive to aluminum and other metals used in the fuel system.

b. With power on the aircraft, close the following circuit breakers:

PILOT'S WAINSCOT CIRCUIT BREAKER PANEL:

Crossfeed Valve, Engine 1
Crossfeed Valve, Engine 2
Crossfeed Valve, Engine 3
Crossfeed Valve, Engine 4
Fuel Jettison, RH & LH
Isolation Valves, Tanks 2 & 3
Tank Interconnect Valves
Fuel Shutoff Valves, Engines 1, 2, 3, & 4
Boost Pump, Tank 2
Boost Pump, Tank 3
Boost Pump Control, Tank 2
Boost Pump Control, Tank 3

COPILOT'S SIDE CIRCUIT BREAKER PANEL;

Boost Pump Control, LH Aux Tank
Isolation Valves, Tanks 1 & 4
Boost Pump Control, RH Aux Tank
Boost Pump Control, Tank 1
Boost Pump Control, Tank 4
Boost Pump, Tank 1
Boost Pump, Tank 4
Boost Pump, LH Aux Tank
Boost Pump, RH Aux Tank

COPILOT'S WAINSCOT CIRCUIT BREAKER PANEL:

Fuel Warning Lights, Low Pressure
Fuel Warning Lights, Filter Clogging

28.5.2 FUEL FLOW AND CROSSFEED OPERATION.

a. Replace the AN824-4 tee (item 15, figure 71-2) with an AN827-4 cross and connect a 0-50 psi pressure gage to the added connection on all four engines. Disconnect the electrical connector at the engine fuel inlet low fuel pressure switch (item 2, figure 71-2) on each engine. Make sure that the fuel tank vent system is clear of any obstruction. Use the throttles to shut off all fuel flow to the engines. Completely fill all fuel tanks according to the instruction in Chapter 12.

b. Position the following switches as designated:

FUEL MANAGEMENT PANEL:

Crossfeed Separation Valve Switch	-Closed
Crossfeed Valve Switches	-Closed
Boost Pump switches	-"OFF."

MAIN INSTRUMENT PANEL:

Fuel Heater switch	-"OFF."
Fire Emergency Handles	-Normal
Fuel Jettison Valve switches	-"OFF."
Fuel Tank Isolation Valve switches	-"OFF."

28.5.2 (Cont).

COPILOT'S CONTROL COLUMN COVER:

Main Tank Interconnect Valve switches -"OFF."

OVERHEAD PANEL:

Engine Start Selector Switch -"OFF."

Fuel Air Bleed Line switch -"CLOSE"

Energize the AC and DC buses.

CAUTION

Bus voltage must be at least 28 volts during the following checks.

RESULT: All main tank low fuel pressure warning lights should remain off. Both external tank low fuel pressure warning lights should remain out except for momentary flashes.

c. Place the No. 1 boost pump switch in the "ON" position.
RESULT: The No. 1 low fuel pressure warning light and the FUEL PRESS LOW module on the annunciator panel should go on momentarily, and then go off. The pressure at the test gage should be at least 19 PSI.

d. Turn the engine start selector switch to engine No. 1 and the fuel air bleed line switch to "OPEN."
RESULT: Fuel should discharge from the overboard vent on the bottom of the nacelle.

e. Return the engine start selector switch to "OFF," and return the fuel air bleed line switch to "CLOSE." Pull the No. 1 fire emergency handle, and bleed pressure downstream of the firewall shutoff valve. Connect the electrical connector to the No. 1 engine low fuel pressure switch (item 2, figure 71-2).
RESULT: The No. 1 low fuel pressure warning light and the FUEL PRESS LOW module on the annunciator panel should go on.

f. Return the fire emergency handle to normal.
RESULT: The No. 1 low fuel pressure warning light should go off.

g. Return the No. 1 boost pump switch to "OFF."
RESULT: The No. 1 low fuel pressure warning light and the FUEL PRESS LOW module on the annunciator panel should go on.

h. Repeat steps (c) through (g) for the No. 2 engine and tank, the No. 3 engine and tank, and the No. 4 engine and tank.

i. Position the left-hand external tank boost pump switch to "ON," and open the No. 1 crossfeed valve.
RESULT: The left-hand external tank low fuel pressure warning light and the FUEL PRESS LOW module on the annunciator panel should go on momentarily, and then go off. The fuel pressure at the No. 1 engine should be at least 28 PSI.

j. Close the No. 1 crossfeed valve, and open the No. 2 crossfeed valve.
RESULT: The fuel pressure at the No. 2 engine should be at least 28 PSI.

k. Close the No. 2 crossfeed valve, and open the crossfeed separation valve and the No. 3 crossfeed valve.
RESULT: The fuel pressure at the No. 3 engine should be at least 28 PSI.

l. Close the No. 3 crossfeed valve, and open the No. 4 crossfeed valve.
RESULT: The fuel pressure at the No. 4 engine should be at least 28 PSI.

m. Return all switches to the positions specified in step (b). Position the right-hand external tank boost pump switch to "ON," and open the No. 4 crossfeed valve.
RESULT: The right-hand external tank low-pressure light and the FUEL PRESS LOW module on the annunciator panel should go on momentarily and then go off. Fuel pressure at the No. 4 engine should be at least 28 PSI.

n. Open the No. 3 crossfeed valve, and close the No. 4 crossfeed valve.
RESULT: The fuel pressure at the No. 3 engine should be at least 28 PSI.

o. Open the No. 2 crossfeed valve and the crossfeed separation valve, and close the No. 3 crossfeed valve.
RESULT: The fuel pressure at the No. 2 engine should be at least 28 PSI.

p. Open the No. 1 crossfeed valve, and close the No. 2 crossfeed valve.
RESULT: The fuel pressure at the No. 1 engine should be at least 28 PSI.

q. Remove pressure gage and replace tee removed in step (a) and return all switches to the positions specified in step (b).

28.5.3 FUEL JETTISON SYSTEM. Connect and clamp a hose to the jettison outlet. The hose should go between the outside of the jettison line and the orifice fairing around the jettison outlet. Connect the other end of the hose to a fuel truck. All the fuel control switches should be positioned as shown in paragraph 28.5.2, step (b).

25.5.3 (Cont).

a. Position the No. 1 tank boost pump switch to "ON." Open the No. 1 crossfeed valve. Position the left-hand jettison switch to "ON."
   RESULT: Fuel should be pumped from the No. 1 tank out of the jettison outlet.

28.5.3 (Cont).

b. Position the left-hand jettison switch to "OFF." Open the crossfeed separation valve, and position the right-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the No. 1 tank out of the jettison outlet.

Note

Inspect inside of aft equipment and empennage areas for any fuel leakage. If leaks are detected, do not continue this check until leaks are repaired.

c. Close the crossfeed separation valve, and position the right-hand jettison switch and the No. 1 tank boost pump switch to "OFF."

d. Position the No. 2 tank boost pump switch to "ON." Open the No. 2 crossfeed valve. Position the left-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the No. 2 tank out of the jettison outlet.

e. Position the left-hand jettison switch to "OFF." Open the crossfeed separation valve, and position the right-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the No. 2 tank out of the jettison outlet.

f. Close the crossfeed separation valve, and position the right-hand jettison switch and the No. 2 boost pump switch to "OFF."

g. Position the left-hand external tank boost pump switch and the left-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the left-hand external tank out of the jettison outlet.

h. Position the left-hand external tank boost pump switch and the left-hand jettison switch to "OFF."

i. Position the No. 4 tank boost pump switch and the right-hand jettison switch to "ON." Open the No. 4 crossfeed valve.
RESULT: Fuel should be pumped from the No. 4 tank out of the jettison outlet.

j. Position the right-hand jettison switch to "OFF," and position the left-hand jettison switch to "ON." Open the crossfeed separation valve.
RESULT: Fuel should be pumped from the No. 4 tank out of the jettison outlet.

k. Position the No. 4 tank boost pump switch and the left-hand jettison switch to "OFF." Close the crossfeed separation valve.

l. Position the No. 3 tank boost pump switch and the right-hand jettison switch to "ON." Open the No. 3 crossfeed valve.
RESULT: Fuel should be pumped from the No. 3 tank out of the jettison outlet.

m. Position the right-hand jettison switch to "OFF," open the crossfeed separation valve, and position the left-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the No. 3 tank out of the jettison outlet.

n. Position the No. 3 tank boost pump switch and the left-hand jettison switch to "OFF." Close the crossfeed separation valve.

o. Position the right-hand external tank boost pump switch and the right-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the right-hand external tank out of the jettison outlet.

p. Position the right-hand jettison switch to "OFF," open the crossfeed separation valve, and position the left-hand jettison switch to "ON."
RESULT: Fuel should be pumped from the right-hand external tank out of the jettison outlet.

q. Position the right-hand external tank boost pump switch and the left-hand jettison switch to "OFF." Close the crossfeed separation valve.

r. Jettison fuel from the four main tanks in unison, by positioning all the main tank boost pump switches to "ON," opening all crossfeed valves, and positioning the jettison switches to "ON." Note the fuel quantity indicator readings when the crossfeed valves close automatically, terminating jettison flow.
RESULT: Each wing tank quantity indicator should show 1,000 (±100) pounds at the time the crossfeed valves close automatically.

s. Make sure that jettison flow has stopped, by monitoring the quantity gages. When the fuel in the main tanks has been depleted to 1,000 pounds in each tank, position the jettison switches to "OFF."
RESULT: All the crossfeed valves should open.

t. Position all the main tank boost pump switches to "OFF," and close all the crossfeed valves.

u. Check the operation of the jettison line manual drain valve by depressing the valve.
RESULT: Fuel should flow from the manual drain valve.

Note

This line should be completely drained before the aircraft is secured.

28.5.4 CENTRAL DEFUELING.

a. Connect a drain hose to the central defueling coupling and to a fuel truck.

b. Open the manual defueling valve, position the No. 1 tank boost pump switch to "ON," the

28.5.4 (Cont).

No. 1 tank isolation valve switch to "OFF," and open the No. 1 crossfeed valve.
RESULT: Fuel should be pumped out of the manual defueling valve.

Note

Fuel may flow from the defuel line by gravity before the boost pump is started. Fuel flow should increase when the pump is started and decrease when the pump is stopped.

c. Close the No. 1 crossfeed valve, with the No. 1 tank boost pump still operating.
RESULT: Fuel flow should stop or show a noticeable decrease.

d. Open No. 1 crossfeed valve. When defueling flow resumes, position the No. 1 tank isolation valve switch to "ON."
RESULT: Fuel flow should stop or show a noticeable decrease.

e. Position the No. 1 tank isolation valve switch and the No. 1 tank boost pump switch to "OFF," and close the No. 1 crossfeed valve.

f. Position the No. 2 tank boost pump switch to "ON," open the No. 2 crossfeed valve, and check that the No. 2 tank isolation valve switch is in "OFF."
RESULT: Fuel should flow from the defueling valve.

g. Close the No. 2 crossfeed valve.
RESULT: Fuel flow should stop or show a noticeable decrease.

h. Open the No. 2 crossfeed valve. When flow resumes, place the No. 2 tank isolation valve switch in "ON."
RESULT: Fuel flow should stop or show a noticeable decrease.

i. Place the No. 2 tank isolation valve switch in "OFF." Turn off the No. 2 tank boost pump.

j. Repeat steps (b) through (i) for tanks No. 3 and No. 4.

k. Position the left-hand external tank boost pump switch to "ON."
RESULT: Fuel should flow from the defuel line.

l. Position the left-hand external tank boost pump switch to "OFF."
RESULT: Fuel flow should decrease or stop.

m. Open the crossfeed separation valve, and position the right-hand external tank boost pump switch to "ON."
RESULT: The fuel flow from the defueling line should increase.

n. Position the right-hand external tank boost pump to "OFF."
RESULT: The fuel flow from the defueling line should decrease or stop.

o. Defuel the left and right-hand external tanks alternately until all the fuel is drained from the tanks. Observe the external tank low fuel pressure warning lights and the fuel quantity indicators on the fuel management panel.
RESULT: When the low-pressure lights go on, the fuel quantity gages should indicate empty.

p. Place the external tank boost pump switches in "OFF," and close the crossfeed isolation shutoff valve and the manual defueling valve. Install the cap on the defueling outlet. Install the access cover over the opening.

28.5.5 TRANSFER EJECTORS, TANK INTERCONNECT, FLOAT SWITCH AND SUMP LEAKAGE

a. Drain all four main tanks.

b. Add 1000 pounds of fuel to No. 1 tank and to No. 4 tank through the gravity fillers.
RESULT: All four LOW SUMP LEVEL warning lights should be on.

c. Position No. 1 and No. 4 main tank boost pump switches "ON."
RESULT: After a few minutes the No. 1 and No. 4 LOW SUMP LEVEL warning lights should go out.

d. Position the No. 1 and 2 and No. 3 and 4 MAIN TANK INTERCONNECT switches to "ON."
RESULT: Fuel should flow from No. 1 tank to the No. 2 tank and from No. 4 tank to No. 3 tank as indicated on the fuel quantity indicators. After a few minutes, the No. 2 and No. 3 LOW SUMP LEVEL warning lights should go out when the quantity of fuel in each tank reads 300 (±100) pounds.

e. When No. 2 and No. 3 quantity indicators read approximately 400 pounds, position No. 1 and 2 and No. 3 and 4 MAIN TANK INTERCONNECT switches to "OFF."

f. Position No. 1 and No. 4 main tank boost pump switches "OFF."

g. Open the No. 2 and No. 3 tank drain valves located just outboard of the sump and check leakage through sump end rib.
RESULT: Leakage rate should not exceed 0.5 gal/min.

h. Close drain valves.

i. Repeat steps (a) through (h) except fill No. 2 tank and No. 3 tank and check for flow into, and sump leakage in, No. 1 and No. 4 tanks.

28.5.6 FUEL SOAK AND STAND PIPE TEST FOR EXTERNAL LEAKAGE.

a. The following test should be performed after repairing any leak that required the tank to be opened. Also, when the tanks are to be opened to repair a known leak, the following stand pipe test will aid in locating leak source and detecting additional slow seeps or potential leaks:

Note

1. Both integral tanks in a wing may be checked simultaneously by opening the tank inter-connect valve.
2. Variations in stand pipe fluid or flow out the tank vents caused by expansion of fuel within the tank is not considered as leakage.
3. Refer to Section 12.1 for fuel servicing and safety precautions.

b. Fill each tank to maximum capacity, replace the filler caps, and allow the tanks to stand for four (4) consecutive hours. Plug the vent outlets with vent plugs (LAC P/N 6096 TSF-J, 6138-TSF-J or equal). Install stand pipes (LAC P/N 6065-TSF-J or equal) in the fuel tank filler ports. Slowly fill each stand pipe to a 30 inch head. Maintain this head for one and one-half (1.5) hours. There shall be no external leakage. Remove the stand pipe and the vent plug. Restore the fuel system to normal.

28.5.7 TANK-TO-TANK LEAKAGE. If tank-to-tank leakage is suspected or whenever the forward top wing panel is removed and replaced, the following inter-tank leakage check should be made:

a. De-fuel the aft tank of the wing. Open the three (3) drain valves in the aft tank and allow the trapped fuel to drain. Install a stand pipe (LAC P/N 6065-TSF-J or equal) in the forward fuel tank filler port. Slowly fill the stand pipe until fuel runs out of the vent tube. Plug the forward vent tube with a vent plug (LAC P/N 6096 TSF-J, 6138-TSF-J, or equal).

b. Fill the stand pipe to a 30 inch head and maintain this head for one (1) hour on the forward tank. With all drain valves in the aft tank open, collect the fuel drainage during the last thirty (30) minutes of the hour. The fuel collected in inter-tank leakage shall not exceed one hundred (100) fluid ounces. Remove the stand pipe and the vent plug. Restore the fuel system to normal.

CHAPTER 29
HYDRAULIC POWER

TABLE OF CONTENTS

Section		Page
29.0	HYDRAULIC POWER - GENERAL	29-3
29.1	NORMAL (NO. 1) AND AUXILIARY (NO. 3) HYDRAULIC SYSTEMS	29-19
29.2	STANDBY (NO. 2) HYDRAULIC SYSTEM	29-35

SECTION 29.0

HYDRAULIC POWER - GENERAL

Contents ... Page

DESCRIPTION AND OPERATION ... 29-3
  Normal (No. 1) and Auxiliary (No. 3) Hydraulic Systems ... 29-3
  Standby (No. 2) Hydraulic System ... 29-3
  Landing Gear ... 29-4
  Wheel Brake and Anti-Skid Systems ... 29-4
  Nose Steering System ... 29-4
  Flight Controls ... 29-4
  Thrust-Reverser System ... 29-4
  Accumulators ... 29-4
  Hydraulic Cylinders and Actuators ... 29-4
  Special Seals ... 29-4
  Fluid Additives ... 29-4-1

GENERAL MAINTENANCE INFORMATION ... 29-4-1
  Hydraulic Fluid ... 29-4-1
  Fluid Contamination ... 29-4A
  Maintenance Practices ... 29-4A
  Flushing ... 29-4A
  Flushing Hydraulic System After Failure of Engine-Driven Hydraulic Pump ... 29-4A
  Cleaning Metal Filter Elements ... 29-5
  Indicating Filter ... 29-6
  Pressure Line Filter ... 29-6
  Flareless Tube Fittings ... 29-6
    Description ... 29-6
    Maintenance of Flareless Tube Unions ... 29-6
    Flareless Tube Union Tightening ... 29-6
    Torque-Tightening Flareless Unions ... 29-6
    Position-Tightening Flareless Unions ... 29-6
    Replacement of Flareless Tube Fittings ... 29-7
    Inspection and Final Assembly After Presetting Flareless Unions ... 29-7
    Normal and Standby Hydraulic Pump Connection Torque Adjustments ... 29-8
  Flexible Hose Installation ... 29-8
  Preservation of the Hydraulic Systems During Prolonged Inactivity ... 29-8
  Depreservation of the Hydraulic Systems After Prolonged Inactivity ... 29-8
  Extreme Climatic and Environmental Information ... 29-8
  Hydraulic Pump Priming ... 29-8A

TROUBLESHOOTING ... 29-8A
  Hydraulic Fluid Transfer ... 29-8A

FIGURES
  29-1. Flareless Fitting Components and Assembly Sequence ... 29-9
  29-2. Preset Flareless Sleeves ... 29-9
  29-3. Torque Values for Tightening Flareless Tube Fittings ... 29-10
  29-4. Minimum Tube-Wall Thickness for Presetting ... 29-10
  29-5. Presetting Tool ... 29-11
  29-6. Presetting by Hand ... 29-11
  29-7. Turn Values for Presetting by Hand ... 29-12
  29-8. Corrosion Preventive Compounds ... 29-13
  29-9. Hydraulic System Schematic Diagram (Airplanes 5001 thru 5040) ... 29-15
  29-9A. Hydraulic System Schematic Diagram (Airplanes 5041 thru 5045) ... 29-16A
  29-10. Hydraulic System Schematic Diagram (Airplanes 5046 thur 5092, 5094 thru 5096) ... 29-17
  29-10A. Hydraulic System Schematic Diagram (Airplanes 5093, 5097, and Up) ... 29-18A

29.0.1 DESCRIPTION AND OPERATION.

29.0.1.1 NORMAL (NO. 1) AND AUXILIARY (NO. 3) HYDRAULIC SYSTEMS. (See figures 29-9 and 29-10.) A normal (No. 1) hydraulic system and an auxiliary (No. 3) hydraulic system power the speed brake, thrust reversers, landing gear and doors, wheel brakes, nose wheel steering, one side of the aileron booster assembly, one side of the elevator booster assembly, and the leading and trailing edge wing flaps. The normal system pump is driven by the No. 2 engine. The auxiliary pump is driven by an electric motor. The auxiliary pump uses the normal system reservoir and system lines, and automatically supplements

29.0.1.1. (Cont).

normal system power during landing-gear-up and normal trailing edge flap-down actuation. The auxiliary pump can be manually selected to power any component that the normal pump can power.

29.0.1.2 STANDBY (NO. 2) HYDRAULIC SYSTEM. (See figures 29-9 and 29-10.) A standby (No. 2) hydraulic system normally powers one side of the aileron booster assembly and one side of the elevator booster assembly, but can be selected to power the wing flaps, the nose wheel steering (5046 & up) and wheel brakes. When using the standby system to power the wheel brakes, the anti-skid protection is lost. When the standby system is used to power the wing flaps, both leading and trailing edge flap actuation is controlled through the trailing edge flap control circuits. The standby system pump is driven by the No. 3 engine.

29.0.1.3 LANDING GEAR. JetStar airplanes have retractable landing gear made up of two main and one nose gear. One actuating cylinder on each gear retracts to extend the gear and extends to retract the gear. Hydraulic power for gear actuation is supplied by the normal hydraulic system with the auxiliary hydraulic pump supplementing this power during gear-up-actuation. The auxiliary pump can be manually selected to actuate the gear independently of the normal system pump. In the absence of both normal and auxiliary power, the gear can be extended by releasing the up-locks and discharging air bottles into the hydraulic actuating cylinders to force the gear into the downlocks.

29.0.1.4 WHEEL BRAKE AND ANTI-SKID SYSTEMS. Four full-circle, multiple-disk brake assemblies, one on each wheel, supply braking action by friction between the disks. The normal hydraulic system normally powers the brake system, but the auxiliary hydraulic pump can be used. In the absence of both normal and auxiliary hydraulic power, the standby hydraulic system can be selected to power the brakes; but when in this configuration, the anti-skid is inoperative.

29.0.1.5 NOSE STEERING SYSTEM. A steering control valve on the nose gear is controlled by the pilot and routes hydraulic fluid to a steering cylinder on the nose gear strut assembly. This cylinder acts through a torque arm to turn the nose strut piston and thus the nose wheels for steering the airplane on the ground.

29.0.1.6 FLIGHT CONTROLS. The ailerons and elevators are hydraulically assisted flight controls, while the wing flaps and speed brake are hydraulically actuated flight controls. The ailerons and elevators are normally powered by both the normal and the standby hydraulic systems, but either system is sufficient to power them independently of the other. The leading and trailing edge flaps are normally powered by the normal hydraulic system, and the auxiliary hydraulic system pump can be selected to power the flaps independently of the normal system pump. In the absence of both normal and auxiliary hydraulic power, the standby hydraulic system can be selected to actuate the flaps. When using standby system power to actuate the flaps, the trailing edge flap control circuits control both the leading and the trailing edge flaps.

29.0.1.7 THRUST-REVERSER SYSTEM. Four thrust-reverser assemblies, one on each engine, are normally powered by the normal hydraulic system. The auxiliary hydraulic pump can be used to power the thrust-reversers. Thrust-reverser levers on the throttles initiate signals to move the thrust-reverser doors into the path of engine exhaust gases to provide reverse thrust.

29.0.1.8 ACCUMULATORS. Several JetStar systems contain accumulators. These accumulators, while different in size, are similar in design, construction, and function. All are of the "free piston" type consisting of an aluminum piston within a steel cylinder which has aluminum end caps. They also contain seals, backup rings, etc. Accumulator precharge should be checked during the pre-flight inspection. The precharge can be checked only when the hydraulic system pressure is zero and when the fluid volume has returned to the reservoir.

Accumulators should not require frequent recharging. If the air precharge is lost over a short period of time (one flight) or if recharging is required frequently (one month or less), it is an indication that the accumulator or the associated air plumbing is leaking. External leaks should be corrected. Internal leaks require the accumulator to be removed and repacked.

Internal leaks may result in air entering the hydraulic fluid. Air in the landing gear system may result in damage to the gear doors or wheel well structure. Air in the thrust reverser system may result in damage to thrust reverser components. As with air contamination from any source, the system should be bled and then cycled at a reduced pressure until all evidence of trapped air ceases.

29.0.1.9 HYDRAULIC CYLINDERS AND ACTUATORS. The hydraulic cylinders and actuators on the JetStar are of conventional design and qualified for service using standard seals and 0-rings. Under normal operating conditions, 0-ring seals will perform satisfactorily; however, proper 0-ring installation techniques are necessary to prevent spiraling and rolling in glands. All actuators need repacking occasionally due to wear. See Chapter 5 for recommended overhaul schedules.

29.0.1.9.1 SPECIAL SEALS. Hydraulic actuator seals may leak occasionally when the ambient temperature is low and the hydraulic pressure is zero or at a reduced level. On JetStar airplanes, the landing gear cylinders and the speed brake actuators are the most susceptible to this type of leakage. This leakage, if not severe, does not require that the actuator be repacked. Special seals

29.0.1.9.1 (Cont).

may be installed in actuators where severe leakage is a chronic problem. These special seals replace the normal O-ring and retainer ring or rings. AGT ring seals manufactured by Greene, Tweed & Co. have proven to be useful in certain JetStar applications. Operators with a requirement for a special seal application should contact the JetStar Support Department of the Lockheed Aeronautical Systems Company for additional information.

29.0.1.9.2 FLUID ADDITIVES. MIL-H-5606 hydraulic fluid contains all the additives necessary for good shock strut and hydraulic system operation and other additives are not required. In particular, organophosphate or phosphate ester based additives deteriorate 0-ring seals and should be avoided.

29.0.2 GENERAL MAINTENANCE INFORMATION.

CAUTION

Do not pressurize the normal (No. 1) hydraulic system with an external supply if the No. 2 engine fire handle is pulled. This could result in a ruptured reservoir or other damage to the system.

29.0.2.1 HYDRAULIC FLUID. Specification MIL-H-5606 hydraulic fluid is used throughout the hydraulic system of JetStar airplanes. When servicing hydraulic systems, make sure that the fluid used is clean. Never use fluid that has been left in an open container even though impurities cannot be seen. Dust particles, metal chips, or other impurities can damage seals, clog small passages, and cause improper functioning of components.

WARNING

Hydraulic fluid may contain Tricresyl Phosphate (TCP) which may be absorbed through the skin and produces paralysis if taken internally. Appropriate protective measures should be taken to avoid such exposures.

29.0.2.1.1 FLUID CONTAMINATION. Hydraulic fluid contamination may be described as any foreign matter whose presence in the fluid may adversely affect the system performance or reliability. Contamination may assume many different forms including liquids, gasses, and solid matter of various composition, sizes and shapes. Solid matter is the type most often found in aircraft hydraulic systems and is generally referred to as particulate contamination. Contamination is always present to some degree, even in new, unused fluid. Contamination should be maintained at a level below that which will adversely affect system operation.

For JetStar aircraft the particulate contamination should not exceed the levels defined as class 6 in NAS 1638 (Cleanliness Requirements of Parts Used in Hydraulic Systems). Water content should not exceed 150 parts per million measured by weight. Chlorine content should not be permitted to exceed 50 parts per million measured by weight.

29.0.2.1.2 MAINTENANCE PRACTICES. Certain precautions, which fall into the categories of good housekeeping and maintenance practices, will assist in controlling contamination. The following is provided as guidance for operators to reduce hydraulic system contamination.

a. Exercise extreme care when working on a hydraulic system in the open, especially under adverse conditions.

b. Exercise extreme caution when working on hydraulic equipment in the vicinity of grinding, blasting, machining, or other contaminant-generating operations. Much of the grit which is harmful cannot be seen with the naked eye.

c. Do not break into hydraulic systems unless absolutely necessary.

d. Use the proper tools for the job.

e. Use only authorized hydraulic fluid, O-rings, lubricants, or filter elements.

f. Use a suitable fluid service unit to dispense hydraulic fluid. Ensure the hydraulic fluid can is clean prior to installation.

g. Keep hydraulic fluid in a closed container at all times.

h. Dispose of all empty hydraulic fluid cans to prevent further use.

i. Seal all hydraulic lines, tubing, hoses, fittings, and components with approved closures.

j. Ensure that quick-disconnect dust covers are reinstalled promptly.

k. Store unused caps and plugs in a clean, covered container.

l. Use wiping cloths to remove exterior contaminants. Use lint-free wiping cloths on surfaces along the fluid path.

m. If possible, have the replacement component on hand for immediate installation upon removal of defective component.

n. If possible, replace filter elements immediately after removal.

o. If possible, fill filter bowl with MIL-H-5606 hydraulic fluid before installing. This minimizes introduction of air into the system.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

p. Avoid use of chlorinated solvents in contact with any hydraulic components. Dry cleaning solvent (Federal Specification P-D-680, Type II) is preferred.

q. Store O-rings, tubing, hoses, fittings, and components in clean packaging.

r. Do not puncture individual packages of O-rings or backup rings until immediately prior to use.

s. Do not use unidentifiable or previously installed O-rings.

t. Seals or backup rings should be replaced with new items when disturbed.

u. When installing O-rings over threaded fittings, use a suitable O-ring installation tool to prevent threads from damaging the O-ring.

v. Decontaminate tubing, hoses, fittings, and components if found or received opened.

w. Decontaminate the system if contamination (including water) is suspected.

x. Ensure that the working area where hydraulic components are repaired, serviced, or stored is clean and free from moisture, metal chips, and other visible contaminants.

29.0.2.2 FLUSHING. Flush hydraulic systems one line at a time, or take advantage of tubing runs where components can be jumpered easily. Refer to the hydraulic schematic diagrams in this manual to determine flow characteristics of the systems. Under less critical conditions, remove all filters in the affected system and replace or clean them as necessary. Reinstall the filters. Operate the system to cause fluid flow. Remove, examine, clean as necessary, and reinstall the filters. Repeat these operations until it is evident that the system is clean.

29.0.2.2.1 FLUSHING HYDRAULIC SYSTEMS AFTER FAILURE OF ENGINE-DRIVEN HYDRAULIC PUMP. It is possible for contamination, resulting from internal failure of an engine-driven hydraulic

29.0.2.2.1 (Cont).

pump, to remain in the system after a normal flushing procedure has been accomplished. To avoid failures, which can be caused by this residue, the affected system should be flushed as follows after internal failure of an engine-driven hydraulic pump:

Note

This procedure is applicable to either the normal (No. 1) system or to the standby (No. 2) *system, following failure of the No. 2 or the* No. 3 engine-driven hydraulic pump, respectively. The only difference is that for the standby (No. 2) system, operated by the No. 3 engine pump, the instructions concerning the auxiliary hydraulic pump and lines should be disregarded.

a. Remove and replace the three flexible lines (pressure, return, and bypass) connected to the affected engine-driven hydraulic pump. It is permissible to remove these lines from the airplane and flush them with an extremely high rate of flow, followed by a visual inspection to determine that all contamination has been removed. However, if there is any doubt that this has been effective, the lines must be replaced by new ones. Remove and replace the quick-disconnect couplings of the pressure and return lines in the nacelle.

b. Thoroughly flush the following lines, remove, flush, and clean the listed components, and inspect visually where possible, for complete removal of contaminants (replace components as required):

Note

The lines may be flushed while installed on the airplane, but the components must be removed and flushed separately.

1. Pressure and supply tubing between nacelle flexible lines and pump inlet and outlet shutoff valves, including the cross-fitting on the suction side.
2. Pump inlet and outlet shutoff valves.
3. Bypass check valve, tee fitting, and tubing connected to supply side.
4. Suction ground quick-disconnect fitting and tubing connecting it to the supply side of the system.

29.0.2.2.1 (Cont).

5. Check valve, tee fitting, and pressure tubing leading to pressure ground connection fitting.
6. Pressure ground quick-disconnect fitting.
7. Pressure line and tee fitting connecting between No. 1 system and auxiliary system pressure sides (only if No. 2 engine pump has failed).
8. Pressure tubing, check valve, and flexible line leading to auxiliary pump pressure port (only if No. 2 engine pump has failed).
9. Flexible line and rigid tubing connected between return port of auxiliary hydraulic pump and tee fitting in No. 1 system suction line (only if No. 2 engine pump has failed).
10. Tubing between auxiliary pump bypass port and tee fitting in No. 1 system bypass line (only if No. 2 engine pump has failed).
11. Examine auxiliary hydraulic pump for contamination. If any is found, remove the pump, and install a new pump (only if No. 2 engine pump has failed).
12. Pressure tubing connecting system pressure side to filter.
13. Examine and clean the pressure filter, as necessary.
14. Supply lines and all fittings between engine-driven pump inlet shutoff valve and system reservoir.
15. Return line and all fittings between system reservoir and return filter.
16. Remove the system reservoir cover, and flush the interior, spraying into all corners between baffle and tank wall and all locations where contaminants may be trapped.
17. Examine and clean or replace the return filter.
18. System bypass line and all fittings, including check valve between bypass connection to suction side of engine-driven pump and bypass connection to return line of system reservoir.

c. Reinstall all components and restore the system to operating condition.

29.0.2.3 CLEANING METAL FILTER ELEMENTS. Ultrasonic cleaning of filter elements gives the best and most reliable results; therefore this method should be used at all times other than emergencies. The filter elements are made from stainless steel mesh and will last indefinitely when properly maintained. Cleaning equipment made by the filter manufacturer, Aircraft Porous Media, Inc., Pinellas Park, Florida, does the most satisfactory cleaning job and is recommended for all normal maintenance procedures. Other ultrasonic cleaning equipment may be used if properly adapted to the cleaning of filter elements. For the correct procedures on cleaning filters, refer to the instructions which accompany the equipment.

Note

In an emergency, clean the filter elements according to the following procedures:

a. Remove the filter element from the filter housing assembly.

b. Apply two gallons per minute of Specification MIL-H-5606 hydraulic fluid in the reverse direction of normal flow.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

c. Immerse the element in a dry cleaning solvent (Federal Specification P-D-680, Type II) until it is saturated.

d. Blow the solvent out of the filter element using 2 to 5 PSI air pressure in the reverse direction of normal flow.

e. Repeat steps (b) through (d); then dry the elements thoroughly. Reinstall the filter elements.

f. If the above method is not available, filter elements can be cleaned by blowing air by mouth through the elements in the reverse direction of normal fluid flow, but elements cleaned in this manner must be cleaned in the normal manner as soon as possible.

Note

Perfectly clean normal and standby system pressure and return filters should allow ten gallons per minute flow at a pressure drop of

29.0.2.3 (Cont).

18 PSI or less. A pressure drop of 50(±5) PSI at a flow of ten gallons per minute indicates that the filter element is clogged and the bypass is relieving hydraulic fluid around the filter element. These values are true only when the fluid is at a temperature of 90 to 100 degrees Fahrenheit.

29.0.2.3.1 INDICATING FILTER. For aircraft 5036 & up, and earlier aircraft which have incorporated a JH1002-2 filter assembly, clogging of the filter element is indicated by a red indicator button, P/NMC230NH090, protruding from the bottom of the filter bowl. The clogged element should be cleaned or replaced with a new element and then the indicator reset by pushing in until it snaps in place. Perform a functional test of the differential pressure indicator each time the filter element is cleaned or removed.

29.0.2.3.2 PRESSURE LINE FILTER. For aircraft 5046 and up, and earlier aircraft which have incorporated a JH1002-3 filter assembly, clogging of the pressure line filter element is indicated by a red indicator button, P/N MC230NH097, protruding from the bottom of the filter bowl. The JH1002-3 filter assembly does not incorporate a bypass valve, therefore cleaning or replacement of the element upon first appearance of the indicator is necessary or partial loss of hydraulic system pressure will be experienced. Perform a functional test of the differential pressure indicator each time the filter element is cleaned or removed.

29.0.2.4 FLARELESS TUBE FITTINGS.

CAUTION

Leaking flareless tube fittings, unlike flared AN fittings, require accurate adjustment of torques. Over-tightening leaking flareless fittings makes the leakage worse and can damage the fittings and tubes beyond repair.

29.0.2.4.1 DESCRIPTION. (See figure 29-1.) Figure 29-1 shows the component parts of a flareless tube union, the assembly sequence, and the completed connection. As the nut engages the threads of the body, the sleeve is forced into the tapered seat of the body. Tightening the nut causes the pilot of the sleeve to contract due to the compression between the sleeve pilot and the tapered seat of the body. The contraction forces the cutting edge of the sleeve to cut a groove in the outer surface of the tube. The sleeve pilot guides the cutting edge and limits the depth of the cut to 0.003 to 0.005 inch. Further tightening of the nut applies pressure to the beveled shoulder of the sleeve head and causes it to clamp tightly on the tube. Fully tightening the nut to the proper torque bows the sleeve at the mid-section, causing a spring action between the body and the nut. This spring action serves as a retaining mechanism for the nut. After presetting the sleeve on the tube, the sleeve is permanently attached to the tube and should not be removed and reused under any circumstances. Disassembly will release some of the bow in the sleeve, but the bow will return upon reassembly and tightening. The sealing points of a flareless union are at the sleeve cut and at the point where the bowed sleeve barrel contacts the tapered seat of the body. Bottoming of the tube end of the reverse angle seat within the body is not intended to serve as a sealing point; its purpose is to support the tube end during assembly and to aid in keeping the cutting edge of the sleeve pressed against the cut to help prevent low pressure seepage leaks.

29.0.2.4.2 MAINTENANCE OF FLARELESS TUBE UNIONS. Certain inherent characteristics of flareless unions cause space problems when connecting plumbing. As the end of the tube must be bottomed against the seat of the body to achieve tube end support, the distance required to withdraw the tube for disconnection is about twice that required for the mating flares of AN unions. This problem can be overcome by providing at least one bend in each tube to allow sufficient springing to permit the tube ends to spread. On large-diameter tubes of hard material, it may be necessary to loosen both ends of the tube to obtain enough working space.

29.0.2.4.3 FLARELESS TUBE UNION TIGHTENING. (See figure 29-2.) Leakage sometimes occurs as a result of overtightening a flareless union. As excessive force is applied to the flareless union, the sleeve cuts into the tube wall until the metal trapped under the pilot of the sleeve prevents additional cutting. Then, either the sleeve shears the metal toward the end of the tube, or the cutting edge of the sleeve breaks; both result in breaking the seal. Extreme overtightening may cause the sleeve barrel to buckle or the tube walls to collapse. In some instances, overtightening may flatten the sleeve against the conical seat of the fitting body and thus cause the system to leak during vibration, although it may seat under static conditions.

29.0.2.4.4 TORQUE-TIGHTENING FLARELESS UNIONS. When proper torque wrenches are available and the nut to be tightened is easily accessible, torque-tighten flareless unions according to the values in figure 29-3. Figure 29-3 applies to flareless tube unions only; it does not apply to hose-end fittings. Hose-end fittings must be position-tightened. Both methods are based on the assumption that the nut turns freely and gives a true bottoming indication. Dirty, scarred, or nicked threads are the most common causes of false bottoming indications, but a cocked tube can also give one. Because of false bottoming indications, the sleeve may not be bowed enough to form a proper seal.

29.0.2.4.5 POSITION-TIGHTENING FLARELESS UNIONS. Position-tighten flareless unions according to these instructions. Following these instruc-

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29.0.2.4.5 (Cont).

tions will result in torques similar to those given in figure 29-3.

a. Use the fingers to bottom the nut and sleeve against the fitting body.

Note

If it is impossible to use the fingers, use a wrench, but be alert for the first signs of bottoming. Final tightening must begin at the point where bottoming begins.

b. From the position obtained in step (a), use a wrench to tighten the nut one-sixth turn (one flat of the hex nut).

c. Pressure-check the union for leaks. If the union leaks, reduce hydraulic pressure to zero; then retighten the nut an additional one-sixth turn. If leakage persists, disconnect the nut and inspect the components for scores, cracks, foreign material, and evidence of improper tightening. Clean, repair, or replace the union as indicated by the findings of the inspection, and then start again at step (a).

CAUTION

Never tighten the nut more than one-third turn. To do so may damage the union beyond repair.

29.0.2.4.6 REPLACEMENT OF FLARELESS TUBE FITTINGS. Spare tubes obtained with nuts and sleeves already attached have the sleeves preset on the tubes. The following procedure is for placing new sleeves and nuts on tubing.

a. Use tubing of the proper length, size, material, and wall thickness. (See figure 29-4.) Tube length should be such that when it is installed it will bottom against the seat of the fitting body. Make sure the tube end where the fitting is to go is squarely cut.

b. Remove all burrs from both the inside and the outside of the cut, but do not chamfer either edge appreciably.

c. Make sure that the tube end is round and that there are no scratches, nicks, or dents where the fitting is to go.

d. Select the proper size presetting tool and clamp it in a vise (see figures 29-5 and 29-6).

Note

If a presetting tool is not available, either a steel or an aluminum fitting body may be used.

CAUTION

Aluminum fitting bodies may be used only once for presetting. Steel fitting bodies may be used no more than five times for presetting. Both may then be used in the final assembly.

e. Slide a nut and then a sleeve onto the tube, making certain that the cutting edge of the sleeve is toward the tube end.

f. Lubricate the threads and the seat of the tool (or fitting body) and the shoulder of the sleeve with Specification MIL-H-5606 hydraulic fluid.

g. Insert a mandrel of the proper size into the tube end. If a mandrel is not available, use any rod with an outside diameter of 0.002 to 0.005 inch less than the inside diameter of the tube. If neither is available, most tubes can be preset without them (see figures 29-4 and 29-7).

h. With the tube end held firmly against the seat of the tool (or fitting body), slowly screw the nut onto the tool threads with a wrench until the tube can no longer be turned with the thumb and forefinger (see figure 29-6). This binding indicates that the sleeve is just starting to grip the tube; this means that the fitting is ready for the final tightening to set the sleeve onto the tube.

Note

If the tube is installed and cannot be rotated, the presetting tool may be rotated instead of the tube to determine when the sleeve starts to grip the tube.

i. Tighten the nut in accordance with the values of figure 29-7.

j. Disassemble and inspect the union for proper preset. (See figure 29-2.)

29.0.2.4.7 INSPECTION AND FINAL ASSEMBLY AFTER PRESETTING FLARELESS UNIONS. (See figure 29-2.) After presetting, uncouple the nut from the threads of the presetting tool or fitting body, and inspect the sleeve and the tube. Visual examination will determine if the sleeve has been properly preset. The following conditions and checks should determine if the sleeve is properly preset on the tube.

a. The sleeve cutting edge should be embedded into the outside diameter of the tube approximately *0.003 to 0.005 inch deep. (This is not a measurable item.)*

b. The sleeve should be slightly bowed.

29.0.2.4.7 (Cont).

c. The sleeve may rotate on the tube, but it should have a longitudinal movement on the tube of no more than 1/64-inch.

d. The sealing surface of the sleeve which contacts the 24-degree angle seat of the fitting body should be smooth, free from scores, and show no longitudinal or circumferential cracks. Crazing cracks in the plating are not detrimental to the safety or function of the fitting.

e. As a final check to determine that the union is preset properly, the tube assembly should be proof-tested at a pressure equal to at least twice the intended working pressure. After inspecting the preset sleeve and tube assembly, make certain that the sleeve, nut, tube, and body seat are free from burrs and foreign material. Before assembly apply a light coat of Specification MIL-H-5606 hydraulic fluid to the threads, the shoulder of the sleeve, and the seat of the union body.

29.0.2.4.8. NORMAL AND STANDBY HYDRAULIC PUMP CONNECTION TORQUE ADJUSTMENTS. After disturbing the torque adjustment of the pump connections, retighten them as follows:

a. Make sure the hose connections are clean and undamaged.

b. Tighten the nuts at both ends of the hose until they bottom. Bottoming is indicated by a sharp rise in the amount of torque required to turn the nuts.

c. After the nuts are bottomed, adjust the torque of each nut by tightening it an additional one-sixth to one-third turn (one to two flats of the hex nut).

29.0.2.5 FLEXIBLE HOSE INSTALLATION. Follow these precautions when installing flexible hose.

a. Never bend the hose to a radius smaller than six (perferable nine) times the diameter of the hose.

b. When wrench-tightening end nuts, hold the hex of the fitting with another wrench.

c. Do not twist the hose during installation. Check for twisted hose by observing the stripe that runs along the length of the hose.

d. Never stretch a hose between two points. Slack equal to 5 to 8 percent of the hoselength should be available for shrinkage.

e. Never allow a hose to rub against adjacent parts.

29.0.2.6 PRESERVATION OF THE HYDRAULIC SYSTEMS DURING PROLONGED INACTIVITY. If the airplane is to be inactive more than 30 days, observe the following precautions to protect the hydraulic systems from deterioration.

a. Check the hydraulic systems for leaks. Make repairs as necessary.

b. Check the hydraulic fluid at the main wheel brake bleed points. If the fluid is clean, replenish the reservoirs with clean Specification MIL-H-5606 hydraulic fluid. If the fluid is dirty or contaminated, continue draining the fluid until clean fluid begins to flow; then replenish the reservoirs with Specification MIL-H-5606 hydraulic fluid.

c. Clean all exposed finished surfaces of actuating rods, hydraulic cylinders, locks, and valves; and coat them with Specification MIL-C-16173 or MIL-C-11796 corrosion preventive oil. (See figure 29-8.)

d. Perform steps (e) through (i) for all hydraulic system accumulators.

e. Dissipate the hydraulic system pressure.

f. Release the air from the accumulators according to the instructions in Chapter 12 and then remove the air filler valve assembly.

g. Pressurize the hydraulic systems to force all air out of the accumulators.

h. Lubricate the air side of the accumulators (inside) using an air-oil spray gun filled with Specification MIL-H-5606 hydraulic fluid.

i. Reinstall the air valves in the accumulators.

j. Perform steps (k) through (m) at 30-day intervals during the inactive period.

k. Check the system for leaks and check the exposed finished surfaces of actuating rods, hydraulic cylinders, locks, and valves for protective coating; and recoat as necessary.

l. Pressurize and operate all operable hydraulic components through three or more cycles.

m. Operate the brake system through 10 complete cycles.

29.0.2.7 DEPRESERVATION OF THE HYDRAULIC SYSTEMS AFTER PROLONGED INACTIVITY. Before returning the hydraulic systems to normal operation, service all accumulators, check the systems for leaks, and check the reservoirs for proper servicing. Refer to Chapter 12 for servicing procedures.

29.0.2.8 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. No special precautions need be taken due to extremes in climate or environment.

29.0.2.9 HYDRAULIC PUMP PRIMING. Following hydraulic pump removal and installation per HOMI Section 71, the following steps should be performed to ensure hydraulic pump priming.

a. If hydraulic system filters have not been removed:

(1) Check hydraulic fluid level and service reservoir as necessary.

(2) Apply air pressure to associated reservoir. This can be accomplished by running the APU, an outboard engine, or using shop air.

(3) Motor affected engine without ignition until a pressure indication appears on the gauge.

CAUTION

Do not exceed starter duty cycle.

b. If hydraulic system filters have been removed and system contamination is suspect:

(1) Follow instructions in paragraph 29.0.2.3.

(2) Apply air pressure to associated reservoir. This can be accomplished by running the APU, an outboard engine, or using shop air.

(3) Loosen the pressure line at the pump outlet and allow fluid to leak through fitting until there is no evidence of air.

(4) Tighten fitting at the pump and service reservoir as necessary.

(5) Motor affected engine without ignition until a pressure indication appears on the gauge.

CAUTION

Do not exceed starter duty cycle.

29.0.3 TROUBLESHOOTING.

29.0.3.1 HYDRAULIC FLUID TRANSFER.

a. **On JetStar airplanes 5001 through 5045, some fluid transfer between hydraulic systems is normal when operating the wing flap systems. For those airplanes, the case drain flow for the leading edge flap drive motor always returns to the normal (No. 1) hydraulic system, and the case drain flow for the trailing edge flap drive motor always returns to the standby (No. 2) hydraulic system regardless of which system is used to power the flaps. The general result of this action is that the normal system reservoir is slightly underserviced while the standby system reservoir is slightly overserviced.**

b. **On JetStar airplanes 5046 and up, there is no normal transfer of fluid between hydraulic systems.**

c. There are numerous factors which may influence the amount of fluid indicated in the reservoirs. The effect of each factor must be taken into account in order to determine whether a fluid transfer has occurred. Some of the major factors are:

(1) The temperature of the hydraulic fluid.

(2) Ambient temperature at present and at the time of the previous servicing.

(3) The attitude of the airplane on the ground, apron slope, tire pressure, etc.

(4) The procedures used for the previous servicing.

(5) Landing gear doors, thrust reverse doors, speed brake, etc., not in positions required by reservoir service instructions.

(6) Fuel load on the airplane.

(7) Passenger and cargo loading.

(8) Condition of accumulators with respect to proper air charge and internal leaks. Fluid volumes should be dissipated when servicing reservoirs.

(9) Air in the system. Air may be introduced into the system from open hydraulic lines, leaking accumulators, discharged emergency landing gear air bottles, etc.

(10) Air blockage in reservoir due to not maintaining the reservoir vent open during servicing.

(11) Air pressure in reservoir due to engine or APU (if installed) operation.

d. There are many normal operating situations which will result in a perceived transfer of hydraulic fluid when no actual transfer has occurred. Small transfers may occasionally take place but should not affect system operation. A transfer sufficient to cause operational problems will be a large quantity of fluid and will be obvious by both the depletion of one reservoir and the overfilling of the other. The first troubleshooting action should be to confirm that a fluid transfer has taken place.

29.0.3.1 HYDRAULIC FLUID TRANSFER (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
CONFIRMED FLUID TRANSFER, SYSTEM 1 TO SYSTEM 2		
Extending wing flaps using Standby system and retracting them using Normal system (airplanes 5001 through 5045 only)	Verify shuttle valve position	Extend and retract flaps using the same hydraulic system.
Defective brake selector valve	Check valve operation.	Replace valve or repair circuit.
Defective steering selector valve (airplane 5046 and up, and airplanes incorporating Service Bulletin (329-214)	Check valve operation.	Replace valve or repair circuit.
No. 1 and No. 2 system wing flap selector valves energized simultaneously	Check power to valve.	Repair circuit as necessary.
Internal leakage through dual shuttle valve (airplanes 5046 and up, and airplanes with Service Bulletin 329-214 incorporated)	Verify valve condition.	Repack shuttle valve.
Internal leakage through Booster Assembly	Verify leakage path.	Replace Booster Assembly.
Internal leakage through Dual-Brake Control valve	Verify leakage path.	Replace Dual-Brake Control valve.
Failure of return line check valve. (Any of several installed)	Verify leakage path.	Replace check valve.
CONFIRMED FLUID TRANSFER, SYSTEM 2 TO SYSTEM 1		
Extending wing flaps using Normal system and retracting them using Standby system (airplanes 5001 through 5045 only)	Verify shuttle valve position.	Extend and retract flaps using the same hydraulic system.
Defective brake selector valve	Check valve operation.	Replace valve or repair circuit.
Defective steering selector valve (airplanes 5046 and up, and airplanes incorporating Service Bulletin 329-214)	Check valve operation.	Replace valve or repair circuit.
No. 1 and No. 2 system wing flap selector valves energized simultaneously	Check power to valve.	Repair circuit as necessary.

29.0.3.1 HYDRAULIC FLUID TRANSFER (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
CONFIRMED FLUID TRANSFER, SYSTEM 2 TO SYSTEM 1 (continued)		
Use of thrust reverse while using No. 2 system braking (airplanes 5001 through 5046 not having Service Bulletin 329-166 incorporated)	Check shuttle valve operation.	Incorporate Service Bulletin 329-166.
Transfer through wing flap plumbing while Normal system shutdown (airplanes 5046 and up, not having Service Bulletin 329-283 incorporated)	Verify transfer path.	Incorporate Service Bulletin 329-283.
Internal leakage through dual shuttle valve (airplanes 5046 and up, and airplanes with Service Bulletin 329-214 incorporated)	Verify valve condition.	Repack shuttle valve.
Internal leakage through Booster Assembly	Verify leakage path.	Replace Booster Assembly.
Internal leakage through Dual-Brake Control valve	Verify leakage path.	Replace Dual-Brake Control valve.
Failure of return line check valve (any of several installed)	Verify leakage path.	Replace check valve.

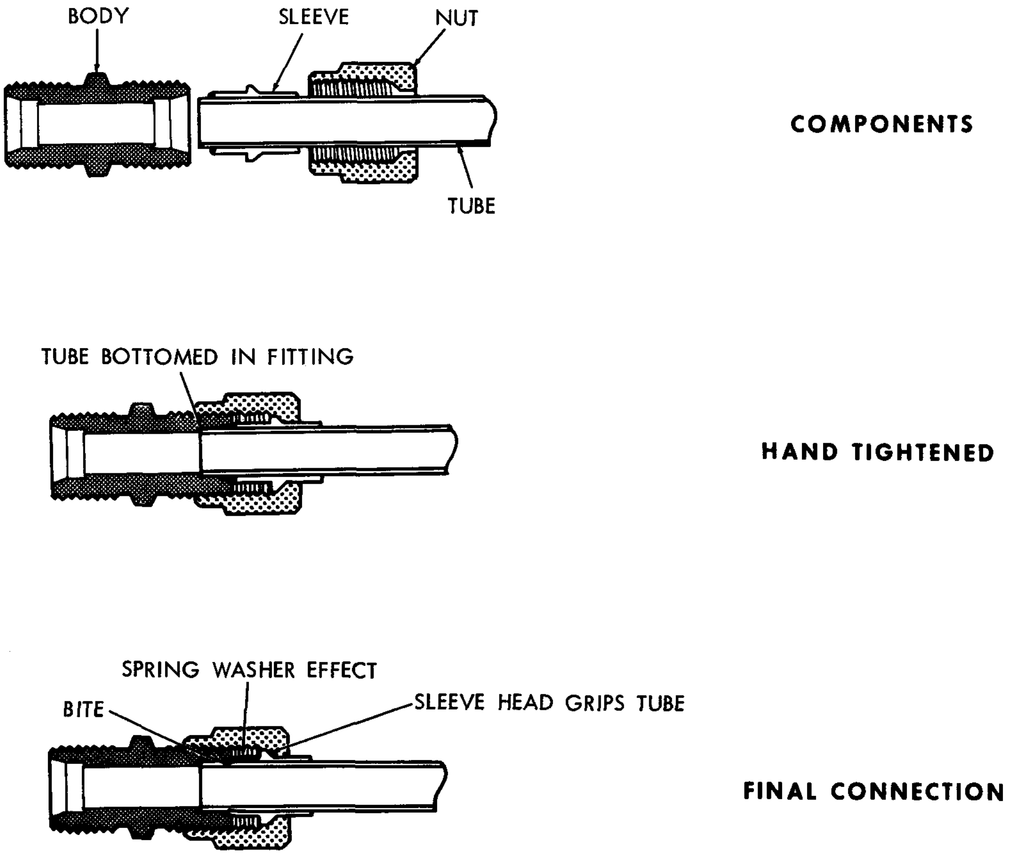
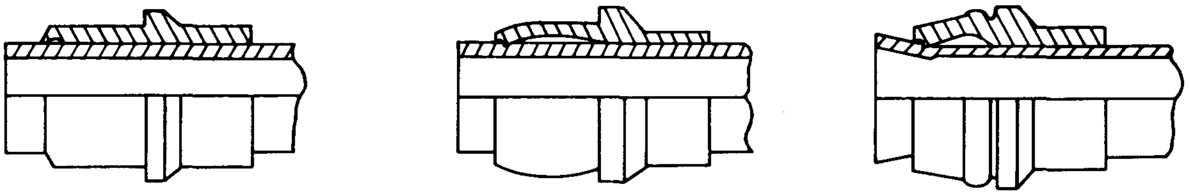


Figure 29-1. Flareless Fitting Components and Assembly Sequence

JS-2-2-2903



UNDERTIGHTENED: LITTLE OR NO CUT ON TUBE, PILOT FORCED CLOSE TO TUBE, NO BOW OR SPRING ACTION.

CORRECT PRESET: GOOD CUT ON TUBE, PILOT CLOSE TO TUBE, GOOD BOW FOR SPRING ACTION.

OVERTIGHTENED: SLEEVE CRUSHED, NO SPRING ACTION, TUBE FLARED OR MUSHROOMED.

JS-2-2-2904

Figure 29-2. Preset Flareless Sleeves

TUBE MATERIAL	TUBE SIZE IN INCHES (O.D. X WALL THICKNESS)	TORQUE VALUE (IN INCH-POUNDS) △1
304 1/8H Steel	1/4 X .016	110-140
	1/4 X .020	110-140
	1/4 X .035	120-150
	3/8 X .020	140-170
	3/8 X .028	190-220
	1/2 X .035	340-370
	5/8 X .042	450-490
6061-T6 Al Alloy	1/4 X .020	80-90
	3/8 X .028	90-110
	1/2 X .035	290-320
	5/8 X .035	350-440
	3/4 X .035	380-470
	1 X .035	700-740

NOTE

△1 THESE VALUES SHOWN ON ALL TUBE ASSEMBLIES BY DECAL:
FOR EXAMPLE T-110-140.

JS-2-2-2916

Figure 29-3. Torque Values for Tightening Flareless Tube Fittings

WITH MANDRELS		WITHOUT MANDRELS		
TUBE O.D. (IN INCHES)	WALL THICKNESS (ALL MATERIALS)	TUBE O.D. (IN INCHES)	304 1/8H STEEL	6061-T6 AL ALLOY
1/4	.016	1/4	.020	.028
3/8	.020	3/8	.025	.025
1/2	.025	1/2	.028	.028
5/8	.028	5/8	.035	.035
3/4	.032	3/4	.035	.035
1	.032	1	.042	.042

JS-2-2-2917

Figure 29-4. Minimum Tube-Wall Thickness for Presetting

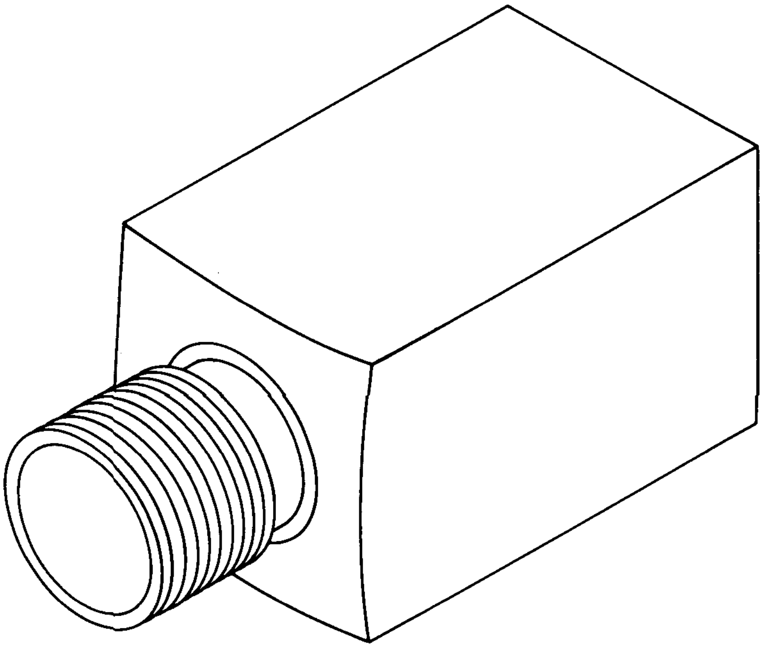
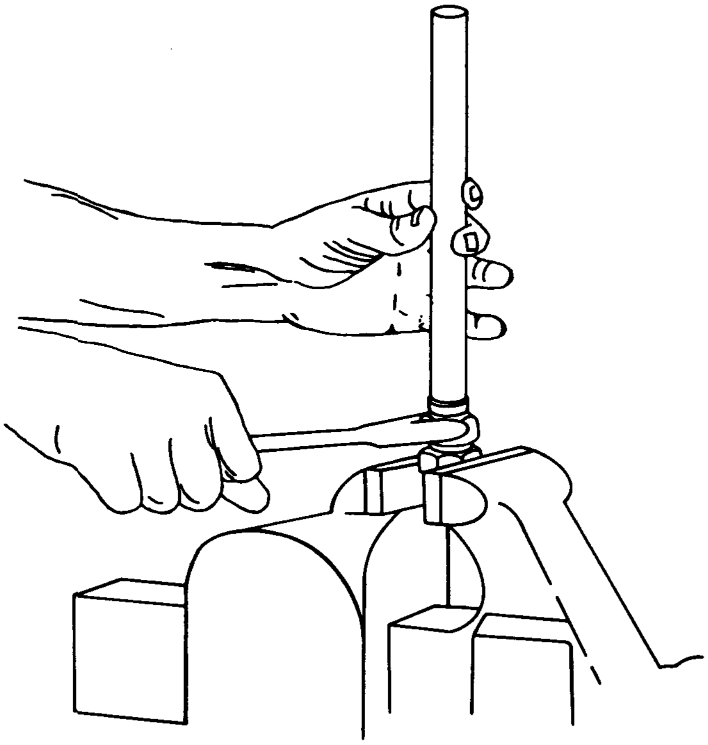


Figure 29-5. Presetting Tool

JS-2-2-2901



JS-2-2-2902

Figure 29-6. Presetting by Hand

TUBE MATERIAL	TUBE SIZE IN INCHES (O.D. BY WALL THICKNESS)	NUMBER OF NUT TURNS WHEN USING MANDRELS	NUMBER OF NUT TURNS WHEN NOT USING MANDRELS
304 1/8H Steel	1/4 X .016	1	Must use mandrel.
	1/4 X .020	1	1 1/6
	1/4 X .035	1	1
	3/8 X .020	5/6	Must use mandrel.
	3/8 X .028	5/6	1 1/6
	1/2 X .035	5/6	1 1/6
	5/8 X .042	2/3	1 1/6
6061-T6 Al Alloy	1/4 X .020	1	Must use mandrel.
	3/8 X .028	5/6	1 1/6
	1/2 X .035	5/6	1 1/6
	5/8 X .035	2/3	1 1/6
	3/4 X .035	2/3	1 1/6
	1 X .035	2/3	Must use mandrel.

JS-2-2-2915

Figure 29-7. Turn Values for Presetting by Hand

MILITARY STANDARD DESIGNATION	MANUFACTURERS' DESIGNATION	MANUFACTURERS' NAMES AND ADDRESSES
SPECIFICATION MIL-C-11796, PETROLATUM, HOT APPLICATION, CLASS 3	BRAYCOTE 248	BRAY OIL COMPANY 1925 MARIANNA AVE. LOS ANGELES, CA 90032
	CODE 312D	SOUTHWEST GREASE & OIL COMPANY, INC. 3148 ROANOKE RD. KANSAS CITY, MO. 64111
	COSMOLINE #1062	E. F. HOUGHTON & CO. MADISON AND VAN BUREN AVE. VALLEY FORGE, PA 19482
	GULF NO-RUST NO. 7	GULF REFINING COMPANY P. O. BOX 7409 PHILADELPHIA, PA
	KENDEX 7012	KENDALL REFINING COMPANY BRADFORD, PA
	L-23B	FRANKLIN OIL CORPORATION 40 SOUTH PARK ST. BEDFORD, OH 44146
	NOKORODE 963A	TOSCO CORPORATION EL DORADO, AR 71730
	NOX-RUST 509	NOX-RUST DIVISION DAUBERT CHEMICAL COMPANY 4700 SOUTH CENTRAL AVE CHICAGO, IL 60638
	PETROTECT P-250	PENRECO P. O. BOX 671, BUTLER, PA 16001
	ROYCOTE 248	ROYAL LUBRICANTS COMPANY RIVER ROAD E. HANOVER, NJ 07936
	TECTYL 437	ASHLAND PETROLEUM COMPANY P. O. BOX 391 ASHLAND, KY 41104

JS-1-2-X0/2-2908-1

Figure 29-8. Corrosion Preventive Compounds (Sheet 1 of 2)

MILITARY STANDARD DESIGNATION	MANUFACTURERS' DESIGNATION	MANUFACTURERS' NAMES AND ADRESSES
SPECIFICATION MIL-C-16173, SOLVENT CUTBACK, COLD APPLICATION GRADE 2	BRAYCOTE 137	BRAY OIL COMPANY 1925 MARIANNA AVE. LOS ANGELES, CA 90032
	NOX-RUST 207	NOX-RUST DIVISION DAUBERT CHEMICAL COMPANY 1200 JORIE BLVD. OAKBROOK, IL 60521
	PETROTECT 2	PENRECO P. O. BOX 671 BUTLER, PA 16001
	RUST PREVENTIVE 1612A	FRANKLIN OIL COMPANY BEDFORD, OH 44146
	STEELGARD MS10	HARRY MILLER CORPORATION PHILADELPHIA, PA 19140
	VALVOLINE TECTYL 502-C	ASHLAND PETROLEUM COMPANY P. O. BOX 391 ASHLAND, KY 41101
	VISCONRUST NO. 1602	VISCOSITY OIL COMPANY 3200 S. WESTERN AVE., CHICAGO, IL 60608
	PZA	STEVENS INDUSTRIES, P. O. BOX 8, -39 AVE. C, BAYONNE, NJ 07002
	ROYCO 137R	ROYAL LUBRICANTS COMPANY RIVER ROAD E. HANOVER, NJ 07936

JS-I-2-X0/2-2908-2

Figure 29-8. Corrosion Preventive Compounds (Sheet 2 of 2)

Figure 29-9. Hydraulic System Schematic Diagram (Airplanes 5001 thru 5040)

Figure 29-9A. Hydraulic System Schematic Diagram (Airplanes 5041 thru 5045)

Figure 29-10. Hydraulic System Schematic Diagram (Airplanes 5046 thru 5092 and Airplanes 5094 thru 5096)
Changed 28 October 1988

Figure 29-10A. Hydraulic System Schematic Diagram (Airplane 5093 and Airplanes 5097 and Up)

SECTION 29.1
NORMAL (NO. 1) AND AUXILIARY (NO. 3) HYDRAULIC SYSTEMS

Contents ..... Page

DESCRIPTION AND OPERATION 29-20
- General 29-20
- Normal (No. 1) Hydraulic System 29-20
- Auxiliary (No. 3) Hydraulic System 29-20
- Auxiliary Hydraulic System Control Circuits 29-20
  - Auxiliary Hydraulic Pump Manual Control Circuits 29-20A
  - Auxiliary Pump Control Through the Landing Gear Circuit (Airplanes 5001 through 5040) 29-20A
  - Auxiliary Pump Control Through the Landing Gear Circuit (Airplanes 5041 and Up) 29-21
  - Auxiliary Pump Control Through the Wing-Flap-Down Circuit 29-21
- Reservoir Pressurization System 29-21
- Reservoir (Airplanes 5001 through 5045) 29-21
- Reservoir (Airplanes 5046 through 5092) 29-21
- Reservoir (Airplanes 5093 and Up) 29-21
- Servicing Indicator Glass 29-21
- Normal System Pump Inlet Shutoff Valve 29-21
- Engine-Driven Hydraulic Pump 29-22
- Electric Motor-Driven Hydraulic Pump 29-22
- Normal System Pump Outlet Shutoff Valve 29-22
- External Hydraulic Power Connections 29-22
- Hydraulic Filters 29-22
- Normal System Accumulator and Air Filler Valve 29-22
- Normal System Pressure Transmitter 29-22
- Hydraulic Dual-Pressure Indicator 29-23
- System High-Pressure Relief Valve 29-23
- Low Pressure Warning Switch (Airplanes 5086 and Up) 29-23
- Check Valves 29-23

OPERATIONAL CHECKOUT 29-23
- Normal (No. 1) Hydraulic System 29-23
- Auxiliary (No. 3) Hydraulic System 29-24
- Reservoir Pressurization System 29-24

TROUBLESHOOTING 29-24A
- Normal (No. 1) Hydraulic System 29-24A
- Auxiliary (No. 3) Hydraulic System 29-26
- Reservoir Pressurization System 29-27
- Flow Requirements 29-28

BLEEDING PROCEDURES 29-28
- Normal and Auxiliary Hydraulic Systems 29-28

FIGURES

- 29-11. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5001 through 5045) 29-29
- 29-12. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5046 through 5092) 29-30
- 29-12A. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5093 and Up) 29-30A
- 29-12B. Flow Requirement Summary - Normal System (No. 1) 29-30B
- 29-12C. Engine Driven Pump Flow vs Pressure (Theoretical) JH1000-1,-2 Pumps 29-30C
- 29-12D. Pump Speed vs Flow vs Engine Speed 29-30C
- 29-13. Normal and Auxiliary Hydraulic System Schematic Diagram Airplanes 5001 through 5045) 29-31

FIGURES

29-14. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5046 through 5085) 29-32
29-14A. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5086 through 5092) 29-32A
29-14B. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5093 and Up) 29-32C
29-14C. Operation of "Negative G" Reservoir 29-32E
29-15. Auxiliary Hydraulic Pump Control Schematic Diagram (Airplanes 5001 through 5085) 29-33
29-15A. Auxiliary Hydraulic Pump Control Schematic Diagram (Airplanes 5086 and Up) 29-34

29.1.1 DESCRIPTION AND OPERATION.

29.1.1.1 GENERAL. (See figures 29-9, 29-10, 29-13 and 29-14.) A normal (No. 1) hydraulic system and an auxiliary (No. 3) hydraulic system power the speed brake, thrust reversers, landing gear, wheel brakes, nose wheel steering, one side of the aileron booster assembly, one side of the elevator booster assembly, and the leading and trailing edge flaps. The auxiliary hydraulic pump uses the reservoir, accumulator, and system lines of the normal system; and automatically supplements normal system power during landing-gear-up and normal trailing edge flap-down actuation. The auxiliary hydraulic pump can be manually selected to supplement normal system power, or to power any normal system component independently of the normal system pump.

29.1.1.2 NORMAL (NO. 1) HYDRAULIC SYSTEM. (See figures 29-11 and 29-12.) The normal hydraulic system derives its power from an engine-driven hydraulic pump on engine No. 2. This 3,000-PSI system contains a pressurized reservoir, an engine-driven pump, a pump inlet shutoff valve, a pump outlet shutoff valve, external hydraulic connections, a pressure filter, an accumulator, a pressure transmitter and indicator, a low-pressure warning switch (airplane 5086 and up), a high-pressure relief valve, a return filter, and check valves to maintain directional flow within the system.

a. Fluid from the normal system reservoir flows through the pump inlet shutoff valve to the engine-driven pump on the No. 2 engine. This fluid leaves the pump under pressure and travels through the outlet shutoff valve, a check valve, and on to the pressure filter, the pressure transmitter, the low-pressure warning switch (airplane 5086 and up), the system high-pressure relief valve, the speed brake system, one side of the aileron booster assembly, one side of the elevator booster assembly, the thrust-reverser system, the landing gear and door system, the nose steering system, and the brake system via anti-skid valves. Fluid returning from these systems passes through the return filter and then dumps into the normal system reservoir. Check valves in the pressure lines of the normal system pump and auxiliary system pump prevent pressure from one pump entering the other. A run-around circuit insures pump lubrication when the fire emergency control handle is pulled to close the pump shutoff valves.

b. On airplanes 5001 through 5040, the following accumulators are pressurized by the normal hydraulic system: the thrust reverser accumulator, the brake accumulator, and the normal hydraulic system accumulator. On airplanes 5041 by the normal hydraulic system, and two additional main gear uplock accumulators are also pressurized by this system.

29.1.1.3 AUXILIARY (NO. 3) HYDRAULIC SYSTEM. (See figures 29-11 and 29-12.) The auxiliary hydraulic system is not an independent system; it has no reservoir or system lines of its own, but uses those of the normal system. However, for airplanes 5046 thru 5092, an auxiliary reservoir is installed to provide a reserve supply of hydraulic fluid. A complete description of this reservoir will be found in paragraph 29.1.1.7. The auxiliary hydraulic system is powered by an electric motor-driven hydraulic pump connected hydraulically in parallel with the normal system pump. The auxiliary hydraulic pump normally energizes to supplement normal system power during landing-gear-up and normal trailing edge flap-down actuation, but it can be manually selected to power normal system components independently of the normal system pump.

CAUTION

When available, use external hydraulic power for ground checkout procedures. Continuous operation of the auxiliary hydraulic pump is not recommended, and to operate it in this manner will not only result in excessive pump wear, but will also deplete the charge on the airplane batteries unless external electrical power is used.

29.1.1.4 AUXILIARY HYDRAULIC SYSTEM CONTROL CIRCUITS. (See figure 29-15.) The auxiliary system pump motor automatically energizes during gear-up and normal trailing edge flap-down actuation to supplement normal system pump power. Manual control of the pump is provided by a two-position ("ON," "OFF") AUX HYD SYS-3 switch on the copilot's side of the main instrument panel.

29.1.1.4.1 AUXILIARY HYDRAULIC PUMP MANUAL CONTROL CIRCUITS. (See figure 29-15.) Placing the AUX HYD SYS-3 switch in the "ON" position closes a circuit from the hydraulic pump control circuit breaker on the 28-volt, DC essential bus through the AUX HYD SYS-3 switch to energize the auxiliary hydraulic system No. 3 relay. Energizing this relay supplies power from the 28-volt, DC essential bus, through a current limiter in the electrical power shield, and through the auxiliary hydraulic system No. 3 relay to energize the auxiliary pump motor. Placing the AUX HYD SYS-3 switch in the "OFF" position de-energizes the auxiliary hydraulic system No. 3 relay to break the power supply circuit to the pump motor.

29.1.1.4.2 AUXILIARY PUMP CONTROL THROUGH THE LANDING GEAR CIRCUIT (AIRPLANES 5001 THROUGH 5040). (See figure 29-15.) Placing the landing gear control handle in the "LG UP" position energizes the landing gear auxiliary hydraulic pump relay to route power from the hydraulic pump control circuit breaker to both main gear inboard door uplock switches. As the main gear inboard doors open at the beginning of the gear-up cycle, either or both of the inboard door uplock switches route power to the auxiliary hydraulic system No. 3 relay. Energizing this relay admits power from the hydraulic pump current limiter to energize the auxiliary hydraulic pump motor. This relay remains energized, and the auxiliary pump continues to run until the gear is up and the inboard doors are closed.

29.1.1.4.3 AUXILIARY PUMP CONTROL THROUGH THE LANDING GEAR CIRCUIT (AIRPLANES 5041 AND UP). (See figure 29-15.) Placing the landing gear control handle in the "LG UP" position closes a circuit to the main gear inboard door uplock switches. As the doors open to start the gear-up cycle, either or both of these switches close a circuit to the auxiliary hydraulic system No. 3 relay to apply power from the hydraulic pump current limiter to the auxiliary pump motor. The relay remains energized and the pump motor runs until the gear is up and the doors are closed.

29.1.1.4.4 AUXILIARY PUMP CONTROL THROUGH THE WING-FLAP-DOWN CIRCUIT. (See figure 29-15.) Placing the flap control lever in the trailing edge "DN" position with the FLAP & BR HYD SYS switch in the "NORM-1" position, closes a circuit through the trailing edge flap followup control switch, through the trailing edge flap followup switch, through the flap disconnect reset relay, through the wing flap selector relay, and through the trailing edge flap safety relay to energize the down solenoid of the trailing edge flap selector valve and the flap boost disconnect relay. As the trailing edge flaps extend, the flap boost disconnect relay routes power from the hydraulic pump control circuit breaker to energize the auxiliary hydraulic system No. 3 relay to start the auxiliary hydraulic pump motor. As the trailing edge flaps reach the selected position, the trailing edge flap followup switch breaks the circuit to stop pump operation.

29.1.1.5 RESERVOIR PRESSURIZATION SYSTEM. (See figures 29-13 and 29-14.) The normal and standby hydraulic system reservoirs are pressurized by engine bleed air to prevent pump cavitation. This bleed air passes through a one-way restrictor, an air filter, a check valve, and a pressure regulator before entering the reservoirs. The air pressure regulator maintains the pressure in the reservoirs at 8.5 (±0.6) PSI, and also vents excessive air pressure overboard. A check valve in the reservoir pressurization system prevents hydraulic fluid from backing up into the engine bleed air system. A bleed nut is installed in a tee in the air pressure line over the normal system reservoir to facilitate bleeding reservoir pressure during servicing operations. This bleed nut can be removed to provide an attaching point for a test gage when checking out the reservoir pressurization system. On airplanes 5001 through 5045, because of case return flow from the wing flap drive motors, the standby system reservoir receives more return fluid than the normal system reservoir during normal operation of the system. When this excess fluid overfills the standby system reservoir, the overflow passes through the air line connecting the reservoirs and dumps into the normal system reservoir. On airplanes 5046 and up, the case return flow is routed to the reservoir of the respective system, and does not overfill the standby reservoir.

29.1.1.6 RESERVOIR. (Airplanes 5001 through 5045.) The normal system reservoir is a cylindrical tank with a servicing indicator glass, a pressure port, a drain port, a return port, and a supply port. Some reservoirs have two indicator glasses, but only the top glass is used. The drain port of the reservoir is connected to a tube which connects to a spring-loaded-closed drain valve near the left hydraulic panel drip pan. A scupper below the filler neck catches spilled fluid and drains it overboard, via the drip pan. For reservoir capacity, refer to Chapter 12.

29.1.1.7 RESERVOIR. (Airplanes 5046 thru 5092.) In addition to the reservoir described in paragraph 29.1.1.6, an auxiliary reservoir, connected in series with the normal reservoir, is installed. The auxiliary reservoir is located aft and below the normal reservoir, and is connected to it by a 1-inch-diameter tube. The normal system return is to the auxiliary reservoir, and the auxiliary pump supply is from the auxiliary reservoir. The auxiliary reservoir is drained by means of the same valve as the normal reservoir and is filled through the normal reservoir filler. This auxiliary reservoir is designed to hold a reserve of hydraulic fluid for the auxiliary pump in the event of fluid loss from the normal reservoir.

29.1.1.8 RESERVOIR. (Airplanes 5093 and Up.) (See figure 29-14C.) The normal system reservoir is a cylindrical tank with a tube-type fluid level gage and ports for: pressurization air, return flow, engine pump supply, auxiliary pump supply, and drain. Internally the reservoir is divided into upper and lower volumes by a baffle assembly. The upper volume provides the space required for displacement fluid and expansion. The baffle assembly restrains fluid in the lower volume to assure continuous pump supply under negative "G" conditions. A standpipe for the engine pump supply isolates a reserve volume for auxiliary pump operation. The referenced figure depicts the various phases of operation.

29.1.1.9 SERVICING INDICATOR GLASS. The servicing indicator glass is a small, round glass with a yellow dot in the center. As this glass is near the top of the reservoir, fluid level is visible only when the reservoir is close to being properly serviced.

Note

> On reservoirs having two indicator glasses use the upper glass. On Airplanes 5046 thru 5092 the aux reservoir indicator glass is for reference only, not servicing.

29.1.1.10 NORMAL SYSTEM PUMP INLET SHUTOFF VALVE. A reversible, motor-operated shutoff valve in the pump suction line shuts off fluid flow to the normal system pump when the fire emergency control handle for engine No. 2 is pulled. Returning the fire emergency control handle for engine No. 2 to the normal (in) position positions the valve to allow fluid flow to the pump. This valve is in the aft compartment, on the tray over the left hydraulic panel.

29.1.1.11 ENGINE-DRIVEN HYDRAULIC PUMP. The normal engine-driven hydraulic pump is a variable-delivery, piston-type pump with a rated output of 3,000 PSI at no flow and 2,525 PSI at 8.6 gallons per minute. It is on an accessory drive pad on the lower right side of the No. 2 engine accessory section.

29.1.1.12 ELECTRIC MOTOR-DRIVEN HYDRAULIC PUMP. The auxiliary hydraulic pump is a variable delivery, electric motor-driven pump, rated at 2,900 PSI delivery pressure. On airplanes 5001 through 5085, the capacity of the pump is 6 gallons per minute at 2,100 PSI. The duty cycle is 30 seconds at 2,100 PSI, and 6 gallons per minute; 29 1/2 minutes at 2,800 PSI, and 1 1/2 gallons per minute: and then back to 30 seconds at 2,100 PSI and 6 gallons per minute: and repeat. On airplanes 5086 and up, the capacity of the pump is 4 1/2 gallons per minute at 1,900 PSI. The duty cycle is 30 seconds at 1,900 PSI and 4 1/2 gallons per minute: 29 1/2 minutes at 2,600 PSI and 1 1/2 gallons per minute: then back to 30 seconds at 1,900 PSI and 4 1/2 gallons per minute; and repeat.

Note

This latter pump is the preferred spare for airplanes 5001 through 5085.

CAUTION

Use external hydraulic power for ground checkouts of hydraulic components when available. Extended continuous operation of the auxiliary hydraulic pump is not recommended.

29.1.1.13 NORMAL SYSTEM PUMP OUTLET SHUTOFF VALVE. A single-solenoid shutoff valve energizes to shut off flow from the normal system pump when the fire emergency control handle for engine No. 2 is pulled. Returning the fire emergency control handle for engine No. 2 to the normal (in) position de-energizes the valve and allows it to shift back to the open position to allow fluid flow from the pump.

29.1.1.14 EXTERNAL HYDRAULIC POWER CONNECTIONS. External hydraulic power connections for the normal hydraulic system are located aft of the left wing-to-fuselage fairing. There are two connections: one is labeled SUCTION and the other is labeled PRESSURE. When not in use, these connections are capped, and a door closes over them for streamlining. For instructions on connecting external hydraulic power, refer to Chapter 12.

29.1.1.15 HYDRAULIC FILTERS. There are two hydraulic filters in the normal hydraulic system: a pressure filter and a return filter. The pressure filter removes contaminants from the fluid leaving the pump before it reaches the subsystems; the return filter removes contaminants from the fluid after it leaves the subsystems and before it dumps into the normal system reservoir. Both filters are of the in-line type with replaceable filter elements. Arrows on the filter housings indicate the direction of fluid flow. On airplanes 5001 through 5045, an internal bypass valve in each filter assembly opens to relieve fluid around clogged filter elements when the differential pressure reaches 50 (±5) PSI. The bypass valves reseat at 32 PSI minimum differential pressure. For airplanes 5046 and up, the pressure filter does not have an internal bypass valve. This is to prevent contamination from a possible pump failure from entering the system. For airplanes 5036 and up, filter pressure drop indicators are installed in the pressure and return filters. Each indicator consists of a mechanism which extends a red button approximately 1/5 inch out of the bottom of the filter assembly when differential pressure across the filter element exceeds 35 (±5) PSI. Once extended, the indicator must be reset manually. Elements should be cleaned or replaced when buttons extend. These indicators do not operate at temperatures below 30 (±20) degrees Fahrenheit. For airplanes 5046 and up, the differential pressure indicator on the pressure filter will extend when the differential pressure across the filter element exceeds 100 (±15) PSI.

Note

Return filter assemblies with differential pressure indicators are available as preferred spares. Also, the no bypass pressure filter assembly is available as a preferred spare. (If the no bypass pressure filter is installed, new nameplates should be installed. Service Bulletin 329-84 provides the necessary parts and information for airplanes 5001 through 5045.)

CAUTION

Do not install a no-bypass filter in the hydraulic return line.

29.1.1.16 NORMAL SYSTEM ACCUMULATOR AND AIR FILLER VALVE. A 25 cubic-inch-capacity, piston-type accumulator minimizes surges in the normal hydraulic system. It carries a 1,700-PSI precharge of dehydrated air or nitrogen. An air filler valve facilitates accumulator servicing. Airplanes 5066 and up are equipped with accumulator pressure gauges to facilitate checking of precharge pressure. The normal system accumulator is attached to the left hydraulic panel in the aft compartment. (Refer to paragraph 29.0.1.8.)

29.1.1.17 NORMAL SYSTEM PRESSURE TRANSMITTER. A normal system pressure transmitter converts hydraulic pressure into electrical signals

29.1.1.17 (Cont).

and sends these signals to the dual-pressure indicator on the main instrument panel. These signals position the No. 1 pointer to show the amount of pressure in the normal hydraulic system. A snubber assembly is installed in the pressure supply line upstream of the pressure transmitter to prevent damage to the transmitter assembly. The pressure transmitter is mounted on the aft side of the station 608 frame at approx WL92 on the left hand side of the aft equipment compartment.

29.1.1.18 HYDRAULIC DUAL-PRESSURE INDICATOR. A dual-pressure indicator on the copilot's side of the main instrument panel shows normal, and/or auxiliary, and standby hydraulic pressures. The No. 1 pointer shows normal and/or auxiliary pressure, and the No. 2 pointer shows standby hydraulic system pressure. This indicator receives its signals from the pressure transmitters in the normal and standby system pressure lines.

29.1.1.19 SYSTEM HIGH-PRESSURE RELIEF VALVE. A system high pressure relief valve cracks at 3,500 (±50) PSI to relieve excessive hydraulic pressure back to the normal system reservoir. This valve is on the left hydraulic panel in the aft compartment.

29.1.1.20 LOW PRESSURE WARNING SWITCH (AIRPLANES 5086 AND UP). One low-pressure warning switch in the normal and/or auxiliary pressure line and one in the stand-by pressure line control a single HYD PRESS LOW warning light on the annunciator warning panel. Low pressure in any system will cause the warning light to illuminate; reference to the hydraulic dual pressure indicator will show which system pressure is low. Each switch is set to make the warning light go on at 1,200 PSI minimum and go out at 1,800 PSI maximum.

29.1.1.21 CHECK VALVES. Several check valve assemblies are installed in the normal hydraulic system plumbing, positioned so as to prevent undesirable hydraulic fluid flow.

a. Check valves are installed in the pressure lines from the engine driven hydraulic pump and from the electric motor driven hydraulic pump. The electric motor driven pump does not operate continuously and it is sometimes necessary to shut down the engine driven pump. These check valves prevent entry of pressure from the operating pump into the inactive pump

b. Airplanes 5057 and up have a check valve installed in the case drain line from the electric motor driven pump to protect the pump housing from entry of solid contaminants and debris.

c. Two check valves are installed in the case drain plumbing from the engine driven pump. These check valves provide a fluid path for cooling flow when the No. 2 engine is shut down and also protect the pump housing from entry of solid contaminants and debris.

29.1.2 OPERATIONAL CHECKOUT.

29.1.2.1 NORMAL (NO. 1) HYDRAULIC SYSTEM.

a. Start engine No. 2 according to the instructions in Chapter 72.

b. At engine idle, observe the dual-pressure indicator on the copilot's side of the main instrument panel.
RESULT: With no hydraulic sub-systems operating, the pointer labeled "1" should show 3,000 (±150) PSI.

c. Shut down the engine.

d. Check the operation of the pump shutoff valves according to the following instructions:

1. Connect an external source of hydraulic power to the normal system pump suction and pressure lines between the pump on engine No. 2 and the shutoff valves in the aft compartment.

CAUTION

Do not connect external hydraulic power to the ground test connections for this check. This could result in a ruptured reservoir or other damage to the system.

2. Connect 28-volt, DC power to the airplane, and energize the 26-volt, AC main bus.

3. With no hydraulic pressure on the system, pull the fire emergency control handle for engine No. 2, and observe the pump inlet shutoff valve on the left hydraulic panel.
RESULT: The indicator on the pump inlet shutoff valve should indicate the closed position.

4. Using fluid from the reservoir of the hydraulic power source, apply 3,000 PSI hydraulic pressure to the pressure line, and observe the dual-pressure indicator on the main instrument panel.
RESULT: The No. 1 pointer of the dual-pressure indicator should not be affected. This indicates that the pump outlet shutoff valve is closed.

5. Reduce hydraulic pressure to zero, and return the fire emergency control handle to the normal (in) position.

6. Slowly apply pressure to the pressure line of the normal hydraulic system, and observe the No. 1 pointer of the dual-pressure indicator.
RESULT: The pointer should move, showing that the pump outlet valve is open.

7. Check the position of the pump inlet shutoff valve.
RESULT: The pump inlet shutoff valve should be in the open position.

8. Reduce hydraulic pressure to zero, and disconnect the power sources from the airplane.

29.1.2.2 AUXILIARY (NO. 3) HYDRAULIC SYSTEM.

Note

This checkout does not test the automatic pump control circuits concerned with landing gear and wing flap operation. For these tests, refer to the landing gear operational checkout in Chapter 32 of this manual, and the wing flap operational checkout in Chapter 27.

a. Apply 28-volt DC power to the service receptacles of the airplane, and energize the 28-volt essential DC bus and the 26-volt main AC bus.

b. Reduce hydraulic pressure in the normal hydraulic system to zero by operating the wheel brakes.

CAUTION

Before depleting the wheel brake pressure, make sure the wheel chocks are in place.

c. Place the AUX HYD SYS-3 switch, on the copilot's side of the main instrument panel, in the "ON" position, and observe the pointer labeled "1" on the dual-pressure indicator.
RESULT: With no hydraulic sub-systems operating, the pointer labeled "1" should indicate 2,900 (±200) PSI. Initial surges up to 3,450 PSI are allowable.

d. Return the AUX HYD SYS-3 switch to the "OFF" position upon completing this test.

29.1.2.3 RESERVOIR PRESSURIZATION SYSTEM.

a. Connect an air pressure gage capable of measuring up to 10 PSI to the tee in the air line over the normal system reservoir.

b. Disconnect the return line from the outlet port of the normal system return filter. This filter is on the left hydraulic panel. Plug the disconnected lines.

c. Disconnect the return line from the outlet port of the standby system return filter. This filter is on the right hydraulic panel. Plug the disconnected lines.

d. Disconnect the hydraulic supply line to the auxiliary hydraulic pump, and plug the line.

e. Disconnect the hydraulic supply lines downstream of the inlet shutoff valves for the normal and standby system pumps. Plug the disconnected lines.

f. Disconnect the reservoir pressurization system at a point between the bleed air system and the restrictor in the pressurization system; and then connect a source of air pressure to the pressurization line. The air pressure source should be capable of delivering approximately 80 PSI.

g. Make sure that the reservoir drain lines and valves are not leaking.

h. Make sure that the reservoir filler caps are tight and not leaking.

i. Make sure that the dust caps are properly installed on the external hydraulic connections for both the normal and the standby hydraulic systems.

j. Apply 18 PSI air pressure to the reservoir pressurization system, and observe the pressure gage in the tee over the normal system reservoir.
RESULT: The gage should show 8.5 (±0.6) PSI.

k. Increase the air pressure to 70 (±10) PSI, and again observe the gage over the normal system reservoir.
RESULT: The gage should still show 8.5 (±0.6) PSI.

l. Slowly loosen the normal system reservoir filler cap to determine that the reservoir is actually pressurized, then retighten the cap.

m. Repeat step (l) for the standby system reservoir.

n. Reduce the air pressure on the system to zero, and then relieve the pressure in the reservoirs by loosening either filler cap. Retighten the filler cap after completing this step.

o. Increase air pressure just enough to obtain an indication of 7 PSI on the gage over the normal system reservoir, and then disconnect the air supply and observe the gage after 3 minutes.
RESULT: After 3 minutes, the gage should show at least 6 PSI.

p. Reconnect the reservoir pressurization system to the bleed air system, remove the pressure gage from the tee over the normal system reservoir, and reinstall the bleed nut in the tee.

q. Remove all plugs from the hydraulic lines and reconnect the lines.

r. Service the reservoirs according to the instructions in Chapter 12.

29.1.3 TROUBLESHOOTING.

29.1.3.1 NORMAL (NO. 1) HYDRAULIC SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW OR FLUCTUATING HYDRAULIC PRESSURE INDICATION.		
Insufficient hydraulic fluid in the reservoir.	Check the normal system for evidence of leakage. If no evidence of leakage is found, check the standby system reservoir fluid level. If the standby reservoir is above normal, check the operation of the wing flap bypass and shuttle valves and also flap motor case return.	Retorque leaking connections. Repair or replace defective components. Service both reservoirs as needed according to the instructions in Chapter 12.
Defective pressure indicating system.	Check operation of the pressure indicating system.	Replace defective components as needed.
	Perform a continuity check of the system.	Repair the circuit as needed.

29.1.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW OR FLUCTUATING HYDRAULIC PRESSURE INDICATION (Cont).		
Pump cavitation.	Check the reservoir pressurization system for leaks.	Retorque leaking connections. Repair or replace defective components as needed.
Defective high-pressure relief valve.	Check the operation of the high-pressure relief valve.	Replace the defective high-pressure relief valve.
NO HYDRAULIC PRESSURE INDICATION.		
Defective pressure indicating system.	Check for disengaged or defective HYD PRESS SYS NO. 1 circuit breaker.	Engage disengaged circuit breaker. Replace defective circuit breaker.
	Check that power is available at the 26-volt main AC bus.	Repair circuit as needed.
	Check the operation of the transmitter and indicator.	Replace defective components as needed.
Insufficient fluid in the reservoir.	Check the normal system for evidence of leakage. If no evidence of leakage is found, check the standby system reservoir fluid level. If the standby system reservoir fluid level is higher than normal, check the operation of the flap bypass and shuttle valves and the case flow from the flap drive motors.	Retorque leaking connections. Repair or replace defective components. Service both reservoirs according to the instructions in Chapter 12.
Pump inlet shutoff valve in the closed position.	Check the position of the indicator on the pump inlet shutoff valve. If the valve indicates closed and the fire emergency control handle for engine No. 2 is in the normal (in) position, check the engine ignition and the fire extinguishing circuits.	Repair the circuit as necessary. Replace the valve if necessary.
Pump outlet shutoff valve in the closed position.	Check the operation of the pump outlet shutoff valve. If the valve is closed and the No. 2 fire emergency control handle is in the normal (in) position, check the engine ignition and fire extinguishing circuits.	Repair the circuits as necessary. Replace the valve if necessary.
Inoperative engine-driven hydraulic pump.	Check the pressure filter for signs of pump failure. If there is no evidence of pump failure, check the pump drive.	Replace defective pump, and flush the system lines as necessary.

29.1.3.2 AUXILIARY (NO. 3) HYDRAULIC SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
AUXILIARY HYDRAULIC PUMP MOTOR FAILS TO RUN WHEN THE AUX HYD SYS-3 SWITCH IS PLACED IN THE "ON" POSITION.		
Hydraulic pump control circuit breaker disengaged or defective.	Check the circuit breaker.	Replace or engage the circuit breaker as needed.
Defective hydraulic pump current limiter.	Check the hydraulic pump current limiter.	Replace the current limiter as needed.
Defective auxiliary hydraulic system No. 3 relay.	Check the relay operation.	Replace the relay as needed.
Defective electric motor.	Check the motor.	Replace the pump assembly as needed.
AUXILIARY HYDRAULIC PUMP MOTOR FAILS TO RUN WHEN THE LANDING GEAR CONTROL HANDLE IS PLACED IN THE "LG UP" POSITION.		
"GEAR REL" handle not fully in (on airplanes 5001 thru 5035 not modified by Service Bulletin 329-24 only).	Check the position of the GEAR REL handle.	Place the GEAR REL handle in the fully in position.
"LG CONT" selector switch in the "EMERG" position (on airplanes 5036 thru 5040 and airplanes modified by Service Bulletin 329-24 only).	Check the position of the switch. The switch is near the pilot's control column and should be in the "NORM" position.	Place the switch in the "NORM" (guarded) position.
Defective landing gear auxiliary hydraulic pump relay (on airplanes 5001 thru 5040 only).	Check the operation of relay.	Replace the relay as needed.
Defective auxiliary hydraulic system No. 3 relay.	Try manual pump selection. If pump fails to operate using AUX HYD SYS-3 switch, check relay operation.	Replace the relay as needed.
Defective main gear inboard door uplock switches.	Check switch operation.	Replace switches as needed.
AUXILIARY HYDRAULIC PUMP FAILS TO RUN WITH TRAILING EDGE FLAPS EXTENDING.		
FLAP & BR HYD SYS switch in the "STANDBY-2" position.	Check the switch position.	Return the switch to "NORM-1" position.
Disengaged or defective hydraulic pump control circuit breaker.	Check the circuit breaker.	Replace or engage the circuit breaker as needed.
Defective hydraulic pump current limiter.	Check the current limiter.	Replace the current limiter as needed.

29.1.3.2 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
AUXILIARY HYDRAULIC PUMP FAILS TO RUN WITH TRAILING EDGE FLAPS EXTENDING (Cont).		
Defective flap boost disconnect relay.	Check the operation of the relay.	Replace the relay as needed.
Defective auxiliary hydraulic system No. 3 relay.	Check the operation of the relay.	Replace the relay as needed.
NO HYDRAULIC PRESSURE INDICATION.		
Defective pressure indicating circuit.	Check for disengaged or defective HYD PRESS SYS NO. 1 circuit breaker.	Engage disengaged circuit breaker. Replace defective circuit breaker.
	Check that 26-volt main AC bus power is available to the circuit breaker.	Obtain power on the 26-volt main AC bus.
	Check continuity of the indicating circuit.	Repair circuit as needed to obtain continuity.
Defective auxiliary hydraulic pump assembly.	Check the operation of the pump and motor.	Replace defective pump assembly.

29.1.3.3 RESERVOIR PRESSURIZATION SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
EXCESSIVE RESERVOIR PRESSURE.		
Defective air pressure regulator.	Perform an operational checkout of the reservoir pressurization system.	Replace defective regulator as needed.
INSUFFICIENT RESERVOIR PRESSURE.		
Air leaks in reservoir pressurization system.	Check for leaks in lines and reservoirs.	Correct leaks as conditions dictate.
Restrictions in reservoir pressurization system.	Check lines and reservoir inlet for restrictions.	Relieve restrictions as needed.
RESERVOIR FAILS TO MAINTAIN AIR PRESSURE.		
Air leaks in reservoir pressurization system including reservoir filler caps and drain valves.	Perform an operational checkout of the reservoir pressurization system and check for air leaks.	Correct leaks as condition dictate.

29.1.3.4 FLOW REQUIREMENTS. During the course of troubleshooting it is sometimes desirable to know what the proper pressure reading should be on the dual hydraulic pressure indicator in the flight station. Calculated flow requirements for various operating conditions are tabulated in figure 29-12B. The actual normal flow requirements for an individual airplane may vary from the values presented in figure 29-12B. A series of plots of pump flow versus pump outlet pressure for various engine speeds is presented in figure 29-12C. The maximum flow available at a given engine speed may be determined from the pump speed versus pump flow versus engine speed graph in figure 29-12D.

29.1.4 BLEEDING PROCEDURES.

29.1.4.1 NORMAL AND AUXILIARY HYDRAULIC SYSTEMS. As the normal and auxiliary hydraulic systems are concerned with the same lines and components, they are treated here as one system.

a. Make sure that the landing gear ground safety locks are installed on the gear.

b. Jack the airplane according to the instructions in Chapter 7. All wheels must clear the ground.

c. Connect an external hydraulic power source to the normal hydraulic system according to the instructions in Chapter 12. Adjust the power source to deliver 250 PSI and increase the pressure when necessary just enough to cycle the individual units.

d. Connect 28-volt DC power to the service connections of the airplane according to Chapter 24.

e. After the airplane is jacked clear of the ground, remove all landing gear and door safety locks; and cycle the landing gear through three complete cycles.

CAUTION

When cycling the gear for bleeding purposes, do not exceed hydraulic pressure of 2,000 PSI and a flow of 1/2 gallon per minute. Exceeding these limits may result in loss of snubbing action in the gear actuating cylinders and damage to the main gear uplock assemblies.

f. After cycling the gear, return the landing gear control handle to the "LG DOWN" position and increase the hydraulic pressure to 3,000 PSI for 30 seconds.

g. Reduce hydraulic pressure to 250 PSI, and reinstall the ground locks on the gear.

h. Cycle the ailerons through two complete cycles with the AILERON SYS 1 switch in the "ON" position.

i. Cycle the elevators through two complete cycles with the ELEVATOR SYS 1 switch in the "ON" position.

j. Disconnect the nose gear scissors.

k. With the landing gear control handle in the "LG DOWN" position, cycle the nose steering actuating cylinder through three complete cycles.

Note

The nose gear scissors switch must be in the on-the-ground position in order to operate the nose steering system.

l. Return the steering control wheel to the centered position, and reconnect the nose gear scissors.

m. Make sure that the speed brake ground control valve is in the "NORMAL" position, and then use the speed brake switch to cycle the speed brake through three completes cycles. Return the speed brake ground control handle to the "SAFE"(pulled) position upon completing this step.

n. Make sure that the FLAP & BR HYD SYS switch on the main instrument panel is in the "NORM-1" position, and then actuate the leading and trailing edge wing flaps through three complete cycles.

CAUTION

On airplanes 5001 through 5045, do not actuate the flaps one-half cycle on one hydraulic system and then complete the cycle on the other system. To do so may overfill one reservoir and deplete the other.

o. Depress the left brake pedal approximately one-third down.

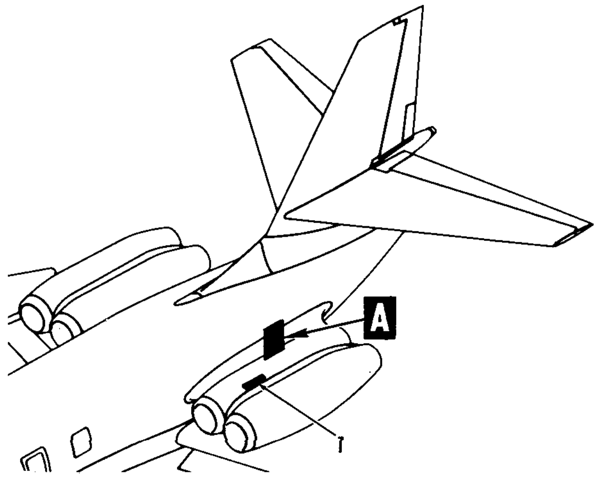
p. Keeping the brake pedal in the position of step (o), open the bleed port of the inboard left brake assembly until fluid runs clear with no air bubbles, and then close the bleed port before releasing the brake pedal.

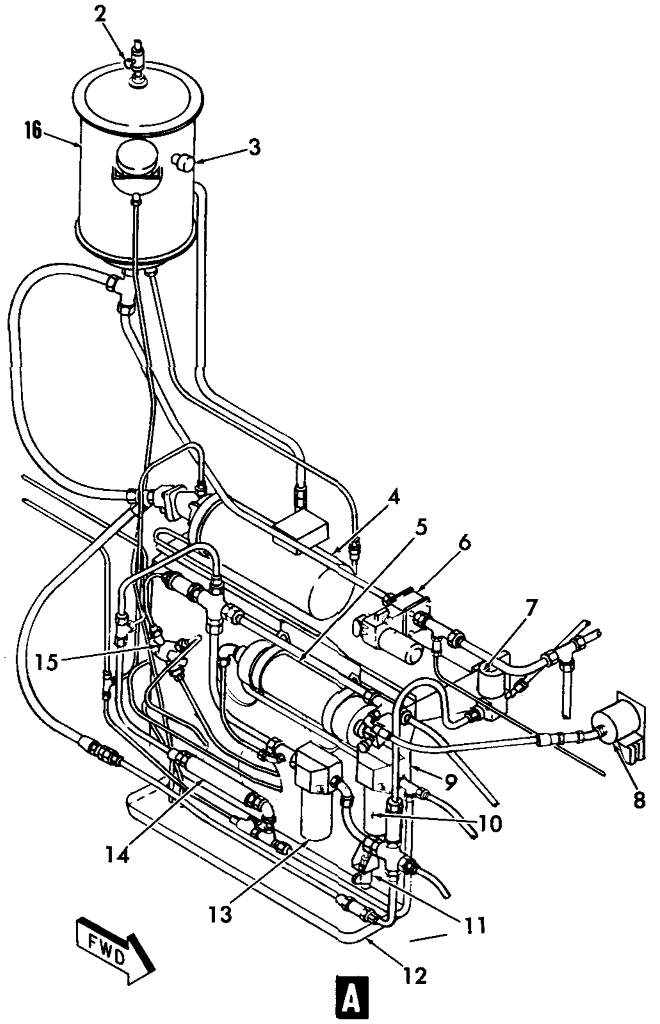
q. Repeat steps (o) and (p) for the outboard left brake assembly, and then release the brake pedal.

r. Repeat steps (o) through (q) for the right wheel brakes.

s. Reduce hydraulic pressure to zero, disconnect external power sources, and service the normal hydraulic system reservoirs and accumulators according to the instructions in Chapter 12.

t. Lower the airplane to the ground in accordance with the instructions in Chapter 7. Remove the jacks.





NORMAL AND AUXILIARY HYDRAULIC SYSTEMS

1. NORMAL HYDRAULIC SYSTEM PUMP
2. RESERVOIR PRESSURIZATION SYSTEM BLEED NUT
3. SERVICING INDICATOR GLASS
4. AUXILIARY HYDRAULIC PUMP
5. NORMAL SYSTEM ACCUMULATOR
6. NORMAL SYSTEM PUMP INLET SHUTOFF VALVE
7. NORMAL SYSTEM PUMP OUTLET SHUTOFF VALVE
8. NORMAL AND AUXILIARY SYSTEM PRESSURE TRANSMITTER
9. LEFT HYDRAULIC PANEL
10. NORMAL SYSTEM RETURN FILTER
11. NORMAL SYSTEM RESERVOIR DRAIN VALVE
12. LEFT HYDRAULIC PANEL DRIP PAN
13. NORMAL SYSTEM PRESSURE FILTER
14. NORMAL AND AUXILIARY SYSTEM HIGH PRESSURE RELIEF VALVE
15. SPEED BRAKE SELECTOR VALVE
16. NORMAL SYSTEM RESERVOIR

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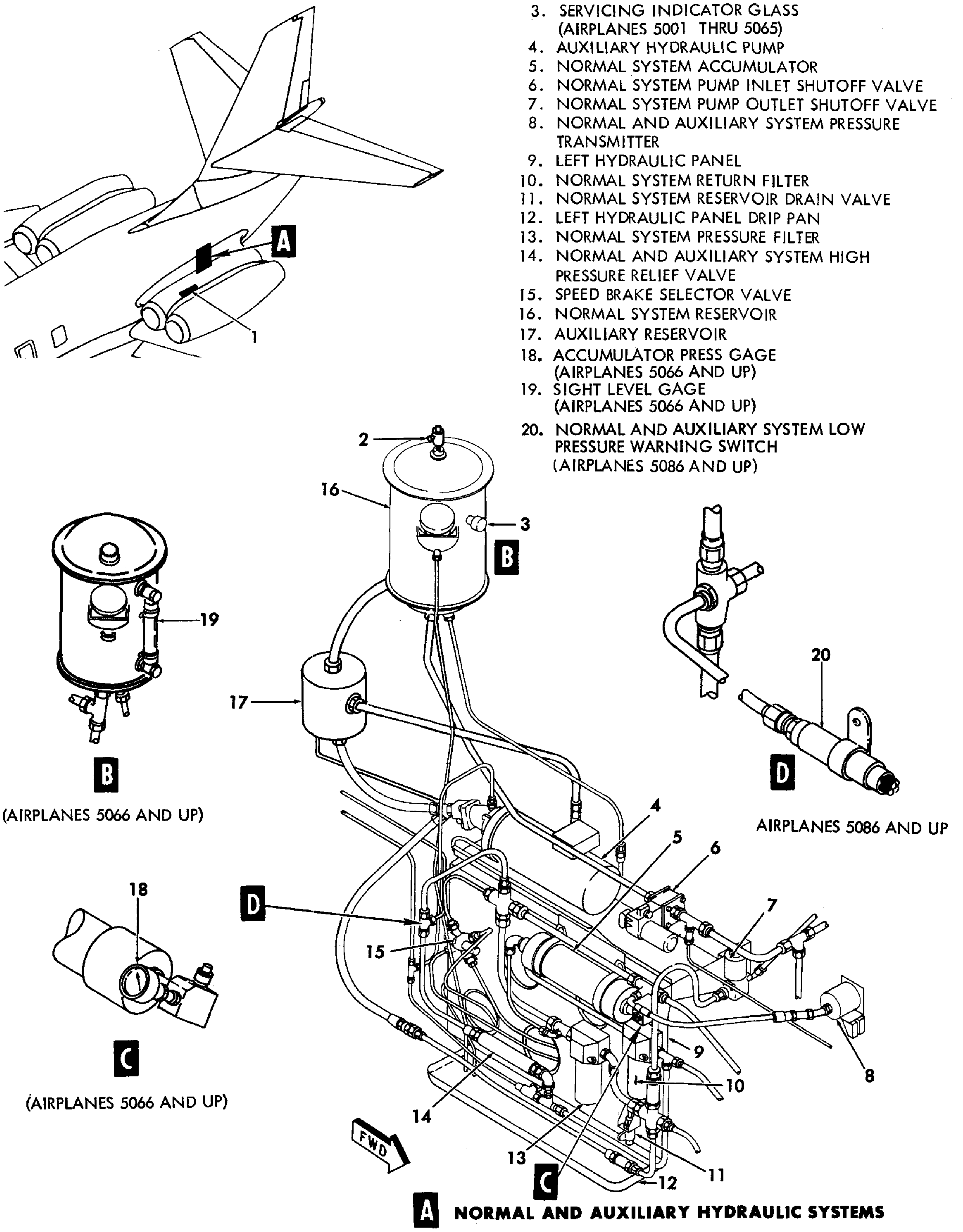
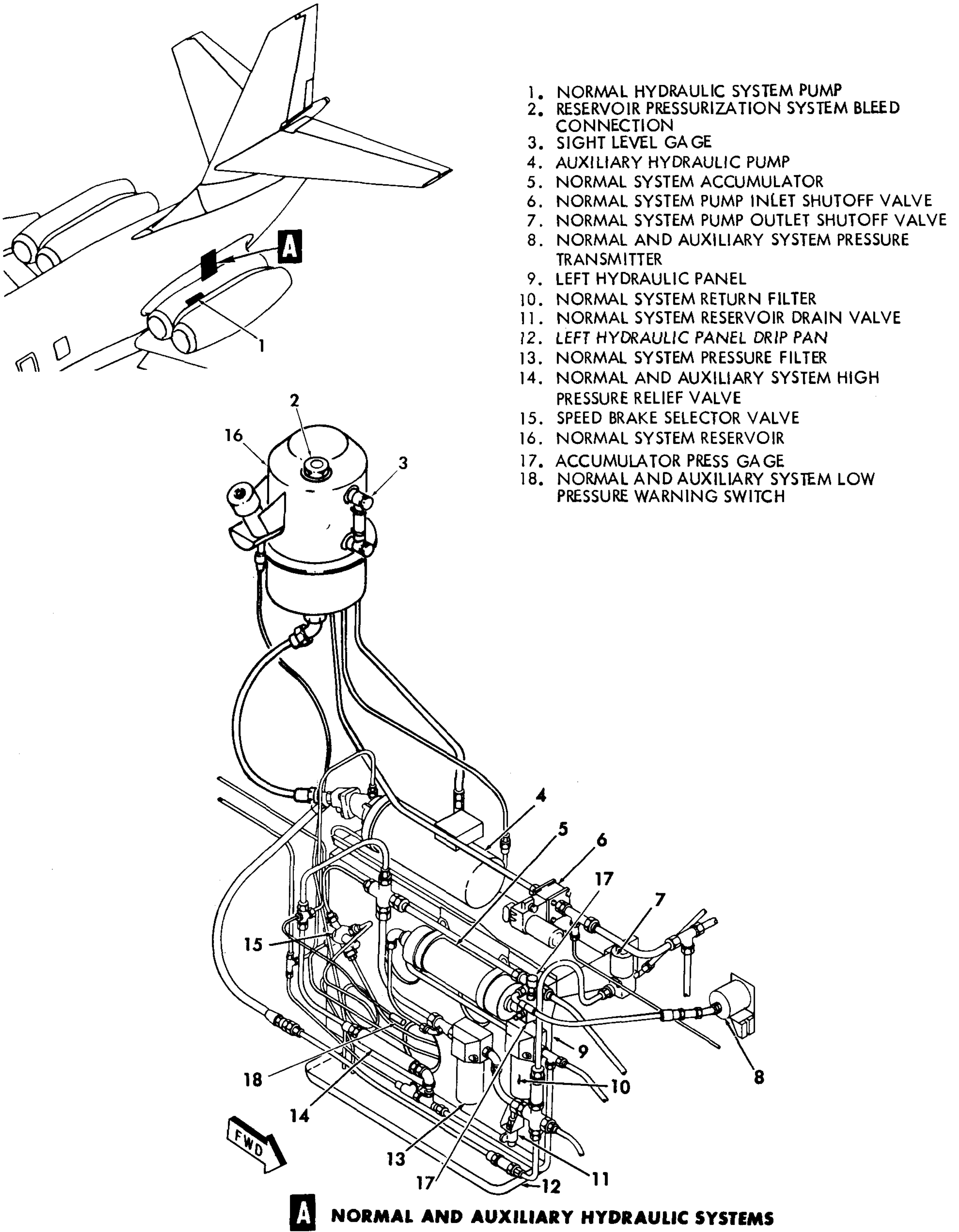
Figure 29-11. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5001 thru 5045)



1. NORMAL HYDRAULIC SYSTEM PUMP
2. RESERVOIR PRESSURIZATION SYSTEM BLEED NUT
3. SERVICING INDICATOR GLASS (AIRPLANES 5001 THRU 5065)
4. AUXILIARY HYDRAULIC PUMP
5. NORMAL SYSTEM ACCUMULATOR
6. NORMAL SYSTEM PUMP INLET SHUTOFF VALVE
7. NORMAL SYSTEM PUMP OUTLET SHUTOFF VALVE
8. NORMAL AND AUXILIARY SYSTEM PRESSURE TRANSMITTER
9. LEFT HYDRAULIC PANEL
10. NORMAL SYSTEM RETURN FILTER
11. NORMAL SYSTEM RESERVOIR DRAIN VALVE
12. LEFT HYDRAULIC PANEL DRIP PAN
13. NORMAL SYSTEM PRESSURE FILTER
14. NORMAL AND AUXILIARY SYSTEM HIGH PRESSURE RELIEF VALVE
15. SPEED BRAKE SELECTOR VALVE
16. NORMAL SYSTEM RESERVOIR
17. AUXILIARY RESERVOIR
18. ACCUMULATOR PRESS GAGE (AIRPLANES 5066 AND UP)
19. SIGHT LEVEL GAGE (AIRPLANES 5066 AND UP)
20. NORMAL AND AUXILIARY SYSTEM LOW PRESSURE WARNING SWITCH (AIRPLANES 5086 AND UP)

A **NORMAL AND AUXILIARY HYDRAULIC SYSTEMS**

Figure 29-12. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5046 thru 5092)



1. NORMAL HYDRAULIC SYSTEM PUMP
2. RESERVOIR PRESSURIZATION SYSTEM BLEED CONNECTION
3. SIGHT LEVEL GAGE
4. AUXILIARY HYDRAULIC PUMP
5. NORMAL SYSTEM ACCUMULATOR
6. NORMAL SYSTEM PUMP INLET SHUTOFF VALVE
7. NORMAL SYSTEM PUMP OUTLET SHUTOFF VALVE
8. NORMAL AND AUXILIARY SYSTEM PRESSURE TRANSMITTER
9. LEFT HYDRAULIC PANEL
10. NORMAL SYSTEM RETURN FILTER
11. NORMAL SYSTEM RESERVOIR DRAIN VALVE
12. LEFT HYDRAULIC PANEL DRIP PAN
13. NORMAL SYSTEM PRESSURE FILTER
14. NORMAL AND AUXILIARY SYSTEM HIGH PRESSURE RELIEF VALVE
15. SPEED BRAKE SELECTOR VALVE
16. NORMAL SYSTEM RESERVOIR
17. ACCUMULATOR PRESS GAGE
18. NORMAL AND AUXILIARY SYSTEM LOW PRESSURE WARNING SWITCH

Figure 29-12A. Normal and Auxiliary Hydraulic System Components and Locations (Airplanes 5093 and Up)

HYDRAULIC FUNCTION	TAXI	TAKE-OFF		CRUISE	APPROACH		LANDING ROLL	OPERATING TIME
		GEAR UP	FLAPS UP		GEAR DOWN	FLAPS DOWN		
	G.P.M.	G.P.M.	G.P.M.	G.P.M.	G.P.M.	G.P.M.	G.P.M.	SEC.
LANDING GEAR	0	6.5	0	0	3.3	0	0	11.2
TRAILING EDGE FLAPS	0	0	5.0	0	0	5.0	0	15.5
LEADING EDGE FLAPS	0	0	2.5	0	0	2.5	0	7.9 △1 9.3 △2
FLIGHT CONTROL BOOST	0	1.0	1.0	0.5	1.0	1.0	0	
THRUST REVERSE	0	0	0	0	0	0	12.0	1.3
NOSE WHEEL STEER	1.2	0	0	0	0	0	0	
WHEEL BRAKES	0.8	0	0	0	0	0	2.8	
SPEED BRAKE	0	0	0	3.0	0	0	0	6.6
SYSTEM LEAKAGE	0.5	0.5	0.5	0.5	0.5	0.5	0.5	
TOTAL REQUIREMENTS	2.5	8.0	9.0	4.0	4.8	9.0	15.3	
ENGINE PUMP	3.9	9.9	9.9	9.1	7.8	7.0	3.9	
MOTOR PUMP	0	5.0	0	0	0	5.0	0	
ACCUMULATORS	0	0	0	0	0	0	11.4	
TOTAL CAPABILITY	3.9	14.9	9.9	9.1	7.8	12.0	15.3	

NOTE

△1 SERIALS 5001 THRU 5054 (JW1007-4 FLAP MOTOR).

△2 SERIALS 5055 AND UP (JW1005-13 FLAP MOTOR).

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Figure 29-12B. Flow Requirement Summary - Normal System (No. 1)

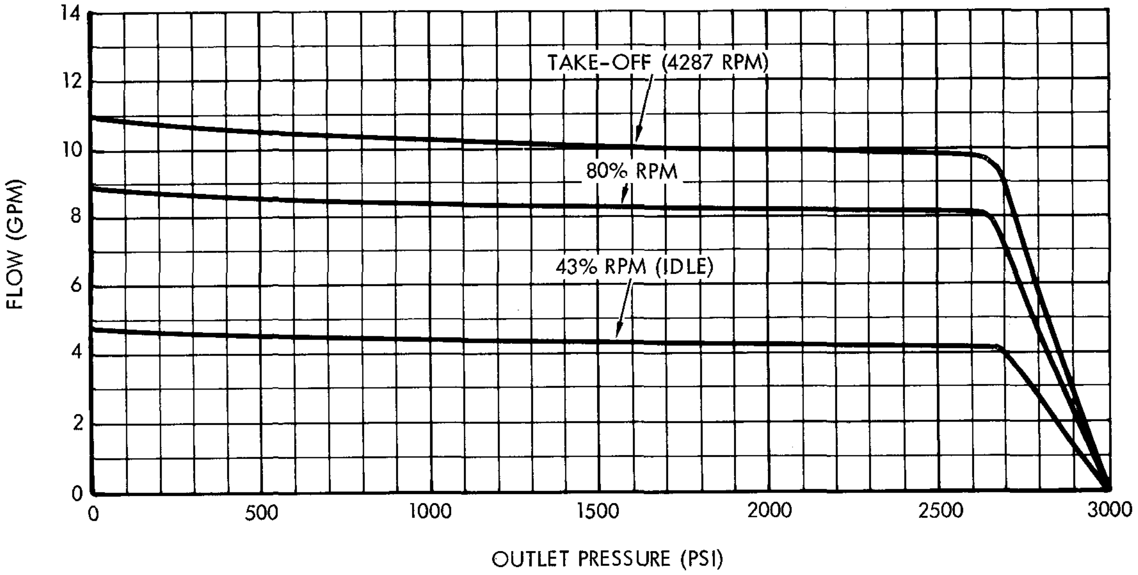

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Figure 29-12C. Engine Driven Pump Flow vs Pressure (Theoretical) JH1000-1,-2 Pumps

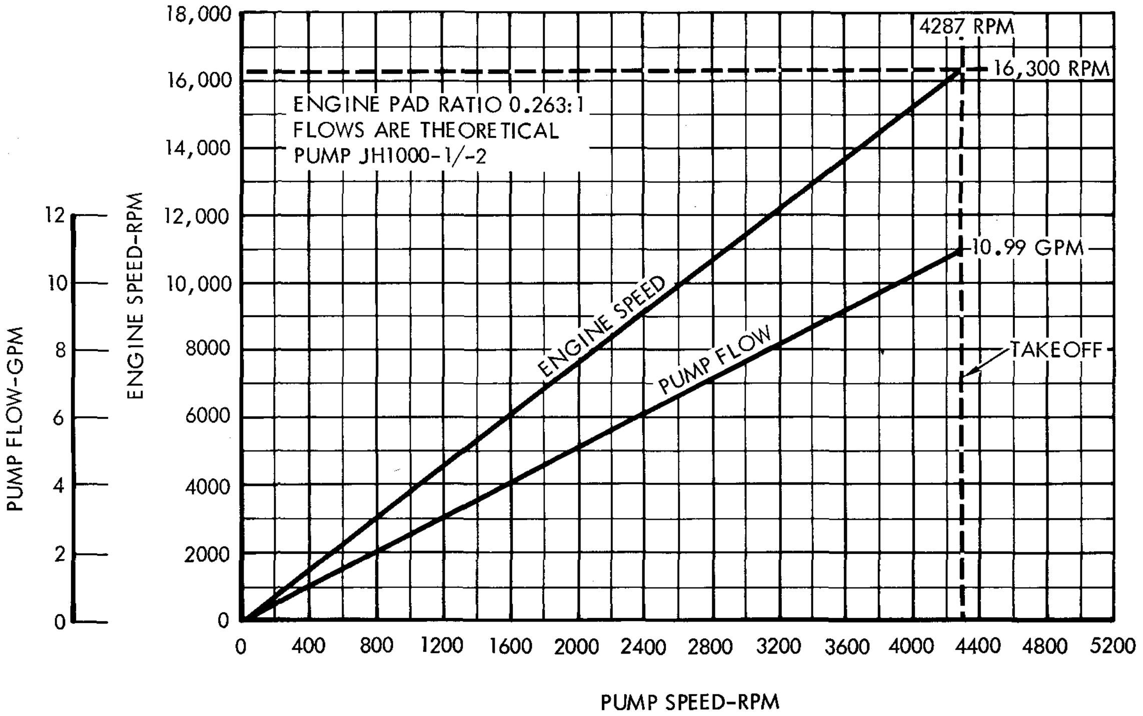

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Figure 29-12D. Pump Speed vs Flow vs Engine Speed

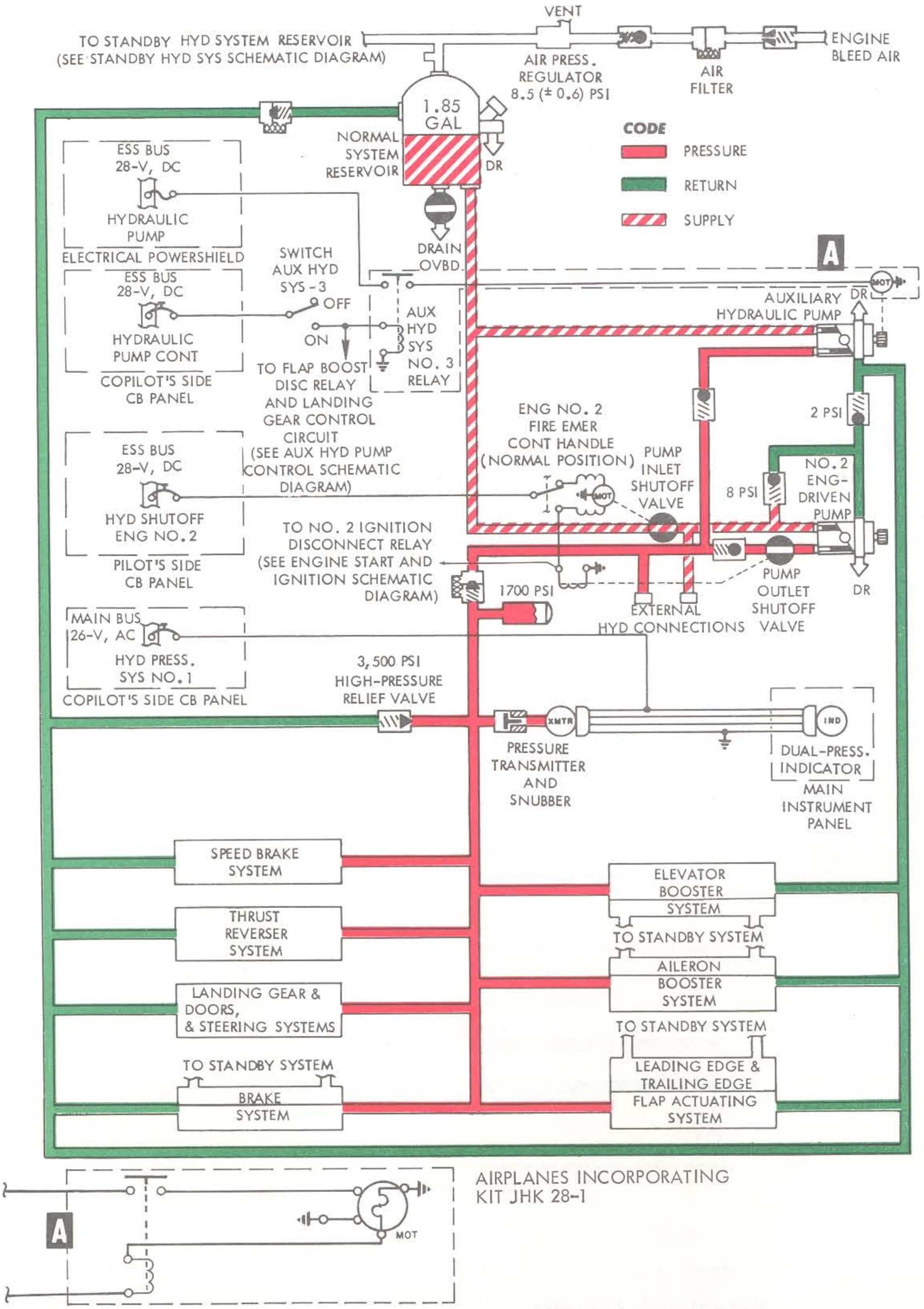


Figure 29-13. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5001 thru 5045)

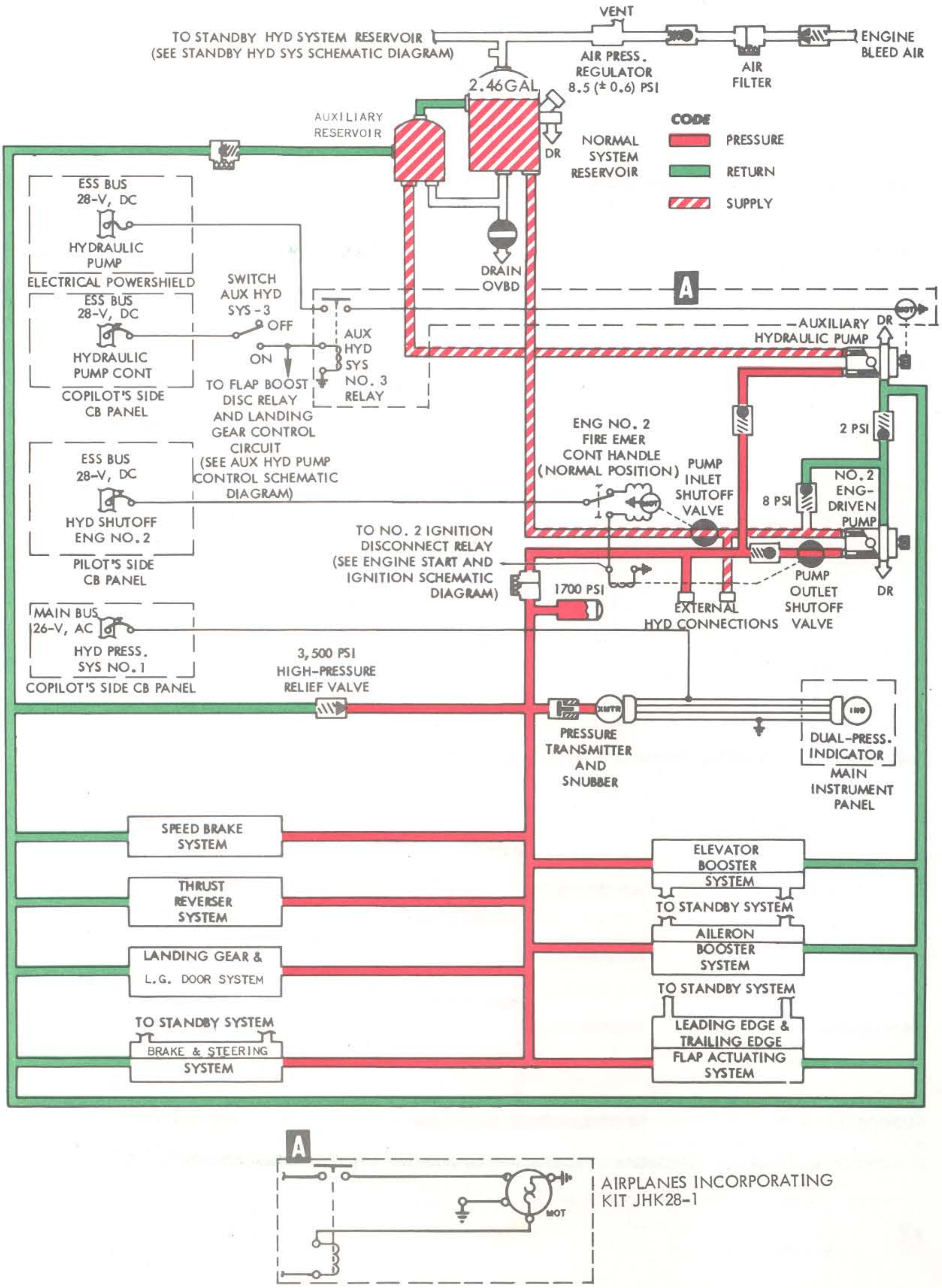


Figure 29-14. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5046 thru 5085)

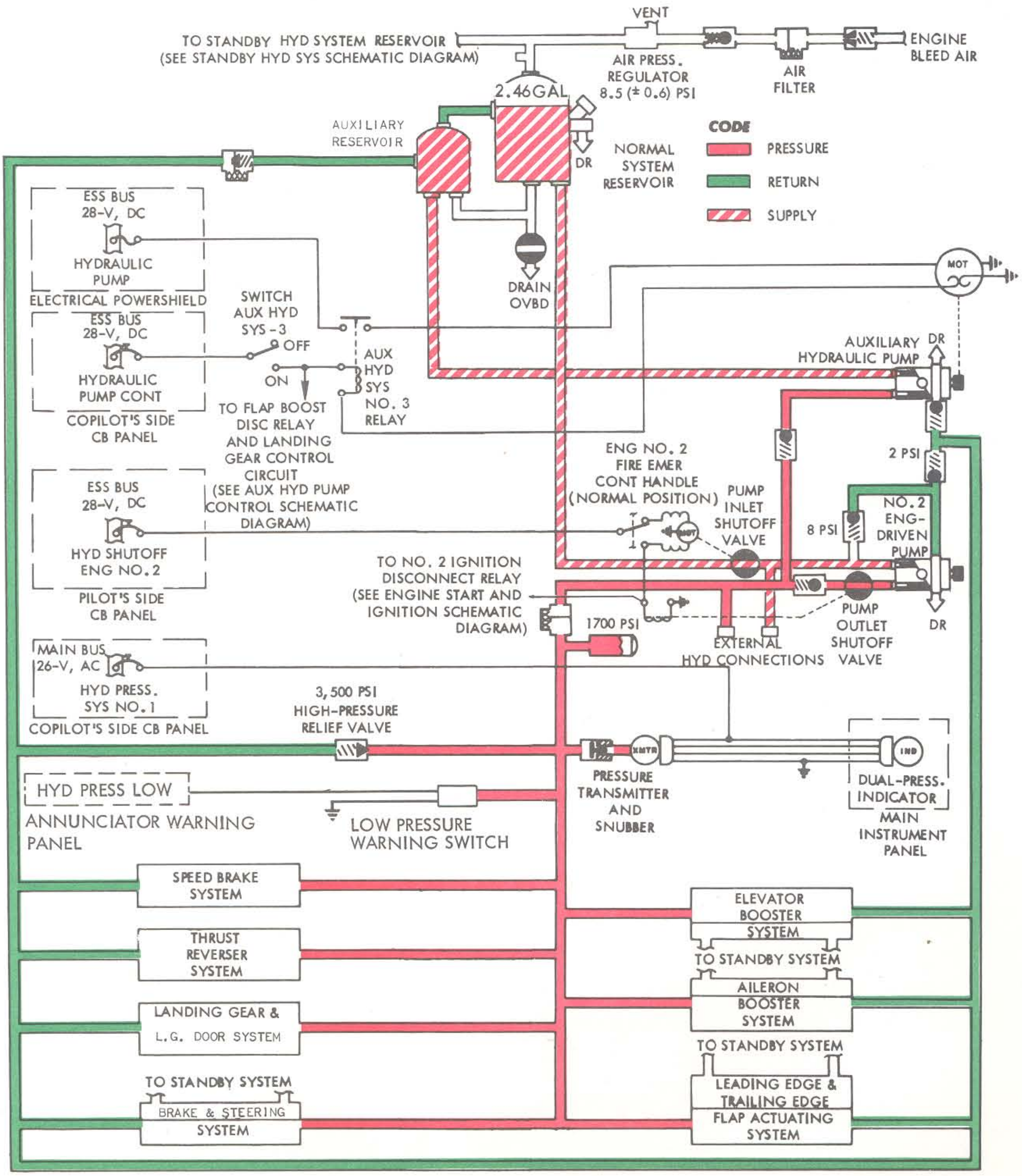


Figure 29-14A. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5086 thru 5092)

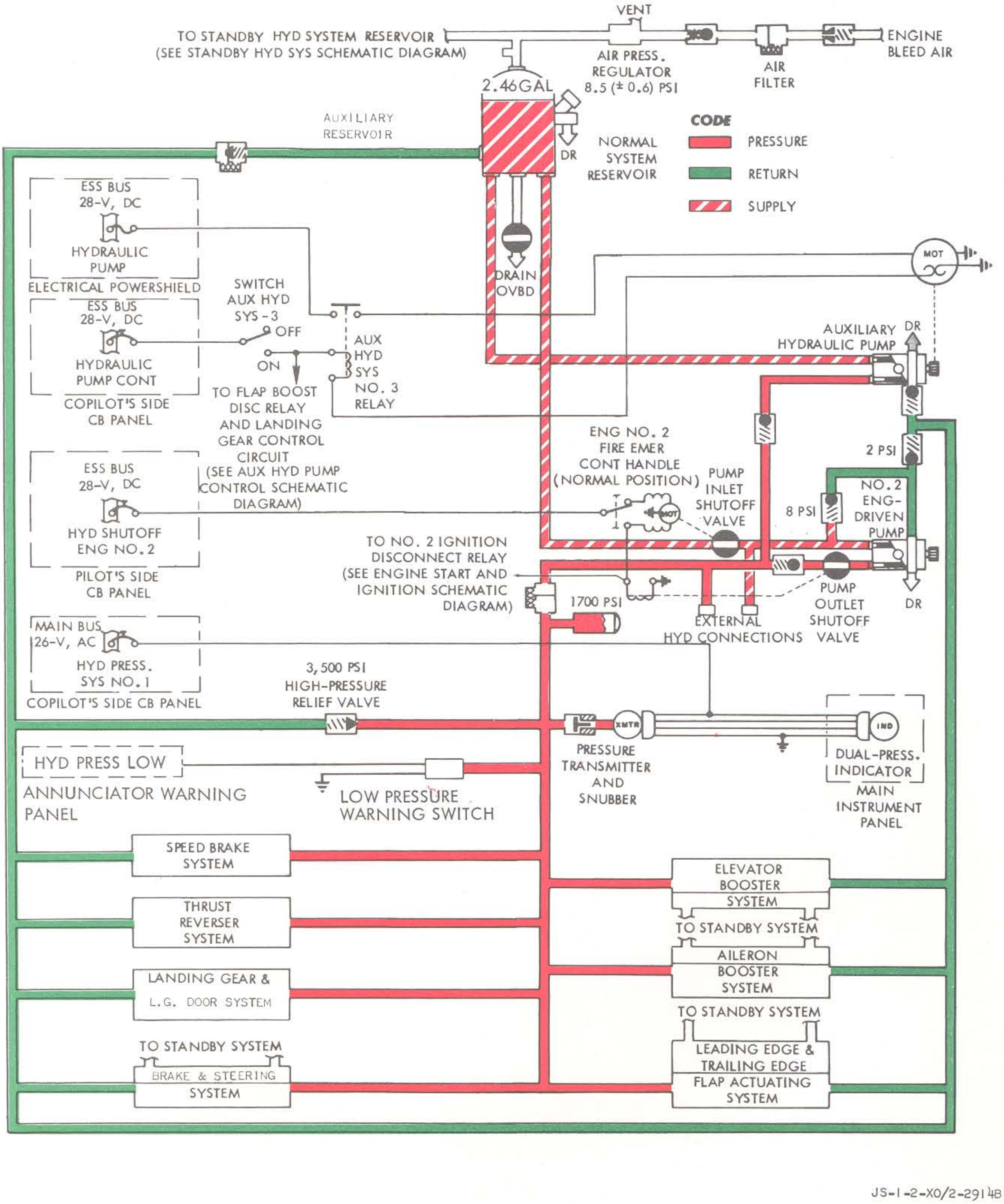
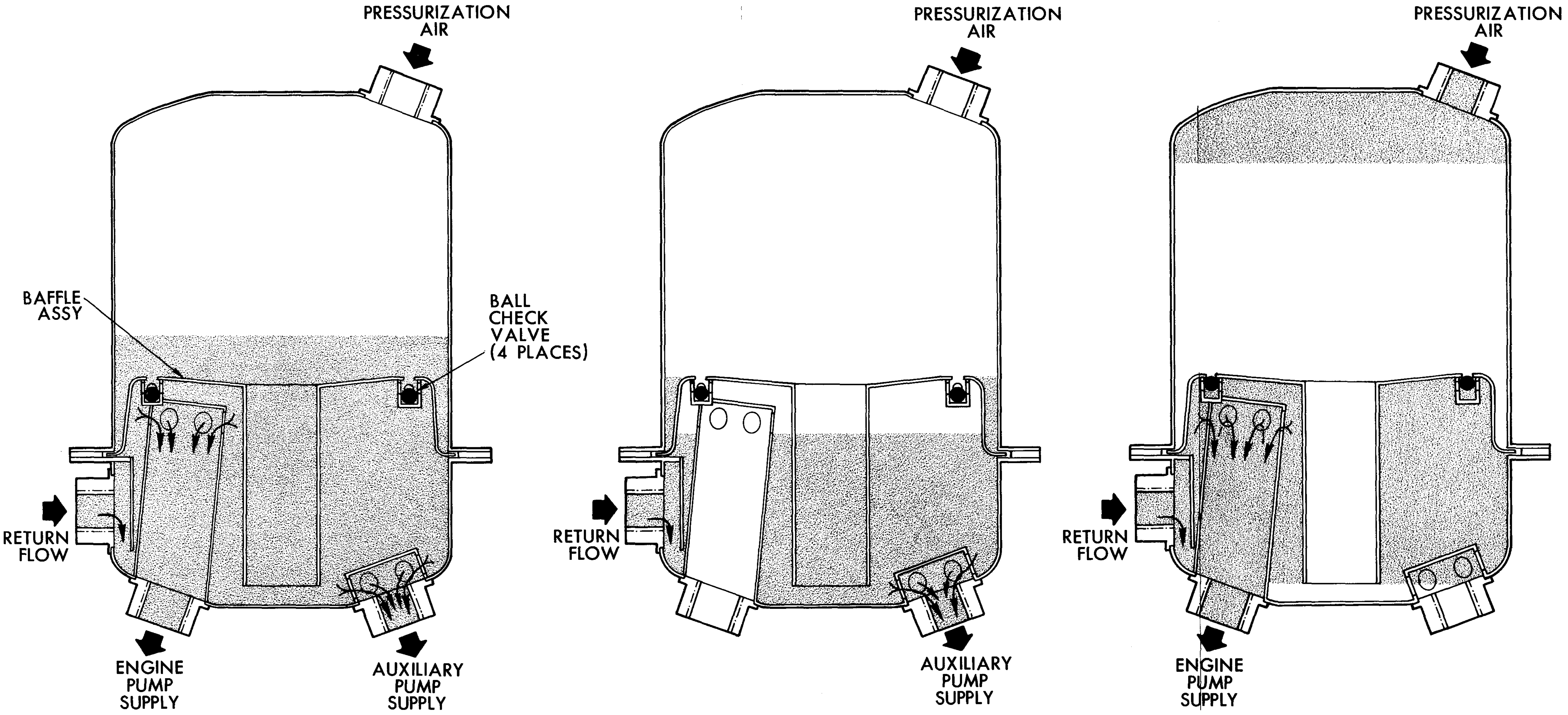


Figure 29-14B. Normal and Auxiliary Hydraulic System Schematic Diagram (Airplanes 5093 and Up)



NORMAL OPERATION

NORMAL OPERATION: DURING NORMAL OPERATION OF THE RESERVOIR OIL FLOWS ARE AS SHOWN ON THE DIAGRAM. THE CENTER STANDPIPE OF THE BAFFLE PERMITS FREE FLOW OF OIL BETWEEN THE UPPER AND LOWER SECTIONS OF THE RESERVOIR. THE BALL CHECK VALVES ALLOW ANY TRAPPED AIR TO BLEED TO THE UPPER SECTION OF THE RESERVOIR.

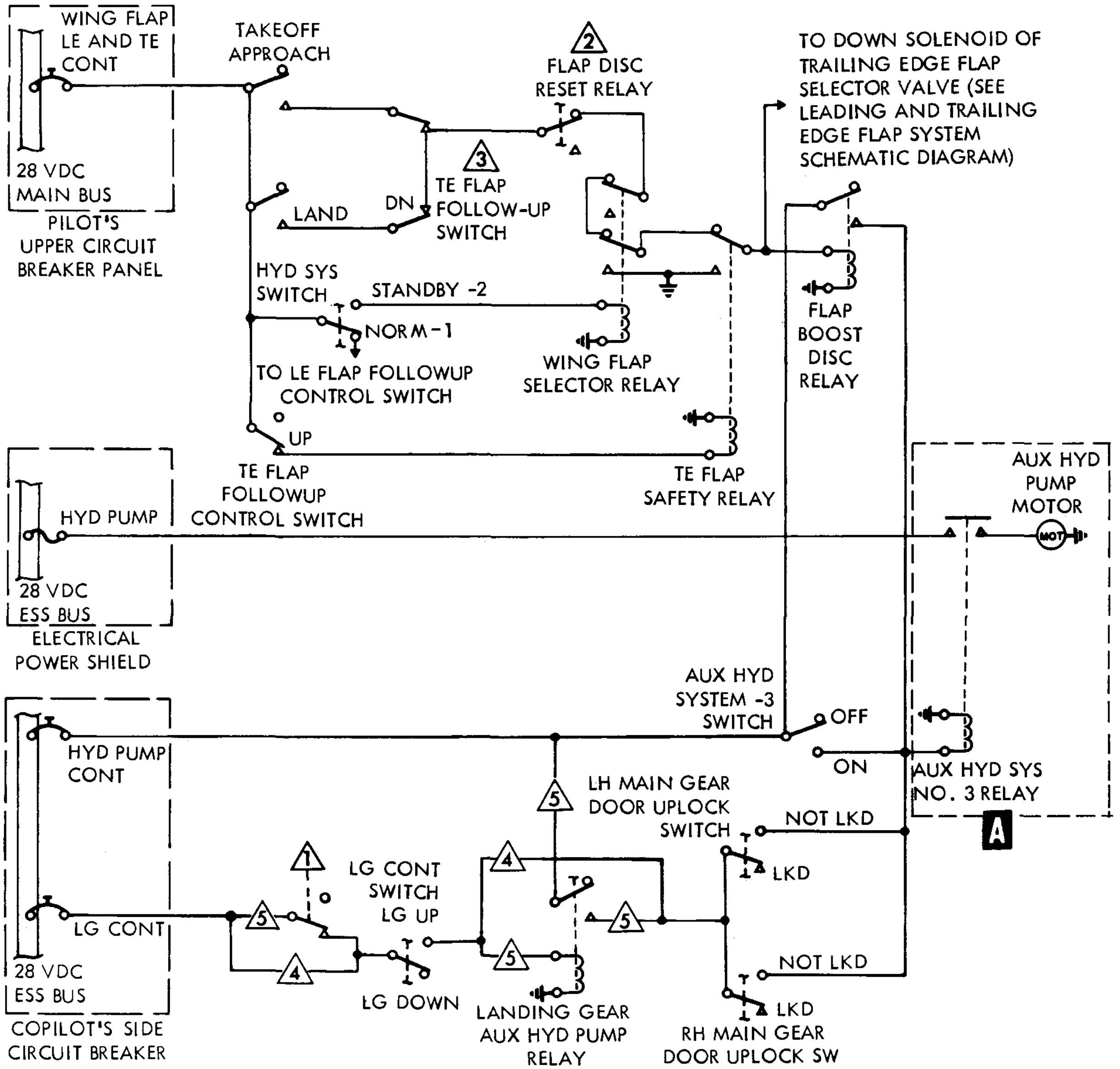
AUXILIARY PUMP OPERATION

AUXILIARY PUMP OPERATION: IN THE EVENT THAT OIL IS LOST THROUGH THE ENGINE PUMP CIRCUIT A RESERVE VOLUME OF OIL IS MAINTAINED IN THE LOWER SECTION OF THE RESERVOIR TO SUPPLY THE AUXILIARY PUMP.

NEGATIVE "G" OPERATION

NEGATIVE "G" OPERATION: IF A NEGATIVE "G" LOAD CONDITION IS IMPOSED ON THE RESERVOIR ASSEMBLY THE NET EFFECT WOULD BE SIMILAR TO INVERTING THE ASSEMBLY AND ALL OIL NOT RESTRAINED BY THE BAFFLE WOULD MOVE TO THE TOP OF THE RESERVOIR ASSEMBLY AS SHOWN IN THE DIAGRAM. A SUPPLY OF OIL WOULD BE MAINTAINED FOR THE ENGINE PUMP BY THE BAFFLE AND SINCE THE RETURN FLOW IS ON THE PUMP SUPPLY SIDE OF THE BAFFLE CONTINUOUS PUMP OPERATION WOULD BE POSSIBLE. THE BALL CHECK VALVES WOULD BE CLOSED DURING NEGATIVE "G" OPERATION.

Figure 29-14C. Operation of "Negative G" Reservoir



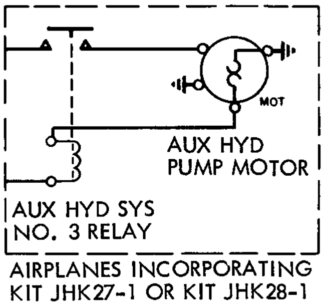
⚠1 MICROSWITCH ON "GEAR REL" HANDLE OF AIRPLANES 5001 THRU 5035 NOT MODIFIED BY SERVICE BULLETIN 329-24; "LG CONT" SWITCH ON AIRPLANES MODIFIED BY SERVICE BULLETIN 329-24.

⚠2 ENERGIZED WHEN FLAPS ARE ASYMMETRICALLY POSITIONED.

⚠3 COMPLETE SWITCH NOT SHOWN: "53%" AND "UP" PORTIONS OMITTED FOR CLARITY.

⚠4 ON AIRPLANES 5041 THRU 5085

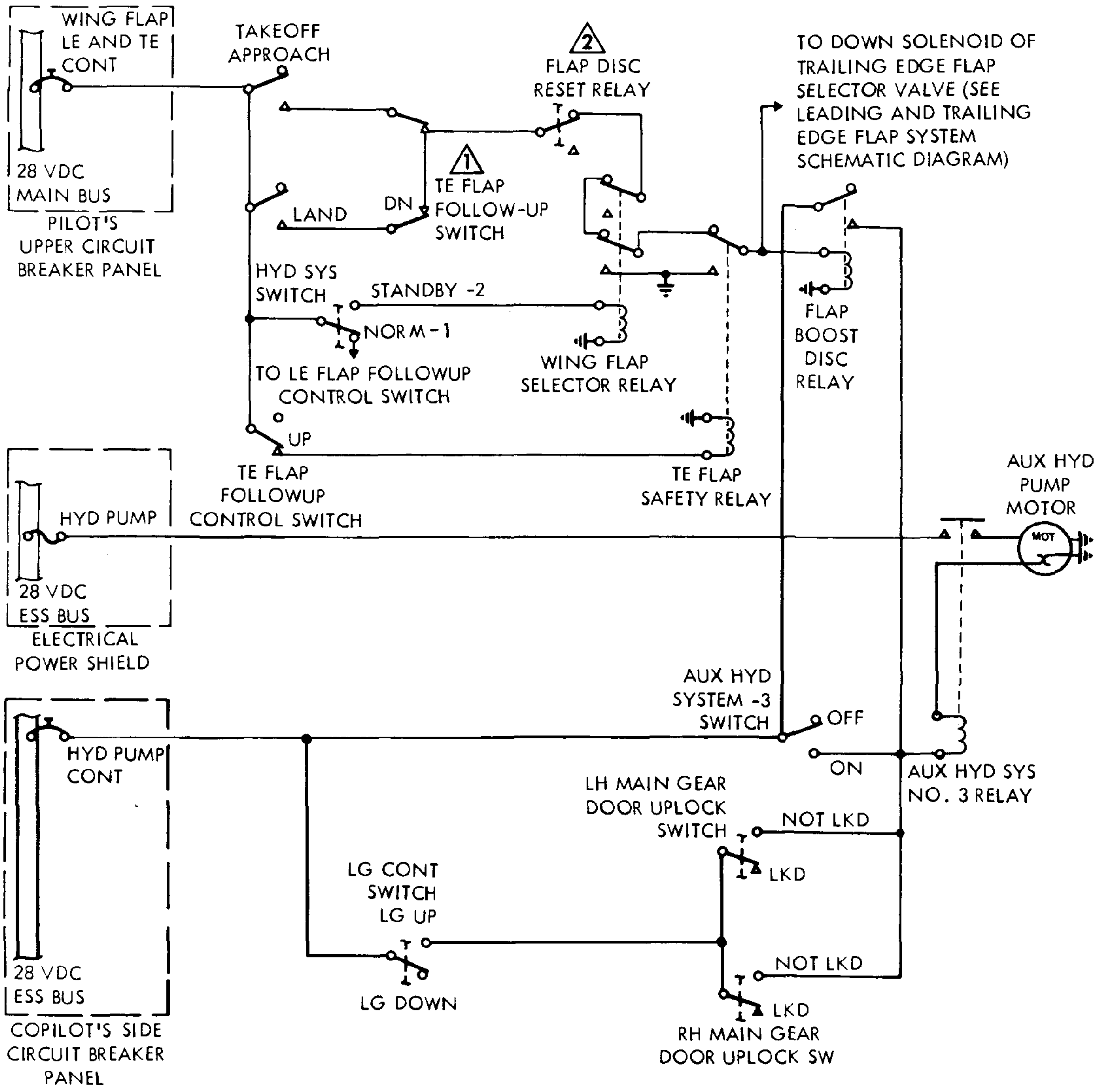
⚠5 ON AIRPLANES 5001 THRU 5040 ONLY.



AIRPLANES INCORPORATING KIT JHK27-1 OR KIT JHK28-1

A

Figure 29-15. Auxiliary Hydraulic Pump Control Schematic Diagram (Airplanes 5001 thru 5085)



1. COMPLETE SWITCH NOT SHOWN: "53%" AND "UP" PORTIONS OMITTED FOR CLARITY.

2. ENERGIZED WHEN FLAPS ARE ASYMMETRICALLY POSITIONED.

Figure 29-15A. Auxiliary Hydraulic Pump Control Schematic Diagram (Airplanes 5086 and Up)

SECTION 29.2
STANDBY (NO. 2) HYDRAULIC SYSTEM

Contents Page

DESCRIPTION AND OPERATION 29-35
General 29-35
Standby (No. 2) Hydraulic System 29-35
Reservoir Pressurization System 29-36
Reservoir (Airplanes 5001 through 5092) 29-36
Reservoir (Airplanes 5093 and Up) 29-36
Servicing Indicator Glass 29-36
Pump Inlet Shutoff Valve 29-36
Engine-Driven Hydraulic Pump 29-36
Pump Outlet Shutoff Valve 29-36
External Hydraulic Power Connections 29-36
Hydraulic Filters 29-36
Standby System Accumulator and Air Filler Valve 29-37
Standby System Pressure Transmitter 29-37
Dual-Pressure Indicator 29-37
System High-Pressure Relief Valve 29-37
Low Pressure Warning Switch (Airplanes 5086 and Up) 29-37
Check Valves 29-37

OPERATIONAL CHECKOUT 29-37
Standby (No. 2) Hydraulic System 29-37
Reservoir Pressurization System 29-37

TROUBLESHOOTING 29-38
Standby (No. 2) Hydraulic System 29-38
Reservoir Pressurization System 29-39

BLEEDING PROCEDURES 29-39
Standby (No. 2) Hydraulic System 29-39

FIGURES

29-16. Standby Hydraulic System Components and Locations 29-41
29-17. Standby Hydraulic System Schematic Diagram (Airplanes 5001 through 5045) 29-42
29-18. Standby Hydraulic System Schematic Diagram (Airplanes 5046 and Up) 29-43

29.2.1 DESCRIPTION AND OPERATION.

29.2.1.1 GENERAL. (See figures 29-9, 29-10, 29-17, and 29-18.) The standby (No. 2) hydraulic system normally powers one side of the aileron booster assembly and one side of the elevator booster assembly, but can be selected to power the wheel brakes, and wing flaps. On airplanes 5046 and up, the standby system can also be selected to power the nose wheel steering system. When using the standby hydraulic system to power the wheel brakes and wing flaps, the anti-skid system and the leading edge flap control circuits are inoperative.

29.2.1.2 STANDBY (NO. 2) HYDRAULIC SYSTEM. (See figure 29-16.)

a. The standby hydraulic system derives its power from an engine-driven pump on engine No. 3. This system consists of a pressurized reservoir, an engine-driven pump, a pump inlet shutoff valve, a pump outlet shutoff valve, a pressure filter, a return filter, an accumulator, a pressure transmitter and indicator, a low-pressure warning switch (airplanes 5086 and up), a high pressure relief valve, external hydraulic connections, and check valves to maintain proper directional flow.

29.2.1.2 (Cont).

b. Fluid from the standby system reservoir flows through the pump inlet shutoff valve and into the engine-driven pump. This fluid leaves the pump under pressure and travels through a check valve, a pressure filter, and the pump outlet shutoff valve. This fluid then pressurizes the pressure transmitter, the low-pressure warning switch (airplanes 5086 and up) and the standby hydraulic system accumulator. The high pressure relief valve relieves excessive pressure back to the reservoir and thereby protects the system from damage due to excessive pressure. Normally the standby hydraulic system powers one side of the aileron booster assembly and one side of the elevator booster assembly. Fluid returning from these components dumps into the standby system reservoir. For airplanes 5001 through 5045, placing the FLAP & BR HYD SYS switch in the "STANDBY-2" position admits standby system power to the wing flap and wheel brake systems. Shuttle valves in the wing flap and wheel brake systems prevent standby system fluid from mixing with the normal system fluid. For airplanes 5046 and up, placing the HYD SYS FLAPS, BRAKES AND STEERING switch in the STANDBY-2 position admits standby system power to the wing flap, wheel brake and nose wheel steering systems. Dual shuttle valves in the wing flap and nose wheel steering systems and single shuttle valves in the wheel brake system separate the standby and normal systems. A pump run-around circuit maintains pump lubrication when the No. 3 fire emergency control handle is pulled.

c. Only one accumulator is pressurized by the standby hydraulic system: the standby system accumulator.

29.2.1.3 RESERVOIR PRESSURIZATION SYSTEM. (See figures 29-13 and 29-14.) One reservoir pressurization system pressurizes both the normal and the standby hydraulic system reservoir. Refer to Section 29.1.1.5 for a description of this system.

29.2.1.4 RESERVOIR. (Airplanes 5001 through 5092.) The standby system reservoir is a cylindrical tank with a servicing indicator glass, a pressure port, a drain port, a return port and an air pressurization port. The drain port is connected to a tube which connects to a spring-loaded-closed drain valve near the right hydraulic panel drip pan. A scupper below the filler neck catches spilled fluid and drains it overboard. For airplanes 5046 and up, the standby reservoir capacity is increased to accommodate nose wheel steering. For reservoir hydraulic fluid capacity, refer to Chapter 12.

29.2.1.5 RESERVOIR (Airplanes 5093 and Up). The standby system reservoir is a cylindrical tank with a tube-type fluid level gage and ports for: pressurization air, return flow, engine pump supply, and drain. Internally the reservoir is divided into upper and lower volumes by a baffle assembly. The upper volume provides the space required for displacement fluid and expansion. The baffle assembly restrains fluid in the lower volume to assure continuous pump supply under negative "G" conditions. See figure 29-14C, Normal Operation and Negative "G" Operation, for a pictorial presentation of these phases of operation. When a reservoir is disassembled for maintenance, care should be taken to ensure that the baffle assembly is reinstalled correctly. The check valves and pipe should be located beneath the baffle.

29.2.1.6 SERVICING INDICATOR GLASS. On airplanes 5001 thru 5065, the servicing indicator glass is a small, round glass with a yellow dot in the center. As this glass is near the top of the reservoir, fluid level is visible only when the reservoir is close to being properly serviced. Airplanes 5066 and up have tube-type fluid level gages to indicate fluid levels in the hydraulic reservoirs. "FULL" and "REFILL" levels are indicated on the level gages. Complete servicing instructions are shown on the servicing placards for the reservoirs.

29.2.1.7 PUMP INLET SHUTOFF VALVE. A motor-operated shutoff valve shuts off fluid flow to the standby system pump when the fire emergency control handle for engine No. 3 is pulled. This valve is in the aft compartment, over the right hydraulic panel.

29.2.1.8 ENGINE-DRIVEN HYDRAULIC PUMP. The standby system pump is a variable-delivery, piston-type pump with a rated output of 3,000 PSI at no flow, and 2,525 PSI at 8.6 gallons per minute. It is on an accessory drive pad on the lower right side of the No. 3 engine accessory section.

29.2.1.9 PUMP OUTLET SHUTOFF VALVE. A solenoid-operated shutoff valve shuts off fluid flow from the engine-driven pump on engine No. 3 when the fire emergency control handle for engine No. 3 is pulled. This valve is spring loaded to open when de-energized and closes when energized.

29.2.1.10 EXTERNAL HYDRAULIC POWER CONNECTIONS. External hydraulic power connections for the standby hydraulic system are located just aft of the wing-to-fuselage fairing on the right side of the airplane. There are two connections: one labeled SUCTION and one labeled PRESSURE. When not in use, these connections are capped and a door is closed over them for streamlining. For instructions on connecting external hydraulic power, refer to Chapter 12.

29.2.1.11 HYDRAULIC FILTERS. There are two hydraulic filters in the standby hydraulic system: a pressure filter and a return filter. The pressure filter removes contaminants from the fluid after it leaves the pump and before it reaches the subsystems; the return filter removes contaminants from the fluid after it leaves the subsystems and before it reaches the reservoir. Both filters are of the in-line type with replaceable filter elements. Arrows on the filter housings indicate the direction of fluid flow. On airplanes 5001 through 5045 an internal bypass valve in each filter assembly unseats at 50 (±5) PSI differential pressure to relieve hydraulic fluid around clogged filter elements. This bypass valve reseats at 32 PSI minimum differential pressure. For airplanes 5046 and up the pressure filter does not have an internal bypass valve. This to prevent contamination from a possible pump failure from entering the system. For airplanes 5036 and up, filter pressure drop indicators are installed in the pressure and return filters. Each indicator consists of a red button which extends approximately 1/5-inch from the bottom of the filter assembly when the differential pressure across the filter exceeds 35 (±5) PSI. Once extended, the indicator must be reset manually. Elements should be cleaned or replaced when the buttons extend. Differential pressure indicators do not operate at temperatures below 30 (±20) degrees Fahrenheit. For airplanes 5046 and up, the filter pressure drop indicator on the no-bypass pressure filter will extend when the differential pressure across the filter element exceeds 100 (±15) PSI.

29.2.1.11 (Cont.)

Note

No-bypass pressure filter assemblies and return assemblies with differential pressure indicators are available as preferred spares. (If the no-by-pass pressure filter is installed, new nameplates should be installed. Service Bulletin 329-84 provides the necessary parts and information for airplanes 5001 through 5045.)

CAUTION

Do not install a no-bypass filter in the hydraulic return line.

29.2.1.12 STANDBY SYSTEM ACCUMULATOR AND AIR FILLER VALVE. A 25 cubic inch capacity, piston-type accumulator minimizes surge in the standby hydraulic system. It carries a 1,700-PSI precharge of dehydrated air or nitrogen. An air filler valve provides a means of servicing the accumulator. Airplanes 5066 and up are equipped with accumulator pressure gauges to facilitate checking of precharge pressure. (Refer to Paragraph 29.0.1.8.)

29.2.1.13 STANDBY SYSTEM PRESSURE TRANSMITTER. A standby hydraulic system pressure transmitter converts hydraulic pressure into electrical signals and routes these signals to the dual-pressure indicator on the main instrument panel. These signals position the No. 2 pointer to show standby system pressure. This transmitter is on the right hydraulic panel in the aft compartment.

29.2.1.14 DUAL-PRESSURE INDICATOR. For a description of this indicator, refer to Section 29.1.1.18.

29.2.1.15 SYSTEM HIGH-PRESSURE RELIEF VALVE. A system high-pressure valve cracks at 3,500 (±50) PSI to relieve excessive hydraulic pressure back to the reservoir, to protect the system from damage due to excessive pressure. This relief valve is attached to the right hydraulic panel in the aft compartment.

29.2.1.16 LOW PRESSURE WARNING SWITCH (AIRPLANES 5086 AND UP.) For a description of this warning switch, refer to para 29.1.1.20.

29.2.1.17 CHECK VALVES. Several check valve assemblies are installed in standby system hydraulic plumbing, positioned so as to prevent undesirable hydraulic fluid flow.

a. A check valve is installed in the pressure line from the No. 3 engine driven hydraulic pump. This check valve prevents reverse flow through the pump when the external hydraulic power connections are being utilized.

b. Two check valves are installed in the case drain plumbing from the engine driven pump. These check valves provide a fluid path for cooling flow when the No. 3 engine is shut down and also protect the pump housing from entry of solid contaminants and debris.

29.2.2 OPERATIONAL CHECKOUT.

29.2.2.1 STANDBY (NO. 2) HYDRAULIC SYSTEM.

a. Start engine No. 3 according to the procedures in Chapter 72.

b. At engine idle, observe the dual-pressure indicator on the copilot's side of the main instrument panel.
RESULT: With no hydraulic subsystems operating, the pointer labeled "2" should show 3,000 (±150) PSI.

c. Shut down the engine.

d. Check the operation of the pump shutoff valves according to the following instructions:

1. Connect an external source of hydraulic power to the ground test connections for the standby hydraulic system according to the instructions in Chapter 12.

2. Connect 28-volt, DC power to the airplane and energize the 26-volt main AC bus.

3. Pull the fire emergency control handle for engine No. 3, and check the position of the standby system pump inlet shutoff valve. This valve is over the right hydraulic panel in the aft compartment.
   RESULT: The indicator on the pump inlet shutoff valve should indicate the closed position.

4. Slowly apply 3,000 PSI to the standby hydraulic system pressure line, using fluid from the power source reservoir, and observe the dual-pressure indicator on the main instrument panel.
   RESULT: The No. 2 pointer of the dual-pressure indicator should not be affected, indicating that the pump outlet shutoff valve is closed.

5. Reduce hydraulic pressure to zero, and then return the fire emergency control handle for engine No. 3 to the normal (in) position.

6. Slowly apply hydraulic pressure to the standby hydraulic system, and observe the dual-pressure indicator on the main instrument panel.
   RESULT: The No. 2 pointer of the indicator should show an increase in pressure, indicating that the pump outlet shutoff valve is open.

7. Check the position of the standby system pump inlet shutoff valve.
   RESULT: The valve should be open.

8. Reduce hydraulic pressure to zero, and disconnect the power sources from the airplane.

29.2.2.2 RESERVOIR PRESSURIZATION SYSTEM. For this procedure, refer to Section 29.1.2.3.

29.2.3 TROUBLESHOOTING.

29.2.3.1 STANDBY (NO.2) HYDRAULIC SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW OR FLUCTUATING HYDRAULIC PRESSURE INDICATION.		
Insufficient hydraulic fluid in the standby system reservoir.	Check the standby system for leaks. If no evidence of leakage is found, check the fluid level in the normal system reservoir. If the fluid level in the normal system reservoir is above normal, check the operation of the flap bypass and shuttle valves.	Retorque leaking connections. Repair or replace defective components. Service both reservoirs according to the instructions in Chapter 12.
Defective pressure indicating system.	Check the operation of the pressure transmitter and indicator.	Replace defective components as needed.
	Perform a continuity check of the system.	Repair the circuit as needed.
Pump cavitation.	Check the reservoir pressurization system for leaks.	Retorque, replace, or repair components as needed.
Defective high pressure relief valve.	Check the operation of the high pressure relief valve.	Replace the defective relief valve.
NO HYDRAULIC PRESSURE INDICATION.		
Defective pressure indicating system.	Check for defective or disengaged HYD PRESS SYS NO. 2 circuit breaker.	Engage disengaged circuit breaker; replace defective circuit breaker.
	Check that power is available at the 26-volt main AC bus.	Repair circuit as needed to obtain power.
	Check the operation of the pressure transmitter and indicator.	Replace defective components as needed.
Insufficient fluid in the standby system reservoir.	Check the standby system for evidence of leakage. If no leakage is found, check the normal system reservoir fluid level. If the normal system fluid level is too high, check the operation of the wing flap shuttle and bypass valves.	Repair, retorque, or replace leaking connections. Replace components as needed. Service both reservoirs according to the instructions in Chapter 12.
Pump inlet shutoff valve in the off position.	Check the position of the indicator on the inlet valve. If the valve indicates closed, and if the fire emergency control handle for engine No. 3 is in the normal (in) position, check the engine ignition and fire extinguishing system circuits.	Repair circuits as needed. Replace defective valve as needed.

29.2.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO HYDRAULIC PRESSURE INDICATION (Cont).		
Pump outlet shutoff valve in the off position.	Check the operation of the pump outlet shutoff valve. If the valve is closed and the fire emergency control handle for engine No. 3 is in the normal (in) position, check the engine ignition and fire extinguishing system circuits.	Repair circuits as needed. Replace defective valve as needed.
Inoperative engine-driven hydraulic pump.	Check the pressure filter for signs of pump failure. If there is no evidence of pump failure, check the pump drive.	Replace defective pump. Clean the system lines as necessary.

29.2.3.2 RESERVOIR PRESSURIZATION SYSTEM. Troubleshoot the reservoir pressurization system according to the procedures in Section 29.1.3.3.

29.2.4 BLEEDING PROCEDURES.

29.2.4.1 STANDBY (NO. 2) HYDRAULIC SYSTEM. Bleed the standby hydraulic system as follows:

a. Connect an external hydraulic power source to the standby hydraulic system according to the instructions in Chapter 12. Adjust the power source to deliver 250 PSI, and increase the pressure just enough to cycle the individual components.

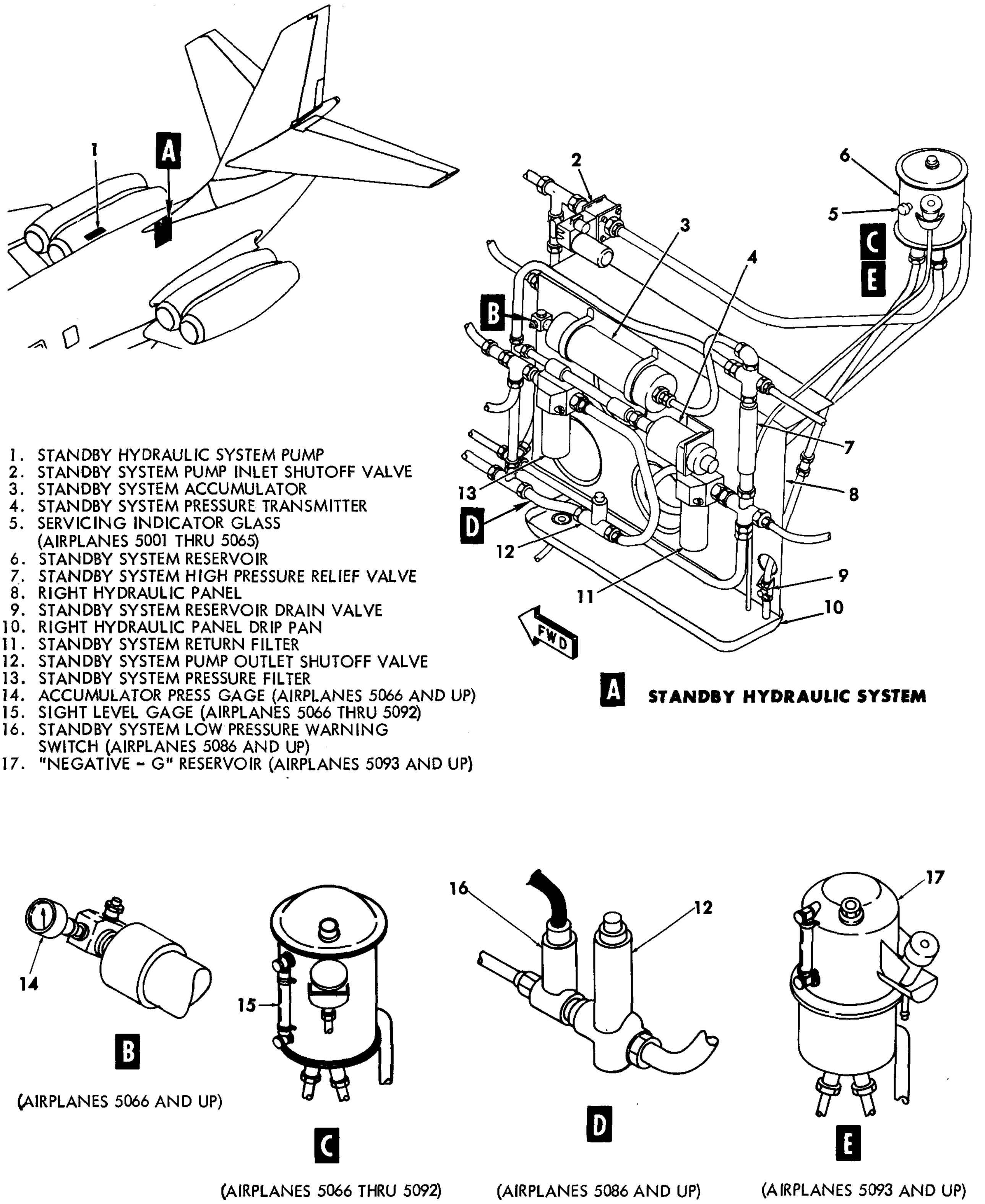
29.2.4.1 (Cont).

b. Connect 28-volt DC power to the airplane service connection.

c. Cycle the ailerons two complete cycles with the AILERON SYS 2 switch in the "ON" position.

d. Cycle the elevators two complete cycles with the ELEVATOR SYS 2 switch in the "ON" position.

e. Place the FLAP & BR HYD SYS switch in the "STANDBY-2" position, and actuate the wing flaps through two complete cycles.



1. STANDBY HYDRAULIC SYSTEM PUMP
2. STANDBY SYSTEM PUMP INLET SHUTOFF VALVE
3. STANDBY SYSTEM ACCUMULATOR
4. STANDBY SYSTEM PRESSURE TRANSMITTER
5. SERVICING INDICATOR GLASS (AIRPLANES 5001 THRU 5065)
6. STANDBY SYSTEM RESERVOIR
7. STANDBY SYSTEM HIGH PRESSURE RELIEF VALVE
8. RIGHT HYDRAULIC PANEL
9. STANDBY SYSTEM RESERVOIR DRAIN VALVE
10. RIGHT HYDRAULIC PANEL DRIP PAN
11. STANDBY SYSTEM RETURN FILTER
12. STANDBY SYSTEM PUMP OUTLET SHUTOFF VALVE
13. STANDBY SYSTEM PRESSURE FILTER
14. ACCUMULATOR PRESS GAGE (AIRPLANES 5066 AND UP)
15. SIGHT LEVEL GAGE (AIRPLANES 5066 THRU 5092)
16. STANDBY SYSTEM LOW PRESSURE WARNING SWITCH (AIRPLANES 5086 AND UP)
17. "NEGATIVE - G" RESERVOIR (AIRPLANES 5093 AND UP)

Figure 29-16. Standby Hydraulic System Components and Locations

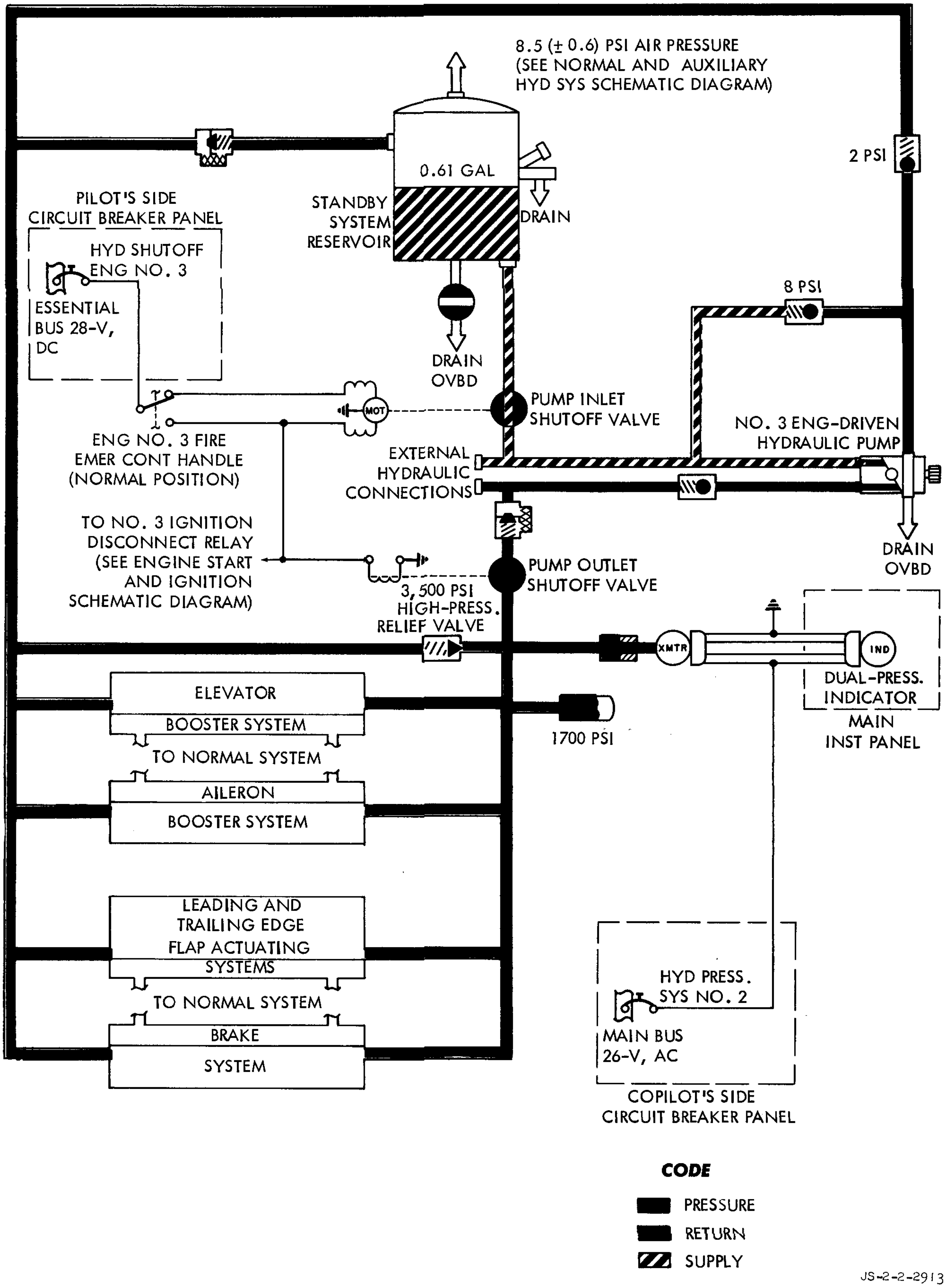
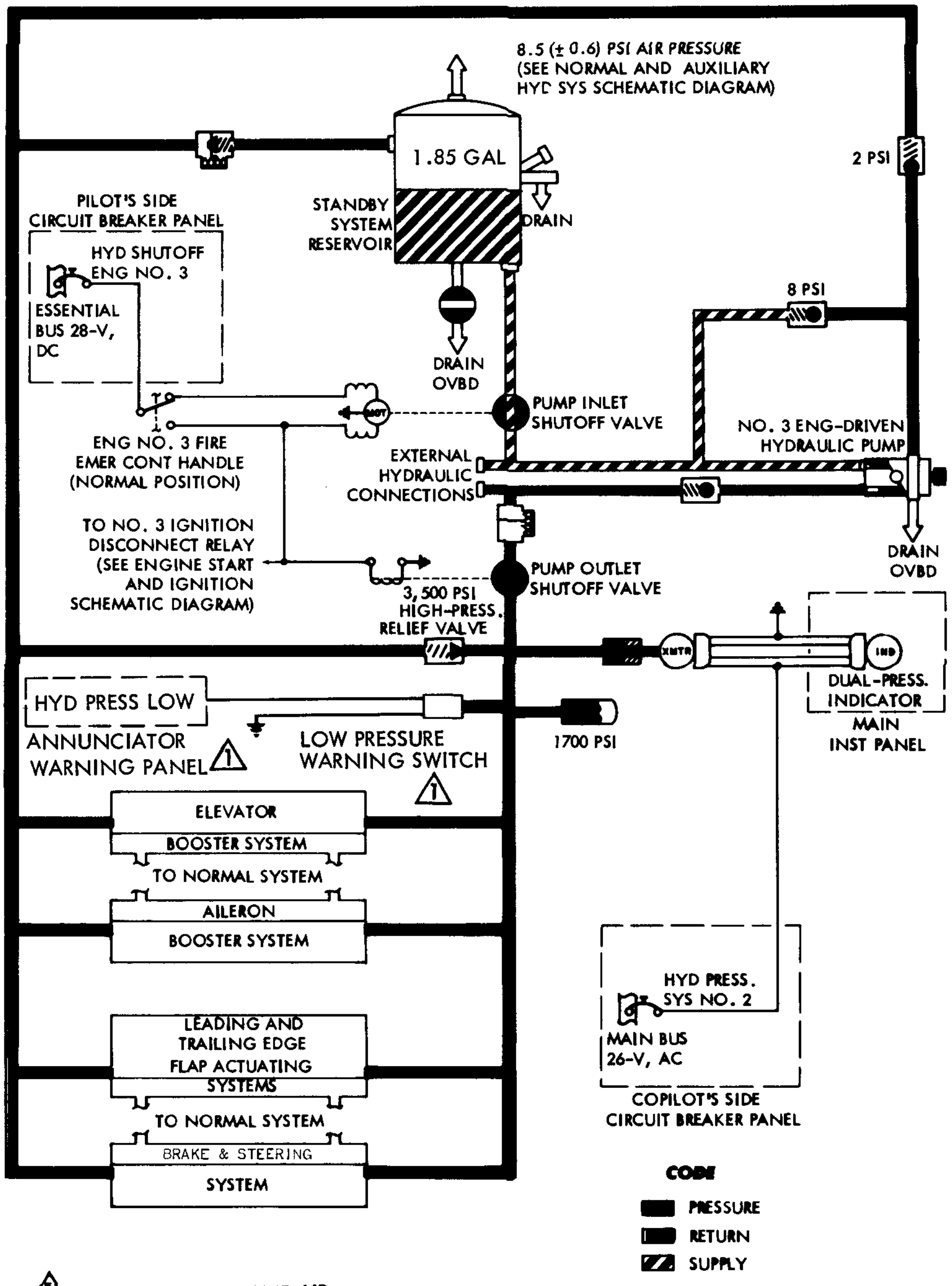


Figure 29-17. Standby Hydraulic System Schematic Diagram (Airplanes 5001 thru 5045)



△1 AIRPLANES 5086 AND UP

Figure 29-18. Standby Hydraulic System Schematic Diagram (Airplanes 5046 and Up)

CHAPTER 30
ICE AND RAIN PROTECTION

TABLE OF CONTENTS

Section		Page
30.0	GENERAL	30-3
30.1	WING AND EMPENNAGE DE-ICING SYSTEM	30-5
30.2	WINDSHIELD ANTI-ICING SYSTEM	30-19
30.3	WINDSHIELD WIPER SYSTEM	30-27
30.4	PITOT TUBE AND STATIC PORT ANTI-ICING SYSTEM	30-35
30.5	WINDSHIELD DE-FOG SYSTEM	30-39
30.6	CABIN WINDOW DE-FOG SYSTEM	30-41
30.7	ENGINE AND NACELLE ICE PROTECTION	30-43

Note

See Chapter 75 for ENGINE ANTI-ICING.

SECTION 30.0
GENERAL

30.0.1 PROTECTION PROVISIONS. Provisions for ice and rain protection on the JetStar include the following:

a. Wing and empennage de-icer boots are pneumatically operated by engine bleed air.

b. Windshield glass is electrically heated from power supplied by windshield inverter.

c. Pitot heads and static ports are electrically heated from power supplied by 28-volt DC main bus.

d. Dual windshield wipers are electrically operated with power supplied by 28-volt DC essential bus for one and 28-volt DC main bus for the other.

e. Windshield de-fogging is provided by air from the air conditioning and pressurization system.

f. Cabin window de-fogging is provided by an air vent system through desiccant tubes.

SECTION 30.1
WING AND EMPENNAGE DE-ICING SYSTEM

Contents Page

DESCRIPTION AND OPERATION 30-5
  General 30-5
  Boots 30-6
  Timer 30-6
  Distributor Valves 30-6
  Low-Vacuum Pressure Warning System 30-7
  De-Icer System Pressure Indicating System 30-7
  Ejector 30-7
  Pressure Regulator 30-7
  Wing and Empennage De-Icing Ground Air Connection 30-7

OPERATIONAL CHECKOUT 30-7
  Wing and Empennage De-Icing System 30-7

TROUBLESHOOTING 30-8
  Wing and Empennage De-Icing System 30-8

REMOVAL AND REPLACEMENT 30-9
  Removal of De-Icer Boots 30-9
  Replacement of Leading Edge External Fuel Vapor Seal 30-9
  Replacement of De-Icer Boots 30-10
  Removal of Timer 30-13
  Replacement of Timer 30-13
  Removal of Distributor Valves 30-13
  Replacement of Distributor Valves 30-13
  Removal of Low-Vacuum Pessure Switch 30-13
  Replacement of Low-Vacuum Pressure Switch 30-14
  Removal of Pressure Transmitter 30-14
  Replacement of Pressure Transmitter 30-14
  Removal of Ejector 30-14
  Replacement of Ejector 30-14
  Removal of Pressure Regulator 30-14
  Replacement of Pressure Regulator 30-14

BOOT REPAIR 30-15

ADJUSTMENT 30-15

PRESERVATION (CLEANING) 30-15

EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION 30-15

FIGURES
  30-1. Wing and Empennage De-Icing System Components 30-16
  30-2. Wing and Empennage De-Icing System Flow Diagram 30-17
  30-3. Wing and Empennage De-Icing System Electrical Schematic Diagram 30-18

30.1.1 DESCRIPTION AND OPERATION.

30.1.1.1 GENERAL. Ice accumulated in flight is removed from the leading edges of the wings and empennage by inflatable rubber de-icer boots. The engine bleed air system is used to produce both the pressure for inflating the boots and the vacuum for deflating them. The alternate inflation and deflation

30.1.1.1 (Cont).

of tubes in the boots causes an expanding and contracting action of the boot surfaces, which cracks the ice formation free from the boots and permits it to be carried away in the air stream.

a. A pressure regulator reduces bleed air pressure to provide the comparatively low positive pressure required to inflate the boots. An ejector, also operated by bleed air, produces the negative pressure used to deflate the boots. The positive and negative pressures are routed to electrically operated distributor valves for the wings and the empennage. Distributor valve operation is controlled by an electrical timer which contains all the operating controls for the system, both automatic and manual.

b. A small amount of bleed air is extracted from the bleed air manifold at a tee adjacent to the No. 3 engine bleed air supply line. A check valve in the line prevents reverse flow. The extracted bleed air is passed through a combined pressure regulator and safety pressure relief valve, which reduces the pressure, and the air is then divided into two flow paths. One is to the distributor valves; the other is to the ejector. The ejector produces a negative pressure by venturi action, which is also routed to the distributor valves. If all the bleed air shutoff valves are closed, de-icing pneumatic power is still supplied by a line tapped from the No. 3 engine bleed air supply line, upstream of the No. 3 bleed air shutoff valve. The bleed air leaving the engines is hot, and the de-icing supply is cooled by routing the line through the right ram air duct before it reaches the pressure regulator.

c. Each of the three distributor valves constantly receives a supply of positive-pressure air and negative-pressure air. Two of the distributor valves control the wing de-icing boots and one controls the empennage de-icing boots. The boots are divided into five separate groups. Reading from inboard to outboard, each wing half has four separate boots, designated A, B, B, and C boots. Each horizontal stabilizer half has a D boot, and the vertical stabilizer has an E boot. During normal operation, the two A boots on the wings inflate first, the four B boots second, and then C boots. On the empennage, the two D boots inflate together, and the E boot inflates separately. The timer controls this sequence automatically so that all boots are not inflated or deflated at the same time. The period during which the boots remain deflated between inflation cycles is variable, to deal with light icing conditions and heavy icing conditions. When icing is light, the system control switch on the timer is placed in the "LIGHT ICING" position; this provides a "rest" period of 153 seconds between inflation cycles. Under heavy icing conditions, the switch is placed in the "HEAVY ICING" position; this reduces the "rest" period to 33 seconds. The period during which boots remain inflated is the same for either mode of operation.

d. The timer also incorporates means for manual control of boot inflation, for use if automatic sequencing fails, consisting of five pushbuttons, one each for the A, B, C, D, and E groups of boots. The de-icing control switch can be in any position for manual operation. Boots remain inflated until the pushbutton is released during this mode of operation. Visual indication of boot sequencing and system activation is provided by means of indicator lights on the timer. A warning light is also provided for low suction pressure, and a gage for positive pressure indication. The timer forms a panel on the copilot's side console.

30.1.1.2 BOOTS. The de-icer boots are cemented to the leading edges. The boots are of conventional design and construction, with the inflatable tubes running in a chordwise direction for the wing and spanwise for the vertical and horizontal stabilizers.

30.1.1.3 TIMER. Automatic operation of the system is obtained by making and breaking the distributor valve circuits by means of switching cams driven by a 28-volt, DC electric motor in the timer. (See figure 30-3.) All controls for the system are also contained in the timer. A three-position ("LIGHT ICING," "OFF," and "HEAVY ICING") switch controls automatic operation. Five pushbutton switches, identified as A, B, C, D, and E, permit manual operation of the selected de-icing boots. A LOW SUCT WARN light provides indication of low vacuum, and a DE-ICER PRESS gage indicates system pressure. Five indicator lights, labeled A, B, C, D, and E, go on when the respective boots are inflated, during both manual and automatic modes of operation. Normal cycling of the timer provides an inflation period of approximately 6 seconds for all boots except the vertical stabilizer boot, which inflates for 3 seconds. Each timer contact is identified with the same letter as that designating the boots which it actuates when closed. The internal circuitry of the timer is so arranged that manual operation by the pushbuttons can be effected with the control switch in any position, including the "OFF" position, provided that power is available to the timer.

30.1.1.4 DISTRIBUTOR VALVES. One distributor valve is in the empennage, and one is in each wing root. Each valve controls operation of the adjacent de-icer boots. The valves are electrically operated by the timer. Each valve comprises two independent, solenoid-operated valves. Pressure and vacuum inlet ports of the distributor valves are connected respectively to the pressure and vacuum lines, and the pressure and vacuum outlet ports are connected to sections of de-icer boots. When the system is turned off, both the solenoids in each valve are de-energized to isolate the pressure line and connect the vacuum line to the boots, thus ensuring complete boot deflation, with no air trapped in the boots to cause distortion of the leading edge contour.

30.1.1.5 LOW-VACUUM PRESSURE WARNING SYSTEM. A pressure switch is tapped into the vacuum manifold in the right wing root fairing area. Contacts in the switch close if system vacuum drops to 2.5 inches of mercury, turning on the LOW SUCT WARN light on the timer. When the vacuum increases to 3.5 inches of mercury, the contacts open to turn off the light.

30.1.1.6 DE-ICER SYSTEM PRESSURE INDICATING SYSTEM. A pressure transmitter is tapped into the de-icer pressure manifold in the right wing root fairing area. The system obtains electrical power from the 26-volt, AC main instrument bus. The indicator is on the timer and shows system pressure in pounds per square inch.

30.1.1.7 EJECTOR. Suction pressure to deflate the boots is furnished by the ejector, which is in the fuselage aft compartment. The ejector provides vacuum equivalent to 6 inches of mercury at sea level. It receives pressure-regulated bleed air. This air is directed through a venturi, to produce vacuum, and is then discharged overboard.

30.1.1.8 PRESSURE REGULATOR. Bleed air pressure is reduced for use in the de-icing system by the pressure regulator. The air supplied by the regulator is at an operating pressure of 18 (±1) PSI. In the event of failure of the pressure regulating mechanism, pressure of the air supplied to the system is automatically limited to 22 (+1, −0) PSI by a pressure-relieving safety valve, which is an integral part of the pressure regulator. The pressure regulator assembly is in the fuselage aft compartment at approximately fuselage station 620.

30.1.1.9 WING AND EMPENNAGE DE-ICING GROUND AIR CONNECTION. A receptacle is provided in the aft compartment for connecting a ground source of compressed air to operate the de-icer system for checkouts. The receptacle is immediately upstream of the pressure regulator and adjacent to it. A chained cap closes the receptacle when it is not in use. The connection is for 5/8-inch outside diameter tubing or hose. Air from the ground source enters the system through the pressure regulator, bypassing the bleed air manifold.

30.1.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Equipment	Description	Use and Application
Air Compressor	Capable of delivering a flow of 30 SCFM and maintaining a pressure of 30 PSIG.	To furnish compressed air for the de-icing system.
Auxiliary Power unit	28V DC electrical power.	To energize de-icing circuits
Stop Watch		To time boot cycling.

30.1.2.1 WING AND EMPENNAGE DE-ICING SYSTEM. Two men will be required for this operational checkout. Man A will be stationed at the copilot's station to operate the controls; man B will be stationed outside the airplane to observe the operation of de-icer boots not visible to man A.

a. Connect the auxiliary power unit to the airplane external 28-volt, DC electrical power receptacle.

b. Man A: Close the WING & EMP DE-ICE PWR circuit breaker on the copilot's wainscot circuit breaker panel. Observe the LOW SUCT WARN light on the de-icing timer.
RESULT: The LOW SUCT WARN light should go on.

c. Energize the AC buses.

30.1.2.1 (Cont).

d. Close the WING & EMP DE-ICE PRESS IND circuit breaker on the copilot's wainscot circuit breaker panel. Observe the DE-ICER PRESS gage on the timer.
RESULT: The gage should indicate no pressure.

e. Man B: Set the air compressor controls for zero pressure, start the compressor, and connect it to the de-icing system ground test connection in the fuselage aft compartment. Slowly apply air pressure until 30 PSI is applied.

CAUTION

Never allow this pressure to exceed 100 PSI, as this might damage the ducts.

30.1.2.1 (Cont).

RESULT: Air should be discharged from the outlet on the right side of the fuselage at station 638. The LOW SUCT WARN light on the timer should go off. The DE-ICER PRESS gage on the timer should indicate 18 (±1) PSI.

CAUTION

If de-icer pressure is higher than 19 PSI, investigate the cause before proceeding with the checkout. Pressure above 30 PSI can damage the boots.

f. Man A: Place the de-icer control switch in the "LIGHT ICING" position, and observe the DE-ICER ON light.
Man B: Observe de-icer boot operation, and check cycling times with the stop watch.
RESULTS: The DE-ICER ON light should go on. The timer should begin to cycle, with alternate illumination of the amber lights on the timer. The boots should cycle in the following order, and inflation cycles should be as listed below:

Boot	Length of Cycle (seconds)
A	6
B	6
C	6
D	6
E	3

When the cycle is completed by inflation and deflation of the E boot, the boots should remain inactive for approximately 153 seconds, and then a second cycle should start by inflation and deflation of the A boots.

g. Man A: During the next cycle, interrupt automatic operation by pressing of each of the five manual control pushbuttons on the timer, and observe the amber lights.
RESULTS: The timer should continue to operate, but the automatic cycle should be interrupted. The amber light corresponding to the pushbutton pressed should go on. As the pushbuttons are released, the normal automatic cycle should resume.

h. Man A: Place the de-icing control switch in the "HEAVY ICING" position, and observe the DE-ICER ON light.
Man B: Observe de-icer boot operation, and check cycling times with the stop watch.
RESULTS: The results should be the same as in step (f), except that the inactive period between cycles should be approximately 33 seconds.

i. Repeat step (g), for the same results.

Note

During the test, only one amber light should be on at one time. The pressure gage reading should not exceed 19 PSI.

j. Man A: Place the de-icing control switch in the "OFF" position, and observe the timer lights.
Man B: Observe the de-icer boots.
RESULTS: All the timer lights should go off, indicating that the system is de-energized, and all the de-icer boots should deflate.

k. Man A: With the de-icing control switch still "OFF," press each of the five manual control pushbuttons on the timer, and observe the amber lights.
Man B: Observe the de-icer boots.
RESULTS: The light corresponding to the pushbutton pressed should go on, and the corresponding boots should inflate. When each pushbutton is released, the corresponding boots should deflate.

l. Shut down the auxiliary power unit and the air compressor, and disconnect them from the airplane. Install the cap on the de-icer ground test connection. Return the airplane to normal.

30.1.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ALL DE-ICER BOOTS DO NOT INFLATE WITH SYSTEM TURNED ON.		
Defective distributor valve, or valve not receiving power.	Check for correct input voltage to valve. If not available, check timer output.	Replace distributor valve, replace timer, or repair circuit, as required.

30.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
LOW-VACUUM WARNING LIGHT DOES NOT GO OUT WITH SYSTEM TURNED ON.		
Pressure switch defective or ejector outlet blocked, or defective pressure regulator.	Check for freedom of ejector air outlet. Check gage for correct system pressure.	Clear the air outlet, or replace pressure switch or pressure regulator, as required.
INCORRECT SYSTEM PRESSURE.		
Defective pressure regulator assembly, or leaking lines, or defective pressure indicating system.	If system pressure above 19 PSI and fluctuating with engine power changes, suspect the regulator. Check for loose plumbing connections. Check pressure indication system.	Replace regulator or repair plumbing, or replace pressure transmitter or indicator, or repair circuit, as required.
SYSTEM DOES NOT OPERATE WHEN TURNED ON.		
Defective timer or power supply to timer.	Check electrical power circuit to timer.	Replace timer or repair circuit.
ALL BOOTS DO NOT DEFLATE WHEN SYSTEM IS TURNED OFF.		
Defective distributor valve for affected boot, or valve suction solenoid still receiving power.	Check power supply to valve. If not available, check timer output voltage.	Replace distributor valve or timer, as required.

30.1.4 REMOVAL AND REPLACEMENT.

30.1.4.1 REMOVAL OF DE-ICER BOOTS. Before removing the boots from the three stabilizer leading edges and the wing leading edges, disconnect all of the flexible air hose from the air supply plumbing (see the access and inspection provisions for the location of the de-icer boot connection access panels). Remove the boots from the leading edges with the flexible hose connected to the de-icer boots. Use methylethylketone (MEK) to aid in removing the de-icing boots. Apply a minimum amount of this solvent to the seam line between the boot and the leading edge, and apply tension to peel back the boot. Peel the boot back slowly enough to allow the solvent to loosen the boot, to prevent injury to the part. When removing the de-icing boots and performing subsequent operations, observe the following precautions:

a. Do not allow the solvent to come in contact with the painted surfaces. The solvent will ruin the paint.

b. When working on the inboard wing leading edge, be careful not to damage the aluminum foil strips used for the external fuel seal. Refer to paragraph 30.1.4.2 for repair of this seal.

30.1.4.1 (Cont).

c. When working on the outboard wing leading edge (movable sections), be very careful not to damage the epoxy filler strips attached to the leading edge.

d. When removing the boot, be very careful not to contaminate the glue-covered surfaces of the leading edge and the boot also, if it is to be reused. After removal of boots, cover these surfaces with Kraft paper to keep them clean.

30.1.4.2 REPLACEMENT OF LEADING EDGE EXTERNAL FUEL VAPOR SEAL. The wing leading edge, inboard of the external fuel tanks, contains fuel and is externally sealed with sealant and aluminum foil prior to installation of the de-icer boots. If it becomes necessary to replace this external seal, do so by the following procedure using 3003 aluminum foil strips 0.002 inches thick by 3-1/2 inches wide:

a. Mask off the areas adjoining the lower spanwise leading edge skin splice. Leave 4 inches of open space between the masking tape with the splice centered in this space.

30.1.4.2 (Cont).

b. Mask off the areas adjoining the rib-to-leading-edge rivet pattern. Leave 2 inches of open space between the masking tape with the rivet pattern centered in this space.

c. Clean the areas enclosed by the masking tape, and both sides of the aluminum foil strips, using clean grease-free cloths and methylethylketone. Wipe the cleaned surfaces dry with clean, grease-free cloths.

Note

Use 3-1/2 inch wide strips for the skin splice and 1-3/4 inch wide strips for the rib rivet patterns. Use lengths sufficient to cover each entire splice or rivet pattern with one strip of foil.

d. Apply a uniform brush coat of 3M Co. EC1675, A-1/2 or Essex Chem. Co., Pro-Seal 890, A-1/2, or Goal Chem. Corp. GC408, A-1/2 sealing compound to the cleaned surfaces; fill gaps from mismatched skins in the process. The sealant thickness should be approximately 0.005 to 0.012 inch.

Note

Apply the sealant and foil to the spanwise splice first, and to the rib attachment rivets last.

e. Roll the aluminum foil down onto the sealing material immediately after the application of the sealant. The thickness of the foil and sealant after proper rolldown should be approximately 0.007 inch.

Note

The ends of the foil over the rib attachment rivets should overlap the foil on the spanwise splice at least 1 inch. The thickness of foil and sealant in the overlapped areas should be approximately 0.015 inch.

f. Apply a brush coat of sealant over all exposed edges of the aluminum foil, completely overlapping the edges a minimum of 1/4 inch. The overall thickness of the foil and sealant overcoat should be approximately 0.020 inch.

g. All tears in the foil, entrapped air bubbles, or loose areas shall be cut out and a patch of aluminum foil applied over the damaged area, using the same cement in Para. 30.1.4.2d or B.F. Goodrich No. A-851-B Buna-N cement. The repair foil shall extend a minimum of 1 inch over the undamaged area in all directions.

h. Allow the sealing compound to cure until it is tack-free before installing the de-icer boot.

Note

The cure time when using sealing compound is about 16 hours at a temperature of 77°F and 50 percent relative humidity. The cure time for foil strips repaired with Buna-N cement is a minimum of 2 hours between 60°F to 100°F, and 20 to 85 percent relative humidity.

30.1.4.3 REPLACEMENT OF DE-ICER BOOTS. This procedure is for installing new boots on new leading edges, and gives all steps necessary to prepare these surfaces for proper installation. If the leading edge has previously had a boot installed and the old glue is still evenly distributed and has been kept clean, omit steps (a), (b), (c), and (f), from this procedure. If the boot has previously been installed, and the old glue is still evenly distributed and has been kept clean, omit step (g) from this procedure.

WARNING

Provide adequate ventilation when using methylethylketone. Avoid breathing of vapor and minimize skin contact.

a. Clean the leading edge surface using a clean grease-free cloth dampened with methylethylketone.

Note

The container for the solvent must be free of grease, dirt, and other foreign materials prior to being filled with methylethylketone. Change cloths frequently to prevent recontaminating the cleaned surfaces.

b. Reclean the leading edge using a cleaned container and fresh methylethylketone. Wipe the cleaned surface dry with a clean dry cloth.

Note

During the cleaning operations, use only clean cloths.

CAUTION

Be very careful not to permit the cleaning solvent to seep through the skin seams into the inner structure. Do not remove any of the sealant applied in the externally sealed areas.

c. Scrub the metal surfaces vigorously using clean, grease-free cloths dampened with hydrofluosilicic acid cleaner, (Leeder 162-D Wipe Solvent, Leeder Chemicals Inc., La Minada, CA 90638) or equivalent. Wipe the surface dry within one minute, using clean, grease-free cloths. Allow the cleaner to evaporate fully (1 hour minimum time at room temperature). A forced, filtered, warm-air dryer may be used to evaporate the cleaner in approximately 15 minutes.

30.1.4.3 (Cont).

WARNING

Use rubber gloves when working with this acid cleaner. This cleaner is harmful to the skin. If this cleaner should contact the skin or eyes, immediately flush the area with water.

Note

Use only capped, polyethylene containers for this acid cleaner. This acid will react with either metal or glass.

d. Apply one brush coat or equivalent spray coats of general purpose chloroprene cement (Minnesota Mining and Manufacturing Co. EC 1300L or Scotchgrip 1300L) uniformly to the leading edge surface.

Note

Examine the cement for condition, immediately before using, by opening the container and stirring the cement with a stick or rod. Remove the stick or rod rapidly from the cement; if the cement is stringy or drips it is acceptable; if the cement snaps, will not flow, or has jelled, it is unacceptable.

e. Allow the coating of cement to dry one hour minimum. Drying times longer than one hour may be required if the relative humidity exceeds 65 percent.

Note

If the remainder of the cementing operation does not immediately follow the first coating, give the cemented surface adequate protection from dirt, water, and grease. Cover the surface with clean Kraft paper.

f. After the minimum cure time has elapsed, test the cement for proper adhesion. Use pressure sensitive tape (PPP-T-60, Type II, Class I; Permacel Product No. 691; Polyken Product No. 231 or PPP-T-60, Type III, Class 2; 3M Co. Product No. 710; Permacel Product No. 99) approximately 3/4 inch wide. Place an 18- to 20-inch length of tape along the cemented surface, and press firmly in place. Jerk the tape free and observe the area tested for peeling of the cement.

RESULT: If the tape test indicates more than 10 percent of the total tested area is defective, remove the cement completely, and reapply fresh cement to the entire area. If the test indicates less than 10 percent of the cement is defective, clean the peeled areas and reapply cement to those areas only.

Note

Pay particular attention during the tape test to the edges of the cemented areas and to the splice joints and rivet heads.

g. Clean the rough back surface of the rubber boot with a clean, grease-free cloth dampened with methylethylketone. Perform this cleaning operation over the entire surface at least three times.

WARNING

Provide adequate ventilation when using methylethylketone (MEK). Avoid prolonged breathing of vapors and minimize skin contact. Failure to comply may result in injury to personnel.

CAUTION

Avoid excessive use of the cleaner around the cemented grommets. The cleaner tends to soften the cement if used too freely.

Note

If the boot is received with a smooth finish on the bonding surface, roughen the surface carefully by mechanical means before cleaning. Be careful not to remove the center line marked on the boot.

h. Apply one brush coat of cement (Minnesota Mining and Mfg. Co. EC 1300L or Scotchgrip 1300L) to the cleaned surface of the boot, and allow it to cure for at least one hour minimum.

Note

Keep the cement coated surface clean by covering loosely with clean Kraft paper until the boot is ready for installation.

i. Apply a second brush coat, or equivalent spray coats, of the cement uniformly over the leading edge surface and over the rough surface of the boot.

j. Allow the cement on both surfaces to air dry. This will require approximately one hour.

Note

Attach the short flexible air hose to the wing and empennage boots before installing the boots on the leading edges. These connections are secured by small metal clamps. Attach the other end of these hose to the air supply system after the boots are installed. (See the access and inspection provisions for the deicing boot connection access panels in Chapter 52.)

k. Line up the centerline of the boot with the centerline of the leading edge.

Note

For the wing leading edge outboard of the external fuel tanks, center the protruding manifold of the boot into the recess on the leading edge of the flap.

30.1.4.3 (Cont)

l. Reactivate the cement on both surfaces, using a clean cloth dampened with methylethylketone or toluene (toluene preferred). Since the de-icer boot is installed by working from the centerline, it is recommended to reactivate only a 6-inch wide strip spanwise along the boot at one time.

Note

Do not use cloths that are too damp, and do not rub the cemented surfaces excessively. This may remove the cement.

m. Use finger pressure to press the surfaces together as the cement is reactivated. If the boot should attach off course, pull it up quickly, and reset properly. Avoid twisting or sharp bending of the boot. Avoid trapping air under the boot as it is applied.

n. Roll the boot firmly in place with a hand roller, first along the centerline, and then outward from the centerline to the edge of the boot. Use strokes perpendicular to the centerline, spaced about 6 inches apart.

Note

For this rolling operation, use a roller of soft rubber approximately two inches wide and 1-1/2 inches in diameter. (B.F. Goodrich Aviation Product P/N 74-451-74.)

o. Repeat the rolling operation to roll the untouched areas.

p. Roll the entire surface of the boot, applying firm pressure.

q. Roll all edges and areas between the air cells firmly with a narrow stitcher roller.

Note

For this rolling operation, use a metal roller approximately 1/4 inch wide and 1-1/2 inches in diameter. (B.F. Goodrich Aviation Product P/N 2D1027.)

r. Apply a thin coat of cement over all exposed, cut, boot edges.

s. If an air pocket or blister is encountered, carefully insert a hypodermic needle (22 gage maximum) into the blister and allow the air to escape so the boot can seal completely. If the blister is within the inflatable tube area, insert the needle between the tubes, being careful not to puncture the tubes.

t. Perform the static test outlined in step (u) on the de-icing boot a maximum of 1/2 hour after installation using methylethylketone, or 4 hours after installation using toluene.

u. Use a 2-inch diameter suction cup attached to a spring scale and apply a 25 pound pull force to the boot at the following locations:

(1) All four corners within one inch of cut edges.

(2) Every 12 inches along the outside edges.

(3) Enough of the cell areas to make sure no individual voids or cement laminations are over 2 square inches in area.

Note

Do not place the suction cup directly over a cell, but place the cup so the cell separator line lies under the center line of the cup.

v. Re-cement and retest any voids or delaminations exceeding the limit of 2 square inches.

w. Perform the following pressure test on the deicing boot a minimum of 1 hour after installation using methylethylketone, or 8 hours after cement activation in paragraph l above.

x. Energize the main DC bus with 28-volt DC, the instrument bus with 26-volt AC, and engage the de-icer system circuit breaker and the de-icer system pressure indicator circuit breaker on the copilot's circuit breaker panel.

y. Pressure-test the de-icer system by inflating the system to 10 PSI through 6 complete cycles.

Note

If adequate ground support equipment is available, perform this test using external power for both, the electrical and pneumatic requirements (a ground test air receptacle for the de-icer system is located in the aft compartment). If the necessary ground support equipment is not available, the test may be performed by operating the de-icer system in the normal manner, using the airplane's power; however, if airplane power is used, the cure times given above must be increased to 8 hours for boots installed using methylethylketone, and to 48 hours for boots installed using toluene.

30.1.4.3 (Cont).

z. Return the system to normal.
RESULT: Check the boot for improper adhesion. A maximum of 2 square inches of improperly cemented area per square foot is permissible. If this is exceeded, this section of the boot must be removed and recemented. Also, no loose areas are permissible within 2 inches of any edge of the boot. The entire boot need not be removed, only the areas needing repair.

aa. For the wing leading edge flap boots, fill gaps and seams around the edges or between adjacent boots with smoothing compound (Minnesota Mining and Mfg. Co., No. EC801, B-1/2 or Product Research Co. PR380-M or Essex Chem. Co. Pro-Seal 247). It is not necessary for the compound to extend over the edges of the boots. Fair this compound in accordance with steps (ab) through (af).

ab. Apply a thin masking tape along the edges of the boots and on the metal leading edge surface approximately 3/8 inch from the boot edge.

ac. Apply a fillet of the smoothing compound between the masking strips, and cover with a strip of polyethylene tape 0.001 to 0.003 inch thick.

ad. Press or roll the tape with a spatula to produce a smooth, tapered joint.

30.1.4.3 (Cont).

ae. Remove the polyethylene tape and masking tape after the smoothing compound has been allowed to cure (cure time is normally about 24 hours at 77°F).

af. If necessary, the fillet may be smoothed further by sanding with 400-grit sandpaper.

CAUTION

Be very careful not to scratch the aluminum skin surface.

ag. For the inboard wing and empennage, seal and fair all boot trailing edges with sealing compound (Minnesota Mining and Mfg. Co., No. EC-801-B-1/2 or Products Research Co., No. PR-380M). The sealant should overlap the boot edge approximately 1/4 inch. Fair the sealant so as to finish with a smooth taper.

Note

Apply the sealant as soon as reasonably possible after boot installation. Remove any tape used in masking the area for this fillet within 1/4 hour after the sealant is applied.

CAUTION

Never allow the boot edges to be exposed to outdoor weathering unless this sealant overcoat is applied first.

ah. After the boot trailing edges have been sealed and cured, clean approximately the last 1 inch of the boot trailing edges, over the sealed edges and approximately 1 inch of the adjacent metal surface, using clean, grease-free cloths dampened with methylethylketone.

ai. Apply a brush coat of conductive rubber preservative, Type R-8-N-C-186 (Willard Tire Co., Woodland Hills, CA, Phone 818-346-7800), approximately 0.010 inch thick, over the cleaned area. An alternate conductive coating is available which requires a primer for use on the adjacent 1-inch metal surface cleaned above. The metal surface only should be primed with Lord 9924 Chemglaze wash primer followed by an application of Lord L300 Chemglaze conductive flexible coating over the 2-inch-wide band cleaned in paragraph ah. These applications should be made in accordance with the manufacturer's instructions (Lord Mfg. Co., Erie, PA 16514-0038, Phone 814-868-3611).

aj. If it is necessary to remove excessive cement, wipe it briskly with a clean cloth moistened with methylethylketone.

CAUTION

Do not attempt to remove cement nearer the boot edge than 1/4 inch. Do not allow the solvent to run under the edges of the boot or to wet the surfaces of the boot.

30.1.4.4 REMOVAL OF TIMER.

a. Disengage Dzus fasteners attaching the timer assembly to the copilot's side panel.

b. Raise the timer assembly enough to gain access to the electrical connector. Disconnect the wire harness, and remove the timer.

30.1.4.5 REPLACEMENT OF TIMER.

a. Replace the timer by reversing the removal procedure.

b. Perform an operational checkout of the wing and empennage de-icing system.

30.1.4.6 REMOVAL OF DISTRIBUTOR VALVES.

a. Obtain access to the affected valve.

Note

The empennage distributor valve is in the top of the aft compartment, between the frames at fuselage stations 659.5 and 666.0. The wing distributor valves are above the respective fuselage belly fairings between fuselage stations 410 and 416, inboard of the leading edge root fairings.

b. Disconnect the lines and electrical harness from the valve.

c. Disassemble the four screws and washers attaching the valve to the brackets between which it is mounted. Withdraw the valve downwards.

30.1.4.7 REPLACEMENT OF DISTRIBUTOR VALVES.

a. Position the valve between the support brackets, and attach it to the brackets with four screws and washers. Secure the screws with 0.032-inch diameter corrosion-resistant steel lockwire conforming to Specification QQ-W-423, Form I, Comp. FS302, Condition A.

b. Connect the lines to the valve ports with the hose and clamps previously removed. Note that the left wing distributor valve has a hose and plug connector to port P-2.

c. Perform an operational checkout of the system.

30.1.4.8 REMOVAL OF LOW-VACUUM PRESSURE SWITCH.

a. Remove the right fuselage belly fairing door to gain access to approximately fuselage station 423 inboard of the wing root forward fairing.

b. Disconnect the vacuum line and the electrical wiring from the pressure switch. Remove two nuts, two screws, and four washers attaching the switch to the bracket, and remove the switch.

30.1.4.9 REPLACEMENT OF LOW-VACUUM PRESSURE SWITCH.

a. Attach the switch to the bracket with two screws, four washers, and two nuts. Connect the vacuum line and the electrical wiring to the switch.

b. Perform steps (a) through (e) of paragraph 30.1.2.1 to check switch operation.

30.1.4.10 REMOVAL OF PRESSURE TRANSMITTER.

a. Remove the right fuselage belly fairing door to gain access to approximately fuselage station 423 inboard of the wing root fairing.

b. Disconnect the pressure line and the electrical wiring from the transmitter.

c. Remove the four screws attaching the transmitter flange to the bracket.

d. Withdraw the transmitter downwards.

30.1.4.11 REPLACEMENT OF PRESSURE TRANSMITTER.

a. Position the transmitter below the bracket, and attach it with four screws.

b. Connect the pressure line and the electrical wiring to the transmitter.

c. Perform steps (a) through (e) of paragraph 30.1.2.1 to check transmitter operation.

30.1.4.12 REMOVAL OF EJECTOR.

a. Remove the clamp securing the ejector to the fuselage frame at station 630.25.

b. Disconnect the overboard discharge line from the aft end of the ejector by removing the two clamps and the connecting hose.

c. Disconnect the tubes from each end of the tee in the ejector vacuum port.

d. Disconnect the tube from the forward end of the ejector, and remove the ejector.

e. Remove the tee from the ejector vacuum port.

30.1.4.13 REPLACEMENT OF EJECTOR.

a. Install the tee in the vacuum port of the ejector with a serviceable O-ring. The tee should be aligned with the forward and aft centerline of the ejector as installed in the airplane.

b. Position the ejector in the airplane. Connect the lines to each end of the tee, using serviceable O-rings.

c. Connect the pressure line to the forward end of the ejector, and connect the aft end of the ejector to the overboard discharge line with the hose and two clamps.

d. Secure the ejector to the frame at fuselage station 630.25 with the clamp and one screw.

e. Perform steps (a) through (e) of paragraph 30.1.2.1 to check ejector operation.

30.1.4.14 REMOVAL OF PRESSURE REGULATOR.

a. Disconnect the bleed air line from the ground test connection tee, which is connected to the forward end of the pressure regulator. Remove the cap from the ground test connection.

b. Disconnect the remaining lines from the pressure regulator.

c. Disassemble the pressure regulator from the support bracket by removing three attaching screws. Remove the pressure regulator.

d. Remove the ground test connection and bleed air supply tee from the regulator.

30.1.4.15 REPLACEMENT OF PRESSURE REGULATOR.

a. Install the ground test connection and bleed air supply tee in the regulator inlet port with a serviceable O-ring. Position the tee so that the ground test connection is aligned with the vertical centerline of the regulator as installed in the airplane.

b. Position the regulator in the airplane, and attach it to the structure with three screws. The short screw is used at the attachment point farthest aft.

c. Connect the bleed air line to the ground test connection and bleed air supply tee. Install the cap on the ground test connection. Connect the remaining lines to the respective regulator ports.

d. Perform steps (a) through (e) of paragraph 30.1.2.1 to check regulator operation.

30.1.5 BOOT REPAIR. A cold patch repair kit No. 74.451-C is available from B. F. Goodrich Co., Akron, Ohio for an effective temporary repair of minor damage. For major repairs, the de-icer boot should be returned to B. F. GOODRICH Co., for vulcanizing.

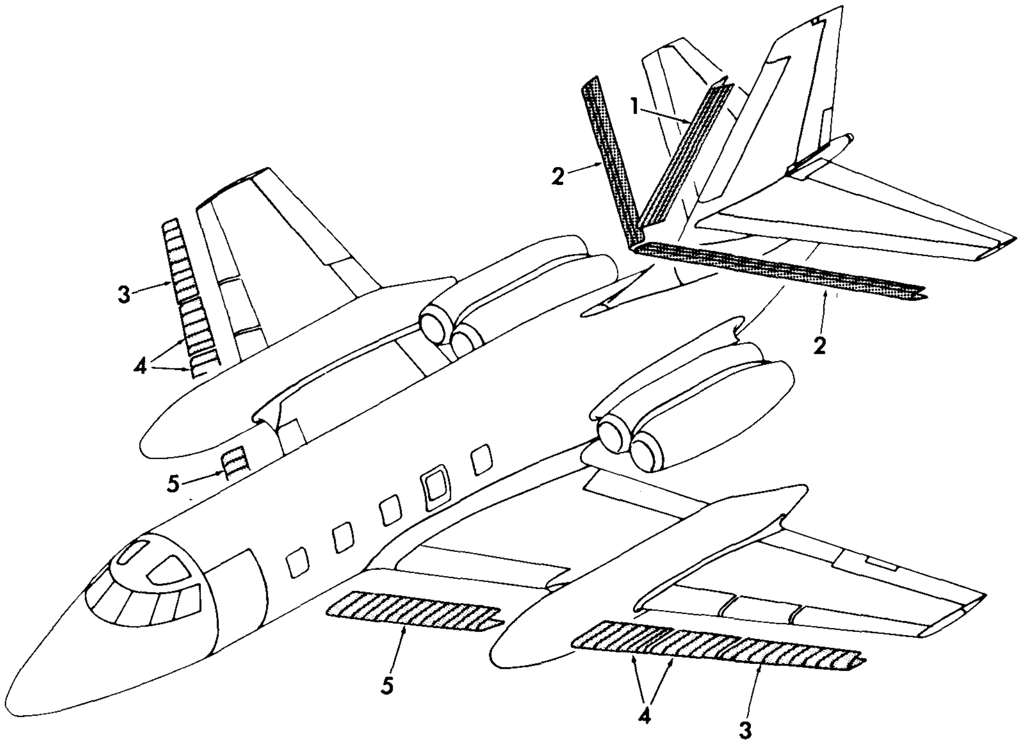
30.1.6 ADJUSTMENT.

30.1.6.1 GENERAL. If boot timer operation becomes erratic, or system pressure or vacuum is incorrect, the defective unit should be isolated and replaced with a serviceable like item. The removed unit should be returned to an authorized overhaul facility.

30.1.7 PRESERVATION (CLEANING).

30.1.7.1 Clean the de-icer boots with a mild detergent soap and water. Rinse throughly with clean water.

30.1.8 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. The wing and empennage de-icing system does not require any special maintenance procedures under extreme operating conditions. Normal routine inspections and observation of system operation should reveal any need for maintenance work on the system. Water/glycol solutions may be used as a de-icing fluid to de-ice the aircraft on the ground. It has been found, however, that glycol in the presence of silver (such as might be found in wiring or electrical connectors and contacts) with electrical current can produce an explosive ignition. To prevent ignition, add 0.5% by weight of Benzotriazole (BZT) or Tolyltriazole to the water/glycol solution. BZT may be obtained from Eastman Kodak Co., Eastern Chemical (A Division of Guardian Chemical Corp.), or J.T. Baker Chemical Co. in Phillipsburg, N.J.



1. E BOOT
2. D BOOT
3. C BOOT
4. B BOOT
5. A BOOT

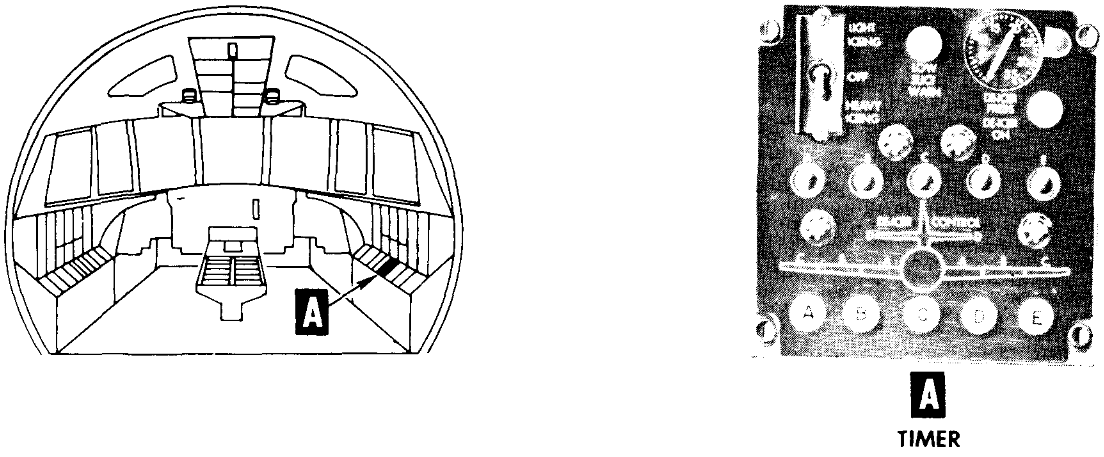
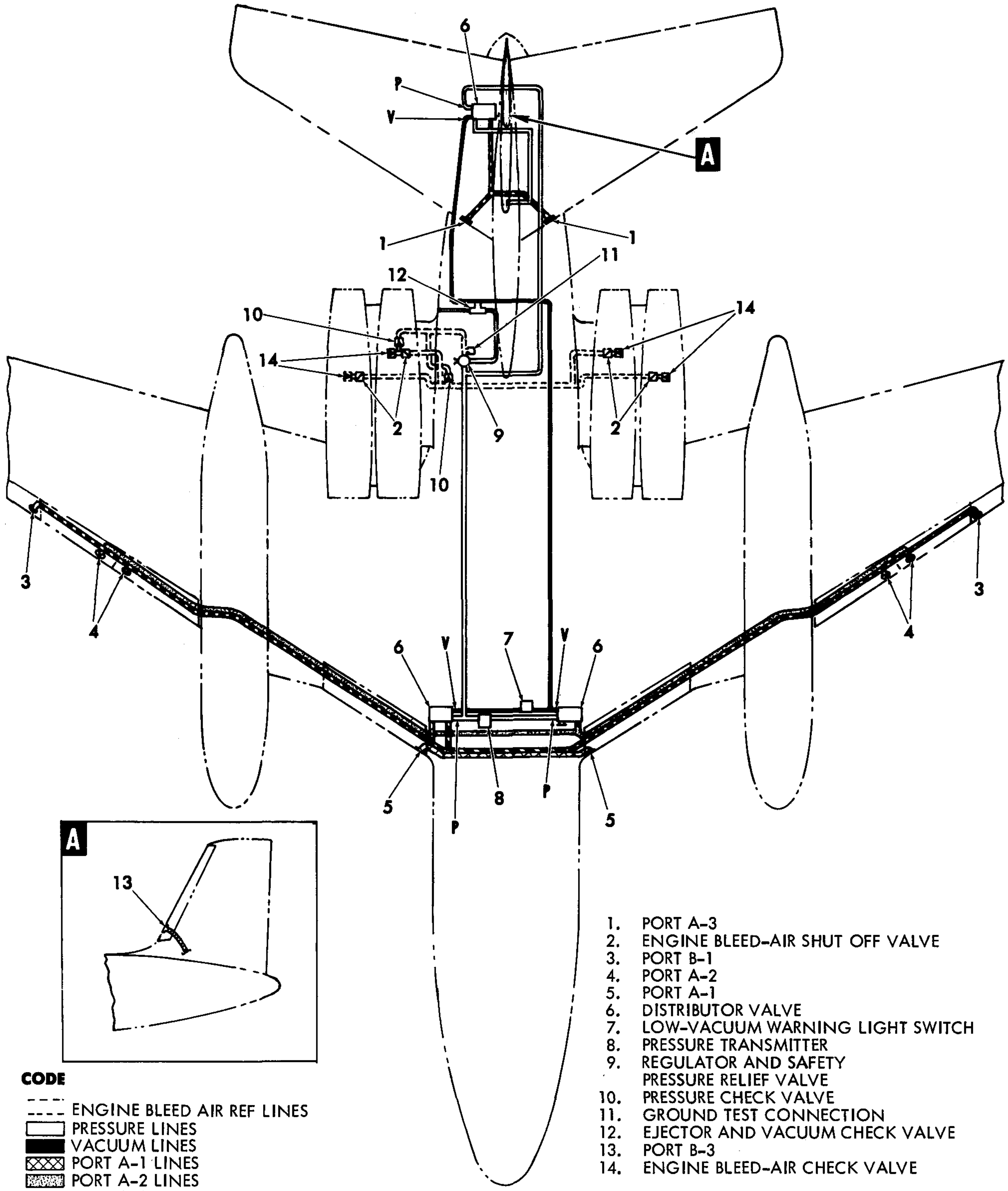


Figure 30-1. Wing and Empennage De-Icing System Components



1. PORT A-3
2. ENGINE BLEED-AIR SHUT OFF VALVE
3. PORT B-1
4. PORT A-2
5. PORT A-1
6. DISTRIBUTOR VALVE
7. LOW-VACUUM WARNING LIGHT SWITCH
8. PRESSURE TRANSMITTER
9. REGULATOR AND SAFETY PRESSURE RELIEF VALVE
10. PRESSURE CHECK VALVE
11. GROUND TEST CONNECTION
12. EJECTOR AND VACUUM CHECK VALVE
13. PORT B-3
14. ENGINE BLEED-AIR CHECK VALVE

CODE

- - - ENGINE BLEED AIR REF LINES
☐ PRESSURE LINES
■ VACUUM LINES
▩ PORT A-1 LINES
▦ PORT A-2 LINES
▥ PORT A-3 LINES
▧ PORT B-1 LINES
▨ PORT B-3 LINES
V VACUUM MANIFOLD ON DISTRIBUTOR VALVE
P PRESSURE MANIFOLD ON DISTRIBUTOR VALVE

Figure 30-2. Wing and Empennage De-Icing System Flow Diagram

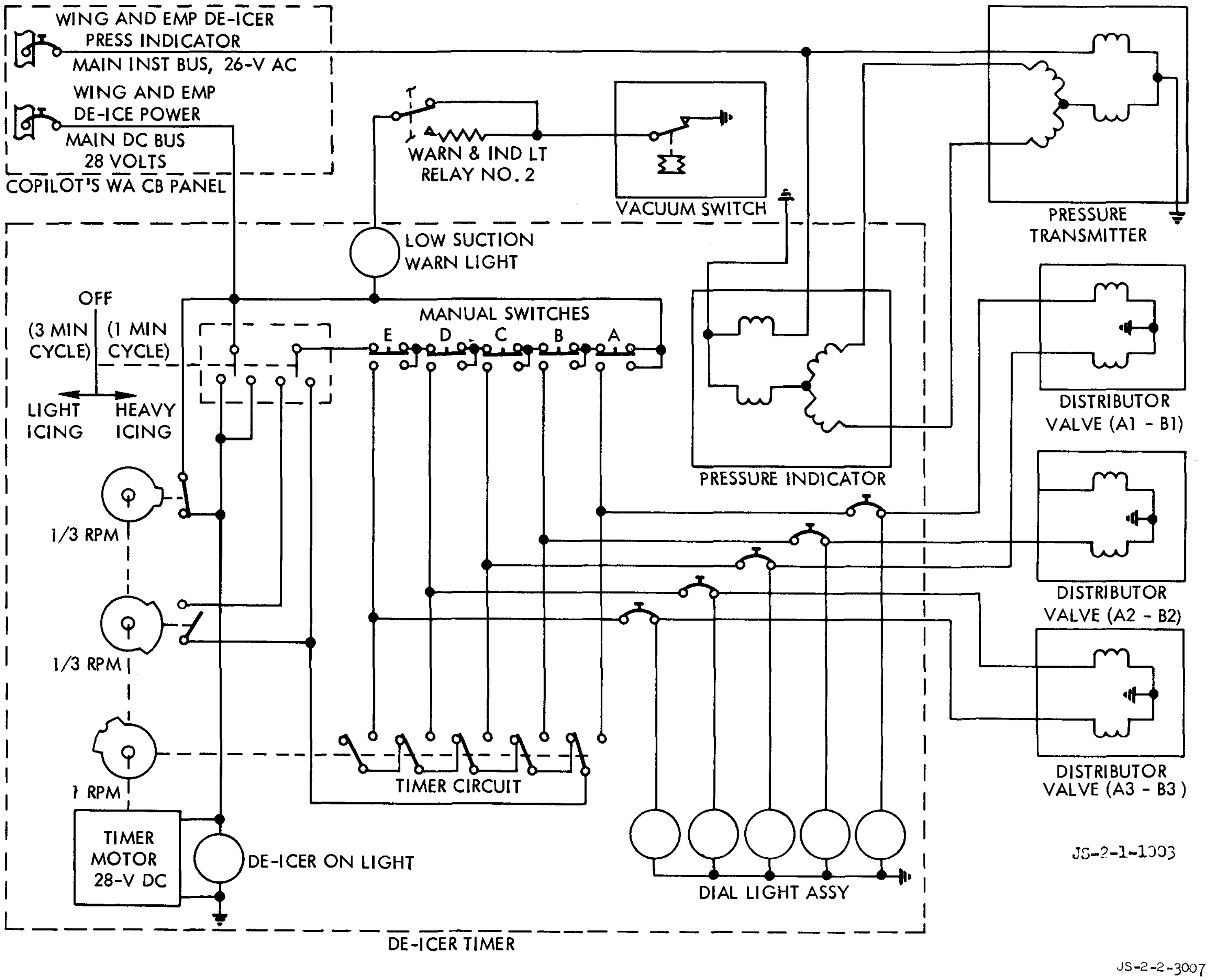


Figure 30-3. Wing and Empennage De-Icing System Electrical Schematic Diagram

SECTION 30.2
WINDSHIELD ANTI-ICING SYSTEM

Contents **Page**

DESCRIPTION AND OPERATION 30-19
- General 30-19
- Windshield Anti-Icing Control Panel 30-19
- Windshield Temperature Control Boxes 30-20
- Windshield Power Transformer 30-20
- Windshield Emergency Transformer 30-20
- Windshield Panels 30-20

OPERATIONAL CHECKOUT 30-20

TROUBLESHOOTING 30-21
- Failed Thermistor 30-22

REMOVAL AND REPLACEMENT 30-22
- Removal and Replacement of Windshield Panels 30-22
- Pilot's and Copilot's Windshield Panels (JF31-7L/H) 30-22A
- Aft Side Windshield Panel Thermostat 30-22A

CLEANING 30-22A

EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION 30-22A

FIGURES
- 30-4. Thermistor Resistance Chart 30-23
- 30-5. Windshield Anti-Icing System Components 30-25
- 30-6. Windshield Anti-Icing System Electrical Schematic Diagram 30-25
- 30-6A. Pilot's and Copilot's Windshield Panels 30-26A

30.2.1 DESCRIPTION AND OPERATION.
(See figures 30-5 and 30-6.)

30.2.1.1 GENERAL. Ice is prevented from forming on the outer surfaces of the center windshield, the three pilot's windshields, and the three copilot's windshields by electrical heating of the glass. A conductive, transparent layer in each laminated windshield panel carries the heating current. Electrical power is obtained from the inverter system through a step-up transformer. The temperature of the pilot's clear vision windshield, the pilot's forward windshield, and the center windshield is controlled during normal heating by a bridge circuit, consisting of a thermistor in the pilot's forward windshield and components in the left-hand temperature control box. The temperature of the copilot's clear vision and forward windshields is similarly controlled during normal heating by a second bridge circuit, consisting of a thermistor in the copilot's forward windshield and components in the right-hand windshield temperature control box. The temperature of these panels is maintained, during normal operation, in the range of 90° to 105°F. A thermal switch on each aft windshield controls its temperature in the range of 70° to 85°F.

30.2.1.2 WINDSHIELD ANTI-ICING CONTROL PANEL. (See figure 30-5.) System operation is controlled from a windshield anti-icing control panel. This panel includes a three-position ("NORM," "OFF," "EMER") WINDSHIELD ANTI-ICING PWR control switch, a two-position ("LOW" and "HIGH") WINDSHIELD ANTI-ICING HEAT control switch, and a LOW AMBIENT START pushbutton. The WINDSHIELD ANTI-ICING PWR control switch is guarded in the "EMER" position. The "NORM" position is for the normal system operation and is used when any two of the inverters are operative. Under these conditions, all the windshield panels are heated, provided the WINDSHIELD ANTI-ICING HEAT control switch is in the "LOW" position. If very severe icing conditions exist, this switch is placed in the "HIGH" position to apply higher voltage to the pilot's and copilot's forward windshield, while not heating the aft windshields. If the windshield inverter becomes inoperative, the windshield heat system can be powered from the standby inverter by placing the WINDSHIELD INV switch, on the overhead AC control panel, in the "TRANSFER" position. This transfers all of the windshield heat loads from the windshield inverter to the standby inverter. When the standby inverter is operating in place of the main inverter and the windshield inverter becomes inoperative,

30.2.1.2 (Cont).

only the pilot's windshield can be heated. This is accomplished by placing the WINDSHIELD ANTI-ICING PWR control switch in the "EMER" position. This then powers the pilot's forward windshield from the essential AC bus, through the emergency windshield transformer. If only the windshield inverter is operative, placing the AC SYS switch in the "EMER" position similarly provides heat for only the pilot's forward windshield from the essential AC bus. (Refer to sections 24-1 and 24-2.) The LOW AMBIENT START pushbutton is only used at ambient temperatures below -55°F. At these temperatures, the resistance of the thermistors becomes high enough to represent an open circuit, and is thus outside the controlling range of the system. Pressing the pushbutton applies power directly to the control boxes to heat the windshields. This, in turn, heats the thermistors and lowers their resistance, to enable the automatic control system to function. The LOW AMBIENT START pushbutton is operative only when the WINDSHIELD ANTI-ICING HEAT control switch is in the "LOW" position.

30.2.1.3 WINDSHIELD TEMPERATURE CONTROL BOXES. The two windshield temperature control boxes control the temperature of the windshields by regulating the length of time during which heating current is applied to the windshields. When an undertemperature condition is sensed by a control box, a corresponding windshield temperature control relay is energized, thereby applying power to the appropriate windshields.

30.2.1.4 WINDSHIELD POWER TRANSFORMER. During normal operation of the system, a windshield power transformer is used to supply electrical power for heating the panels. This is an autotransformer with a 115-volt, 400-cycle primary, and a multi-tapped secondary for windshield resistance matching. Also, certain taps are used for high-heat operation and others for low-heat operation.

30.2.1.5 WINDSHIELD EMERGENCY TRANSFORMER. During emergency operation of the system, when only the pilot's forward windshield is heated, power is supplied by a windshield emergency transformer. This is an autotransformer with a 115-volt, 400-cycle primary and a high-voltage (330-volt) secondary which furnishes heating voltage for the pilot's forward windshield during emergency operation.

30.2.1.6 WINDSHIELD PANELS. The seven windshield panels are laminated. The five forward panels consist of an inner layer of glass, a stannic oxide coating, a layer of vinyl plastic, a middle layer of glass, a second layer of vinyl plastic, and an outside layer of glass. The aft side panels consist of an inner layer of glass, a stannic oxide coating, a layer of vinyl plastic and an outside layer of glass. The heat is produced by the resistance of the stannic oxide coating when electric current is passed through it.

30.2.2 OPERATIONAL CHECKOUT.

Note

If the ambient temperature is above the low limit control point of the windshields it will be necessary to cool the glass in order to operate the system. This may be accomplished by applying ice packs to the windshields or by blowing refrigerated air directly on the windshields. Dry ice is not recommended.

CAUTION

When testing the anti-icing system, do not allow any windshields to overheat. If a windshield becomes too hot to touch with bare hands, turn off the control switch and correct the trouble.

a. Connect external power to the airplane and place the windshield inverter switch in the "ON" position.

b. Engage the WINDSHIELD INV CONTROL circuit breaker on the copilot's console circuit breaker panel.

c. Engage the EMER WINDSHIELD HEAT circuit breaker and the following WINDSHIELD HEAT CONTROL circuit breakers on the pilot's wainscot circuit breaker panel: LH ACTUATOR, RH ACTUATOR, LH BRIDGE, RH BRIDGE, and RELAYS.

d. Engage the following circuit breakers on the power shield: NORM LEFT WINDSHIELD, EMER LEFT WINDSHIELD, and EMER WINDSHIELD POWER.

e. Connect AC voltmeters between terminal C2 of the RH WINDSHIELD CONTROL relay and ground, and between terminal B2 of the LH WINDSHIELD CONTROL relay and ground.

f. Place the WINDSHIELD ANTI-ICING HEAT control switch in the "LOW" position.

g. Place the WINDSHIELD ANTI-ICING PWR control switch in the "NORM" position. Check the temperature of all the windshield panels by touching them with the hands.
RESULT: All the windshield panels should become warm, but not too hot to touch. The voltmeters should read between 210 volts and 216 volts.

h. Place the WINDSHIELD ANTI-ICING PWR control switch in the "OFF" position, and allow the windshields to cool.

30.2.2 (Cont).

i. When the windshield panels have cooled, place the WINDSHIELD ANTI-ICING HEAT control switch in the "HIGH" position.

j. Place the WINDSHIELD ANTI-ICING PWR control switch in the "NORM" position. The voltmeters should read between 320 volts and 340 volts. After a few seconds, place the switch in the "OFF" position. Check the temperature of all the windshield panels.
RESULT: The pilot's and copilot's forward windshields should heat rapidly, the clear vision and center windshields should heat normally (more slowly than the forward windshields), and the aft windshields should remain cold.

k. Place the WINDSHIELD ANTI-ICING HEAT control switch in the "LOW" position.

l. Remove the electrical connector from both the left and right windshield heat control boxes, and connect a voltmeter capable of measuring 28-volts DC to pins D (-) and E (+) on the connector removed from one of the control boxes.

m. Place the WINDSHIELD ANTI-ICING HEAT control switch in the "LOW" position.

n. Depress the LOW AMBIENT START pushbutton, and observe the voltmeter.
RESULT: The voltmeter connector to the control box wiring should indicate 28 (±1) volts.

o. Release the LOW AMBIENT START pushbutton. Disconnect the voltmeter from the electrical connector, and connect it to pins D and E of the connector on the wiring of the other control box. Repeat step (l) for the same result.

p. Place the WINDSHIELD ANTI-ICING PWR control switch in the "OFF" position. Disconnect the voltmeter from the control box connector. Connect the wiring to both control boxes.

q. Energize the main inverter.

r. Place the WINDSHIELD ANTI-ICING PWR control switch in the "EMER" position, wait a few seconds, and return the switch to the "OFF" position.
RESULT: The pilot's forward windshield should heat rapidly while the switch is in "EMER," The other windshields should not heat.

s. Turn off all power, and return the system to normal.

30.2.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ALL WINDSHIELD DO NOT HEAT DURING "NORM" OPERATION.		
Defective control switch.	Check continuity of control switch.	Replace control switch.
Defective control box.	Check power supply to control box connector pins D and E.	If control box is receiving power, replace control box.
Defective wiring between control box and windshield.	Perform continuity check of windshield power supply wiring between control box and windshield terminals.	Repair wiring as necessary.
Defective control relay.	Check operational of relay.	Replace relay.
Defective thermal switch in aft windshields.	Check operation of thermal switches.	Replace thermal switches as needed.
Defective thermistor.	Check resistance of thermistor (see figure 30-4).	Replace windshield panel.

30.2.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
WINDSHIELDS HEATING INCORRECTLY IN "EMER" OPERATION.		
Defective control switch or emergency circuits.	Check continuity of control switch and emergency circuits.	Replace switch or repair as necessary.
WINDSHIELDS DO NOT START AT LOW AMBIENT TEMPERATURE.		
Control switch position incorrect.	Check that control switch is in "LOW" position.	Place control switch in "LOW" position.
Defective LOW AMBIENT START pushbutton or pushbutton wiring.	Check pushbutton and wiring continuity.	Replace pushbutton or repair wiring as necessary.
Defective control box.	Check power supply to control box connector pins D and E, with pushbutton depressed.	If control box is receiving power, replace control box.

30.2.3.1 FAILED THERMISTOR. If left or right windshields fail to heat during flight and the problem will not repeat on the ground while un-pressurized, the thermistor in the pilot's or copilot's windshields is probably opening when the aircraft is pressurized. This can be confirmed by pressurizing the aircraft on the ground and checking the thermistor resistance with a Volt-Ohmmeter. If this test confirms that the thermistor is open when the aircraft is pressurized, the center windshield thermistor should be checked for serviceability before proceeding with the following paragraphs.

If the copilot's windshield thermistor is found to be opening when the aircraft is pressurized and the panel is otherwise serviceable, operation may be restored by moving thermistor wires from terminals 2 and 3 on the copilot's windshield to terminals 2 and 3 on the center windshield. It will also be necessary to move wires from C1 and C2 on LH windshield heat control relay to terminals B1 and B2 or A1 and A2 (unused terminals) on RH windshield heat control relay.

If the pilot's windshield thermistor is found to be opening and the panel is otherwise serviceable, operation may be restored by moving thermistor wires from terminals 2 and 3 on the pilot's windshield to terminals 2 and 3 on the center windshield.

If this wiring change is made, it should be entered in the appropriate aircraft forms. When the panel with the open thermistor is replaced, the wiring should be changed to the normal configuration. Reference wiring diagram JR716.

CAUTION

If the control circuits have been transferred from the pilot's panel to the center panel, which is not heated in the EMERGENCY mode, the temperature of the pilot's panel should be monitored closely (by touch) and cycled manually with the control switch.

30.2.4 REMOVAL AND REPLACEMENT.

30.2.4.1 REMOVAL AND REPLACEMENT OF WINDSHIELD PANELS. Refer to Chapter 56 for instructions for removal and replacement.

CAUTION

When replacing the clear vision panels, make sure the wiring harness is sufficiently clear of surrounding structure at all panel positions to avoid chafing during panel movement. Allow enough slack in the wiring between the terminals and the clamp to prevent strain on the wiring. The correct wiring terminals are AMP type 34112; these should always be used.

30.2.4.1 (Cont).

Each windshield panel is marked to show the resistance range of its heating element. Low resistance panels are marked RA, average resistance panels are marked RB and high resistance panels are marked RC. This resistance designation is marked on the glass near the outer edge. It is important that the resistance of all windshield panels, except the clear vision panels, be matched to the transformer output by connecting to the proper output terminal. For proper connections, refer to the Electrical and Electronic Wiring Diagram Book (REF JR716).

Occasionally the resistance markings on the JF30 (center windshield panel), JF31 (pilot's/copilot's windshield panels), JF41 (L/R clear vision panels) and JF489 (L/R aft side panels) windshields are difficult to read. The table below lists bus-to-bus resistance for each of the windshields mentioned above.

	JF 30	JF 31	JF 41	JF489
MARKING	CENTER PANEL	PILOT'S/ COPILOT'S PANEL	CLEAR VISION	AFT SIDE PANEL
R-A	66.1 to 74.5 ohms	97.7 to 110.1 ohms	72.1 to 81.1 ohms	144.2to 161.7ohms
R-B	73.1 to 82.5 ohms	108.1 to 121.9 ohms	79.7 to 89.9 ohms	158.5to 177.6ohms
R-C	79.5 to 89.7 ohms	117.3 to 132.3 ohms	86.0 to 97.5 ohms	174.1to 195.2ohms
Resistance checked between terminals	1 & 4	1 & 4	1 & 4	1&2

30.2.4.2 PILOT'S AND COPILOT'S WINDSHIELD PANELS (JF31-7 L/H). The JF31-7 L/H windshield panels are provided with a spare thermistor which saves changing a windshield if a thermistor fails. The panels have six terminals, two of which are for the spare thermistor. Figure 30-7 shows the location and identification of these terminals. When the JF31-7 L/R is installed it will be necessary to drill two 5/8-inch diameter holes through the JF400-6 L/R upper retainer in line with the two new terminals (see IPB figure 23, item 22).

30.2.4.3 AFT SIDE WINDSHIELD PANEL THERMOSTAT. The vendor acid etches the spot on the inside of the window glass where the thermostat should be placed. The etched spot or control point is located at the hottest spot (for most effective control) after manufacturing and during testing. The "hot" spot can lay along a line 2.38 inches from the upper edge (parallel with the upper frame) within the limits of the heated area. When the windshield is replaced it may be necessary to lengthen the thermostat wiring in order to provide for proper placement of the thermostat. If this is the case, splice in whatever length is required. Use wire per MIL-W-81044/9, Number 20.

30.2.5 CLEANING. Refer to Chaper 12 for instructions on the proper materials and methods for cleaning the windshields.

30.2.6 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. The electrically heated windshields require no special procedures under extreme temperature conditions. When the airplane is operated in dusty conditions, abrasion of the glass can be caused by attempting to remove dust with a dry cloth. Always clean the glass according to the instructions in Chapter 12.

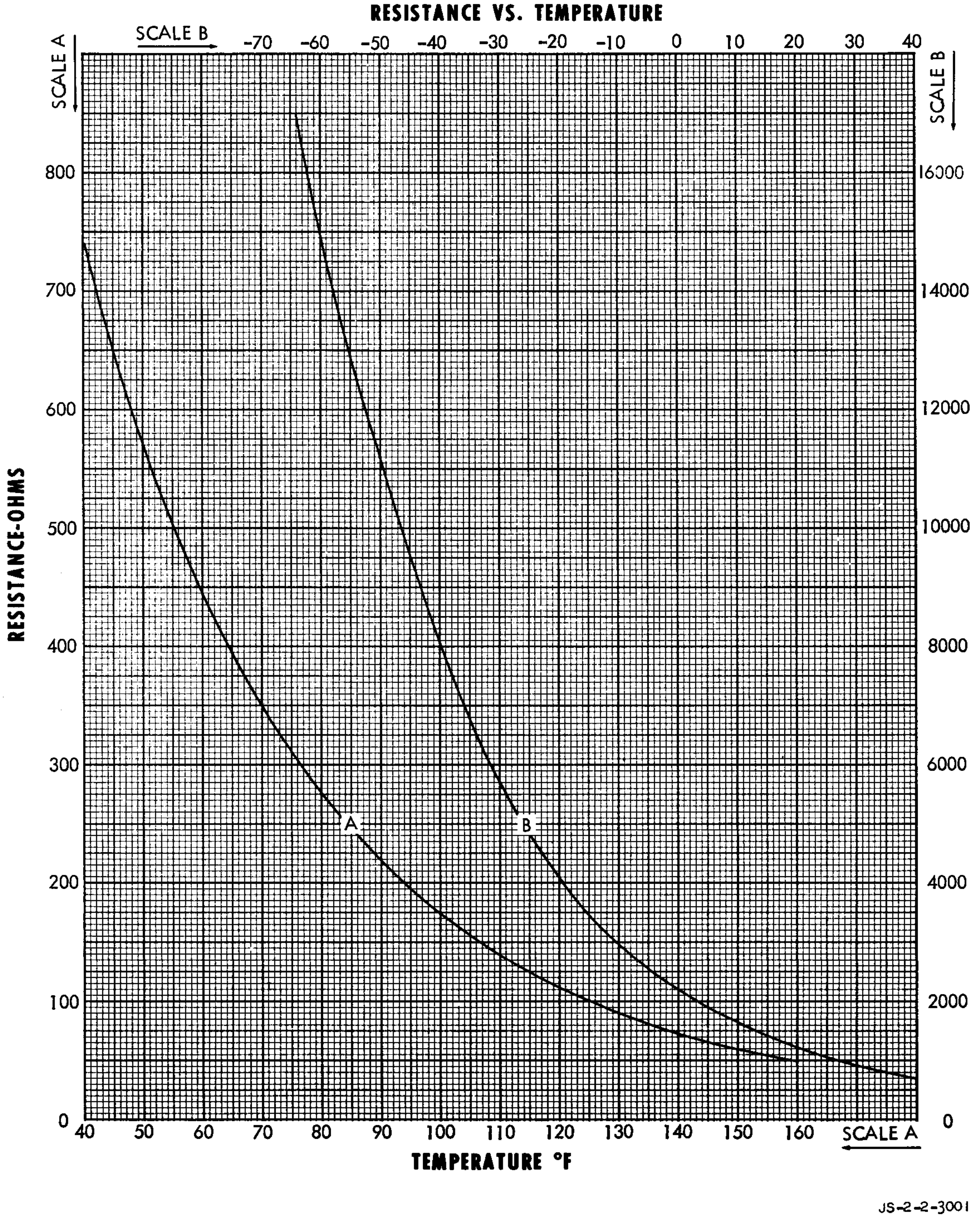
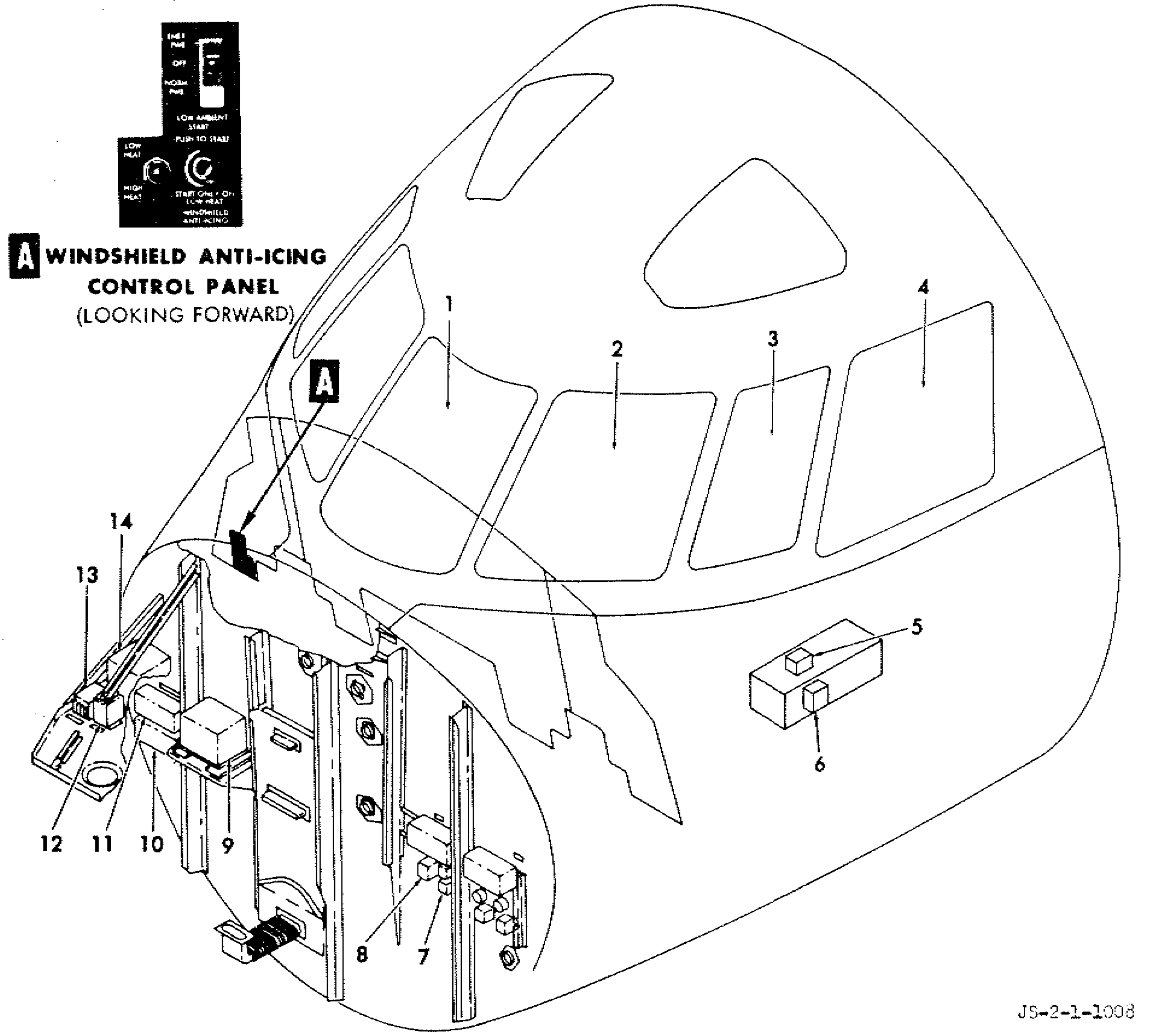
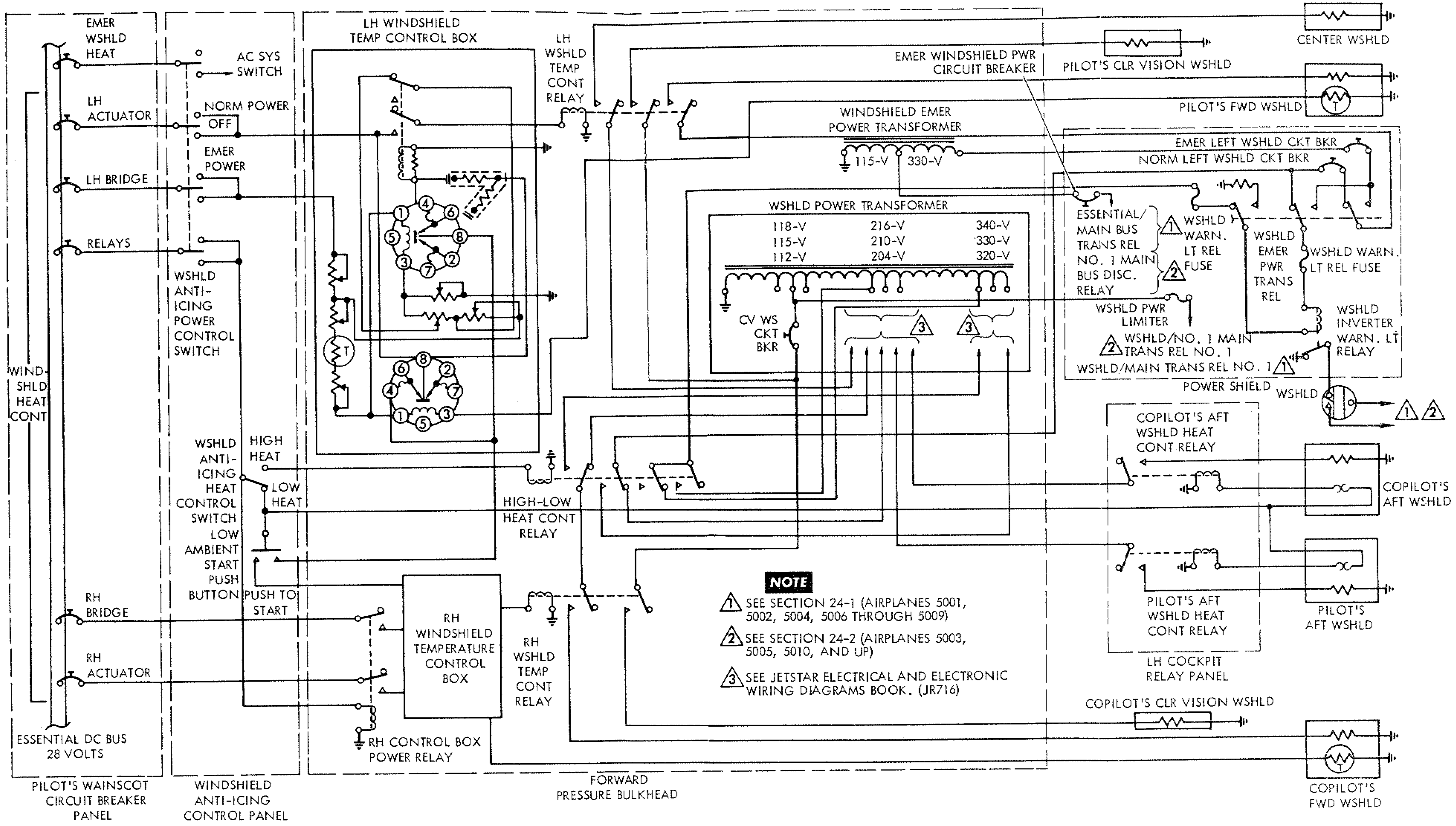


Figure 30-4. Thermistor Resistance Chart



1. CENTER WINDSHIELD
2. PILOT'S FORWARD WINDSHIELD
3. PILOT'S CLEAR VISION WINDSHIELD
4. PILOT'S AFT WINDSHIELD
5. COPILOT'S AFT WINDSHIELD HEAT CONTROL RELAY
6. PILOT'S AFT WINDSHIELD HEAT CONTROL RELAY
7. RIGHT-HAND CONTROL BOX POWER RELAY
8. HIGH-LOW HEAT CONTROL RELAY
9. RIGHT-HAND WINDSHIELD TEMPERATURE CONTROL BOX
10. WINDSHIELD EMERGENCY POWER TRANSFORMER
11. WINDSHIELD POWER TRANSFORMER
12. LEFT-HAND WINDSHIELD TEMPERATURE CONTROL RELAY
13. RIGHT-HAND WINDSHIELD TEMPERATURE CONTROL RELAY
14. LEFT-HAND WINDSHIELD TEMPERATURE CONTROL BOX

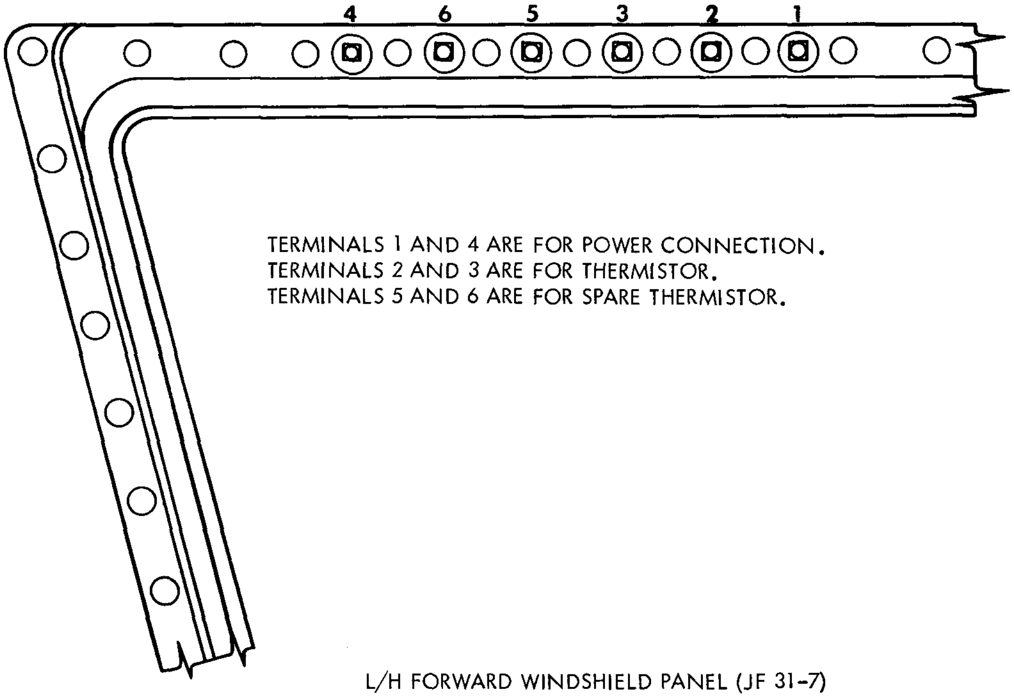
Figure 30-5. Windshield Anti-Icing System Components


EMER WSHLD HEAT
LH ACTUATOR
LH BRIDGE
RELAYS
WIND-SHLD HEAT CONT
RH BRIDGE
RH ACTUATOR
ESSENTIAL DC BUS 28 VOLTS
PILOT'S WAINSCOT CIRCUIT BREAKER PANEL
AC SYS SWITCH
NORM POWER
OFF
EMER POWER
WSHLD ANTI-ICING POWER CONTROL SWITCH
WSHLD ANTI-ICING HEAT CONTROL SWITCH
HIGH HEAT
LOW HEAT
LOW AMBIENT START PUSH BUTTON
PUSH TO START
WINDSHIELD ANTI-ICING CONTROL PANEL
LH WINDSHIELD TEMP CONTROL BOX
LH WSHLD TEMP CONT RELAY
HIGH-LOW HEAT CONT RELAY
RH WINDSHIELD TEMPERATURE CONTROL BOX
RH WSHLD TEMP CONT RELAY
RH CONTROL BOX POWER RELAY
FORWARD PRESSURE BULKHEAD
EMER WINDSHIELD PWR CIRCUIT BREAKER
WINDSHIELD EMER POWER TRANSFORMER
115-V
330-V
WSHLD POWER TRANSFORMER
118-V
115-V
112-V
216-V
210-V
204-V
340-V
330-V
320-V
CV WS CKT BKR
CENTER WSHLD
PILOT'S CLR VISION WSHLD
PILOT'S FWD WSHLD
EMER LEFT WSHLD CKT BKR
NORM LEFT WSHLD CKT BKR
ESSENTIAL/ MAIN BUS TRANS REL NO. 1 MAIN BUS DISC. RELAY
WSHLD WARN. LT REL FUSE
WSHLD EMER PWR TRANS REL
WSHLD WARN. LT REL FUSE
WSHLD PWR LIMITER
WSHLD/NO. 1 MAIN TRANS REL NO. 1
WSHLD/MAIN TRANS REL NO. 1
WSHLD INVERTER WARN. LT RELAY
POWER SHIELD
WSHLD
COPILOT'S AFT WSHLD HEAT CONT RELAY
COPILOT'S AFT WSHLD
PILOT'S AFT WSHLD HEAT CONT RELAY
LH COCKPIT RELAY PANEL
PILOT'S AFT WSHLD
COPILOT'S CLR VISION WSHLD
COPILOT'S FWD WSHLD
T

NOTE

1. SEE SECTION 24-1 (AIRPLANES 5001, 5002, 5004, 5006 THROUGH 5009)
2. SEE SECTION 24-2 (AIRPLANES 5003, 5005, 5010, AND UP)
3. SEE JETSTAR ELECTRICAL AND ELECTRONIC WIRING DIAGRAMS BOOK. (JR716)

Figure 30-6. Windshield Anti-Icing System Electrical Schematic Diagram



TERMINALS 1 AND 4 ARE FOR POWER CONNECTION.
TERMINALS 2 AND 3 ARE FOR THERMISTOR.
TERMINALS 5 AND 6 ARE FOR SPARE THERMISTOR.

L/H FORWARD WINDSHIELD PANEL (JF 31-7)

JS-1-2-X0/2-3006

Figure 30-6A. Pilot's and Copilot's Windshield Panels

SECTION 30.3
WINDSHIELD WIPER SYSTEM

Contents — Page

DESCRIPTION AND OPERATION 30-27
  General 30-27
  Windshield Wiper Control Circuit 30-27

OPERATIONAL CHECKOUT 30-28

TROUBLESHOOTING 30-29

REMOVAL AND REPLACEMENT 30-29
  Removal of Windshield Wiper Blade Assembly 30-29
  Replacement of Windshield Wiper Blade 30-29
  Removal of Windshield Wiper Arm Assembly 30-30
  Replacement of Windshield Wiper Arm Assembly 30-30
  Removal of Windshield Wiper Converter 30-30
  Replacement of Windshield Wiper Converter 30-30
  Removal of Windshield Wiper Flex Drive 30-30
  Replacement of Windshield Wiper Flex Drive 30-30
  Removal of Windshield Wiper Motor 30-30A
  Replacement of Windshield Wiper Motor 30-30A
  Removal of Windshield Wiper Control Switch 30-30A
  Replacement of Windshield Wiper Control Switch 30-30A
  Removal of Windshield Wiper Circuit Resistor 30-30A
  Replacement of Windshield Wiper Circuit Resistor 30-30A

ADJUSTMENT 30-30A

EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION 30-30B

FIGURES
  30-7. Windshield Wiper System Components 30-31
  30-8. Windshield Wiper System Electrical Schematic Diagram 30-33

30.3.1 DESCRIPTION AND OPERATION.

30.3.1.1 GENERAL.

a. Electrically driven windshield wipers are provided for the pilot's and copilot's forward windshield. A 28-volt, DC, variable-speed motor drives each wiper blade through a flexible drive shaft and a converter. The converter output shaft drives the wiper arm and blade assembly. The converter changes rotary motion of the flexible drive shaft to oscillating motion for blade operation. The pressure exerted on the windshield glass by a wiper blade can be adjusted by turning a blade pressure adjusting nut. The location of the area swept by a blade can be adjusted by rotating the blade arm on the converter output shaft, to which it is connected by vernier splines.

b. The two windshield wiper motors, one for the pilot's and one for the copilot's windshield wiper, obtain electrical power through an LH WINDSHIELD WIPER and an RH WINDSHIELD WIPER circuit breaker on the pilot's wainscot circuit breaker panel. Both wipers are controlled by a single, four-position ("LOW," "HIGH," "OFF," and "PARK") rotary switch on the exterior lights control panel. This switch varies the input voltage to the motors, through windshield wiper circuit resistors, to vary the motor speed. Overload and overheat protection is provided by an automatically resetting thermal protector in each motor assembly. If motor current draw reaches 8 to 10 amperes, or if the thermal protection device reaches approximately 300°F, the motor circuit is opened.

CAUTION

Operation of the windshield wiper system above flight speeds of 205 knots will cause lift off of the wiper blades from the windshield. Operation above flight speeds of 425 knots will damage the complete system.

30.3.1.2 WINDSHIELD WIPER CONTROL CIRCUIT. When the WINDSHIELD WIPERS control switch is placed in the "HIGH" position, 28-volt, DC electrical power passes directly through the drive motors to ground, bypassing the resistors. (See figure 30-8.) The wipers then operate at full speed, 275 strokes per minute. A wiper stroke is the complete blade movement from one side of the windshield to the other.

30.3.1.2 (Cont).

Placing the WINDSHIELD WIPERS control switch in the "LOW" position connects the windshield circuit resistors in series with the drive motors, providing a lower blade speed. In the "OFF" position, the control switch disconnects all electrical power from the drive motors, and the blades stop immediately, regardless of their positions on the windshields. The "PARK" position of the switch is a momentary position used to move the blades to the inboard edges of the windshield and to stop them there. With the control switch held in the "PARK" position, partial voltage is applied to the drive motors until the blades reach the inboard edges of the windshields. The pilot's blade is thus parked on the right side of the left windshield, and the copilot's blade on the left side of the right windshield, presenting the minimum obstruction to vision through the windshield. When the blades reach the parked positions, a switch in each motor closes to complete a short circuit across the motor armature. The blades then remain in the parked position. When the control switch is released from the "PARK" position, the switch in each motor automatically opens to remove power from the motor. Each motor obtains electrical power from a separate bus, to reduce hazards due to an electrical power failure.

30.3.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Equipment	Description	Use and Application
Auxiliary Power Unit	28V DC electrical power.	To furnish power for the windshield wiper system.
Spring Scale.	0 to 20 lbs. capacity or equivalent.	To adjust windshield wiper blade pressure on windshield.
Water supply, spray nozzle, and hose.		To wet windshields while wipers are operating.

CAUTION

The windshield must be wet during wiper operation or the blades or glass can be damaged.

a. Check the pressure of the wiper blades against the windshields, according to the instructions in paragraph 30.3.5 and adjust blade pressure if necessary.

b. Connect the auxiliary power unit to the airplane external 28-volt, DC electrical power receptacle.

c. Close the LH and RH WINDSHIELD WIPER circuit breakers on the pilot's wainscot circuit breaker panel.

d. Make sure the clear vision windshields are closed and secured.

e. Spray water on the outside of the forward windshields. Continue to spray during all wiper operation.

f. Place the WINDSHIELD WIPERS control switch in the "LOW" position, and observe wiper operation.
RESULT: Both windshield wipers should operate at a low speed.

g. Place the WINDSHIELD WIPERS control switch in the "HIGH" position, and observe wiper operation.
RESULT: Both windshield wipers should operate at not less than 275 strokes per minute. In each stroke, each blade should effectively sweep its corresponding windshield area.

h. Momentarily hold the WINDSHIELD WIPERS control switch in the "PARK" position.
RESULT: Both blades should stop and be parked at the inboard edge of the respective windshield. The distance between the blade tip centerline and the edge should be, as nearly as possible, 0.10 inch.

i. Place the WINDSHIELD WIPERS control switch in the "OFF" position. Remove the water hose and spray nozzle. Disconnect external electrical power, and return the airplane to normal.

30.3.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
WIPERS DO NOT OPERATE WHEN TURNED ON.		
Power supply to motors interrupted, or both motors defective	Check for 28 volts, DC at motor terminals. If not available, turn off all power and perform a continuity check of the circuit.	If 28 volts, DC available at motor terminals, replace both motors. If power is not available, replace defective parts or repair circuit.
WIPER SPEED LOW.		
Flexible drive shaft or converter defective, or motor receiving insufficient power.	Check for 28 volts, DC at motor terminals. Inspect drive train for shaft binding or kinking.	Repair circuit, or replace defective drive components.
Blade pressure too high.	Check blade pressure on glass according to instructions in paragraph 30.3.5.	Adjust blade pressure.
BLADES DO NOT WIPE GLASS EFFECTIVELY.		
Worn or defective blades, or insufficient blade pressure.	Check blade condition. Check blade pressure according to instructions in paragraph 30.3.5.	Replace defective blade, or adjust blade pressure.
BLADE DOES NOT PARK CORRECTLY.		
Parking switch in motor out of adjustment, or system rigged incorrectly.	Check blade rigging according to instructions in paragraph 30.3.5.	Adjust blade rigging or replace motor, as necessary.

30.3.4 REMOVAL AND REPLACEMENT.
(See figure 30-7, sheets 1 and 2.)

Note

Procedure for removal and replacement of both pilot's and copilot's windshield wiper system components are the same except as specifically noted in the following steps.

CAUTION

Pull the LH and RH WINDSHIELD WIPER circuit breakers on the pilot's wainscot circuit breaker panel before removing components of the windshield wiper system.

30.3.4.1 REMOVAL OF WINDSHIELD WIPER BLADE ASSEMBLY. Before removing the windshield wiper blade assembly, secure a protective cover over the windshield. Remove cotter pin from the castellated nut on wiper blade assembly shaft. Remove castellated nut and washer. Remove locknut from serrated disc stud. Lift the wiper arm up and remove the wiper blade assembly. Gently return the wiper arm to rest on the protective cover, taking care not to let the wiper arm damage the windshield glass.

30.3.4.1.A Removal of the Windshield Wiper Blade. With a protective cover secured in position over the windshield, remove locknut from bolt which retains wiper blade in a U-channel. Lift wiper arm from protected windshield and remove retaining bolt, taking care not to let wiper blade fall. Remove the wiper blade. Gently return the wiper arm to rest on the protective cover so as not to damage the windshield glass.

30.3.4.2 REPLACEMENT OF THE WINDSHIELD WIPER BLADE ASSEMBLY. Position the wiper blade assembly as follows:

a. Attachment stud is directly over the wiper arm attachment hole.

30.3.4.2 (Cont).

b. Long end of wiper blade is upward.

c. Edge of assembly is approximately 0.10 inch from edge of windshield glass.

1. Position serrated disc stud in line with wiper arm.
2. Adjust wiper blade to 0.10 inch from edge of windshield glass.

Lift the wiper arm, position wiper blade assembly, and lower wiper arm onto wiper blade assembly. Install washer and castellated nut on wiper blade assembly attachment stud. Install locknut on serrated disc stud. Install cotter pin in castellated nut. Check the pressure and sweep area of the wiper blade assembly per procedure in 30.3.2.

a. Replacement of the Windshield Wiper Blade. Lift the wiper arm and position wiper blade in U channel with the long end of the blade upward. Install attachment bolt and locknut. Check the pressure and sweep area of the wiper blade assembly per procedure in 30.3.2.

30.3.4.3 REMOVAL OF WINDSHIELD WIPER ARM ASSEMBLY. Back off the wiper blade tension adjusting nut to remove blade pressure against the glass. Remove lockwire from shoulder bolt. Remove shoulder bolt from the converter shaft. Remove wiper arm from converter shaft, taking care not to lose the serrated adjustment sleeve.

30.3.4.4 REPLACEMENT OF THE WINDSHIELD WIPER ARM ASSEMBLY. Before installing the wiper arm, secure a protective cover over the windshield. Adjust and install wiper arm per procedure in 30.3.5.

30.3.4.5 REMOVAL OF WINDSHIELD WIPER CONVERTER. Remove the windshield wiper arm per 30.3.4.3. Remove the overhead lighting control panel in the flight station. Remove fiberglass insulation sufficiently to permit access to converter. Disconnect flex drive and cap. Remove converter attaching screws, washers, and rubber grommets. Gently withdraw converter, taking care not to damage O-ring, rubber pad, and external fairing where converter shaft extends through cab top.

30.3.4.6 REPLACEMENT OF WINDSHIELD WIPER CONVERTER. Locate and position converter on support structure.

Note

A 1/8-inch thick rubber pad (MIL-R-6130A, Grade A, Type II) is cemented to support bracket. If damaged, replace before positioning converter. Cover exposed surfaces of converter with two (2) layers of one-inch-wide lead tape (Minnesota Mining & Mfg. Co. No. 420 or equivalent). Punch clearance holes through lead tape for converter mounting holes and flex drive connection.

Note

Adjacent structure is covered with lead tape. If lead tape is damaged, repair or replace.

Position converter and gently push up so as not to damage O-ring and gasket in external fairing. Align mounting holes in converter on rubber padded support bracket. Assemble on a NAS1633-19 screw, two (2) AN960D10 washers, and one (1) MS35489-4 grommet so that grommet rests on converter when screw is installed. Install four (4) NAS1633-19 screw assemblies, described above, and torque until crushing of grommet is observed. Connect flex drive. Install wiper arm assembly and wiper blade assembly per 30.3.4.4 and 30.3.4.2. Perform operational checkout per 30.3.2. Secure flex drive coupling nut with lockwire to opposite converter end plug cap and wrap flexible drive coupling with two layers of one-inch-wide lead tape. Overlap the tape by 1/64 to 3/16 inch at each wrap. Reinstall overhead lighting control panel. Perform necessary operational checkouts on other equipment or systems that may have been disturbed.

30.3.4.7 REMOVAL OF WINDSHIELD WIPER FLEX. Remove the overhead trim panels and overhead lighting control panel in the flight station. Remove necessary fiberglass insulation adjacent to overhead sill structure. Disconnect flex drive from converter, wiper motor, and cap. Remove clamps (three places) and flex drive. Remove alignment support tubes from flex drive and remove all lead tape.

30.3.4.8 REPLACEMENT OF WINDSHIELD WIPER FLEX DRIVE. Check fiberglass insulation and lead tape in area adjacent to overhead sill. Repair or replace if damaged. Position flex drive from wiper motor to converter; note approximate location of clamp supports and mark on flex drive. Locate and install alignment support tubes on flex drive. Connect flex drive to motor. Wrap alignment support tubes with two layers of one-inch-wide lead tape (Minnesota Mining and Mfg. Co. No. 420 or equivalent). Overlap the tape by 1/64 to 3/16 inch at each wrap. Install clamps around alignment support tubes. Attach clamps to clips (2 places) and overhead support angle. Replace MS35489-4 rubber grommets in attachment holes of clips if damaged. Prior to tightening attachment screws, allow clamps to seek neutral aligning position. Tighten attachment screw until crushing of grommet is observed. Connect flex drive coupling to converter. Check for proper system operation per 30.3.2. Secure flex drive coupling nuts to converter and motor with lockwire and wrap couplings with two layers of one-inch-wide lead tape as described above. Reinstall fiberglass insulation, overhead trim panels, and overhead lighting control panel. Perform necessary operational checkouts on other equipment or systems that may have been disturbed.

30.3.4.9 REMOVAL OF WINDSHIELD WIPER MOTOR. Remove aft overhead trim panel and necessary fiberglass insulation. Disconnect electrical connector, bonding jumper and flex drive coupling. Disengage shoulder bolts from motor base and remove motor. Cap flex drive coupling.

30.3.4.10 REPLACEMENT OF WINDSHIELD WIPER MOTOR. Position the motor with electrical connector facing aft on pilot's side. Position the motor with electrical connector facing forward on the copilot's side.

Note

> A special rubber grommet is installed in the support bracket mounting holes under an AN960-616 washer which is under the head of each shoulder bolt. Cemented to the support bracket is a 1/2-inch thick silicone rubber pad which absorbs motor vibrations and noise. Replace the grommet or pad if either is damaged.

Install motor, tighten shoulder bolts, and lockwire shoulder bolts to each other. Connect electrical connector, bonding jumper, and flex drive coupling. Perform operational checkout per 30.3.2. Secure the flex drive coupling nut by lockwiring to the plug cap on the opposite end of the motor. Wrap flex drive coupling with two layers of one-inch-wide lead tape (Minnesota Mining and Mfg. Co. No. 420 or equivalent). Overlap the tape by 1/62 to 3/16 inch at each wrap. Reinstall the aft overhead trim panel. Perform necessary operational checkouts on other equipment or systems that may have been disturbed.

30.3.4.11 REMOVAL OF WINDSHIELD WIPER CONTROL SWITCH. Remove the exterior lighting control panel from the overhead console in the flight station. Remove edge lighted plastic panel. Remove switch from metal backup panel.

30.3.4.12 REPLACEMENT OF WINDSHIELD WIPER CONTROL SWITCH. Orientate switch on metal backup panel and install. Connect panel wiring to applicable switch terminals per Wiring Diagram Handbook. Install edge lighted plastic panel, lens caps, and plastic knob on switch, taking care to maintain a .015-inch maximum gap between top surface of edge lighted plastic panel and bottom surface of plastic knob. Perform continuity check on disturbed wiring. Reinstall exterior lighting control panel. Perform operational checkout per 30.3.2. Perform necessary operational checkouts on other equipment or systems that may have been disturbed.

30.3.4.13 REMOVAL OF WINDSHIELD WIPER CIRCUIT RESISTOR. Remove the copilot's seat and copilot's aftmost floor panel. Locate resistor on forward side of 232 bulkhead by identifying decal. Remove resistor.

30.3.4.14 REPLACEMENT OF WINDSHIELD WIPER CIRCUIT RESISTOR. After replacing resistor, perform operational checkout per 30.3.2. Reinstall copilot's aftmost floor panel and copilot's seat.

30.3.5 ADJUSTMENT. Use the following procedure to adjust the location of the swept area, the parking position, and the blade pressure of the windshield wipers.

a. Remove the windshield wiper arm and blade assembly from each converter shaft, and operate the system with the WINDSHIELD WIPERS control switch in the "LOW" position, while observing the movement of the pilot's converter shaft to determine the extremes of rotation.

b. Place the WINDSHIELD WIPERS control switch in the "OFF" position to stop the pilot's converter at either end of its travel. It may be necessary to repeat this operation to obtain a satisfactory result, due to coasting of the motor after the switch has been placed "OFF."

c. Install the internally and externally serrated sleeve in any position in the wiper arm hub, and install the sleeve, arm, and blade assembly on the converter shaft so that the centerline of the bottom end of the blade is as close as possible to 0.10 inch from the edge of the glass, on the side towards which the converter shaft was stopped.

Note

> The internal and external serrations on the sleeve, in conjunction with the serrations on the converter shaft and in the arm hub, form a vernier adjusting device which allows the lower tip of the blade to be positioned in increments of 0.034 inch along its path when the sleeve is re-indexed in the arm hub by one serration.

d. Measure the distance between the centerline of the lower tip of the blade and a point 0.10 inch inward from the edge of the glass. Divide this distance by 0.034, to the nearest whole number. Remove the arm and sleeve from the converter shaft, with the sleeve still in position in the arm hub.

e. Remove the sleeve from the arm hub, and re-index it in the hub the same number of serrations around from its original location as the whole number obtained in the calculation made in step (d), and in the direction required to position the blade tip centerline 0.10 inch from the edge of the glass. Install the arm and sleeve on the converter shaft in this position.

f. Attach the wiper arm to the converter shaft with the shoulder bolt. Secure the bolt with lockwire. Back off the blade tension adjusting nut to remove blade pressure against the glass.

30.3.5 (Cont).

g. Place the WINDSHIELD WIPERS control switch in the "LOW" position, and observe the movement of the copilot's converter shaft to determine the extremes of rotation.

h. Repeat steps (b) through (f) to position and install the copilot's windshield wiper arm and blade.

i. Adjust the pressure of both blades against the glass by tightening the blade tension adjusting nuts until a spring scale, attached to the wiper arm at the blade assembly attachments and castellated nut location, just lifts the rubber from the glass when the scale indicates 10 (±1) pounds.

j. Operate the wipers on wet glass for at least 30 seconds, observing that the blades sweep the desired areas.

CAUTION

To avoid damage to blades and glass, spray the glass with water while the blades are operating.

k. Recheck the blade pressure, and adjust as necessary. If adjustment was required, repeat step (j), and again recheck the pressure. Adjust, operate the wipers, and recheck the pressure until the reading stabilizes at 10 (±1) pounds.

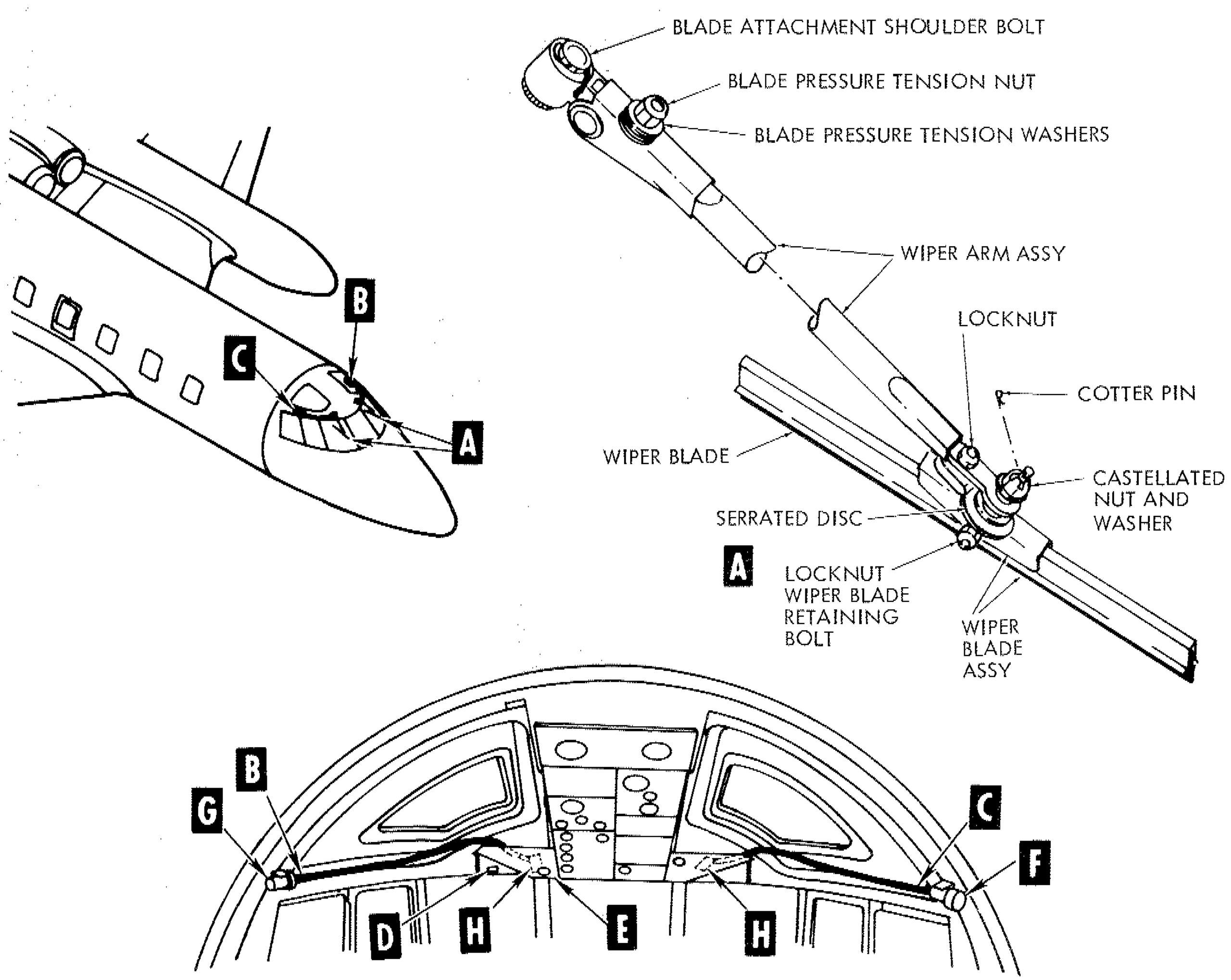
l. Spray the windshields with water, applied at various pressures, and operate the wipers to check their ability to clear the glass. Do not allow the wipers to operate on dry glass at any time during this check.

m. Place the WINDSHIELD WIPERS control switch in the "PARK" position, passing through the "OFF" position, and check that the centerline of each blade tip is positioned as described in step (e). If either blade stops in other than the correct position, disconnect the flexible drive shaft from the applicable converter, relieve blade friction by lifting the blade from the glass, and rotate the converter manually until the blade is positioned correctly.

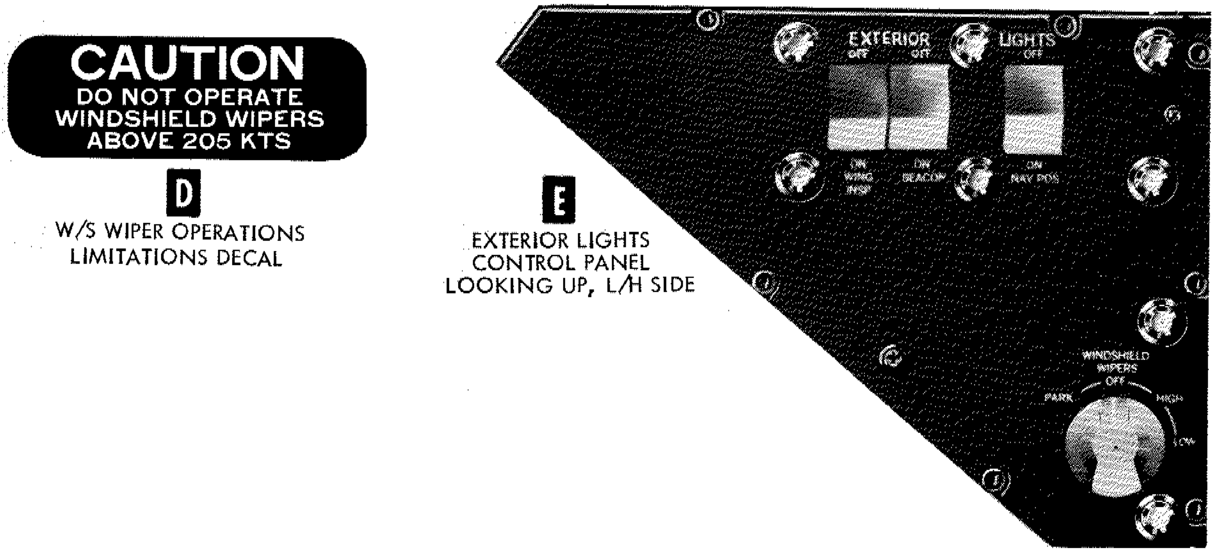
n. Connect the flexible drive shaft to the converter, operate the wipers with the glass wet, and recheck the parked position.

o. Secure the flexible drive shaft nut by lockwiring between the nut and the lockwire hole in the motor base. Wrap any flexible drive couplings which may have been disturbed with two layers of 1-inch-wide lead tape (Minnesota Mining and Manufacturing Co. No. 420 or equivalent). Overlap the tape by 1/64 to 3/16 inch at each wrap.

30.3.6 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. The windshield wiper system is designed to operate under any conditions that the airplane is expected to encounter. Blade life will be reduced by continual exposure to high ambient temperatures and by attempting to clear ice from the windshields by operating the wipers. Normal routine checks should reveal any necessity for maintenance of the system.


BLADE ATTACHMENT SHOULDER BOLT
BLADE PRESSURE TENSION NUT
BLADE PRESSURE TENSION WASHERS
WIPER ARM ASSY
LOCKNUT
COTTER PIN
WIPER BLADE
CASTELLATED NUT AND WASHER
SERRATED DISC
A
LOCKNUT WIPER BLADE RETAINING BOLT
WIPER BLADE ASSY
B
C
A
B
G
C
F
D
H
E
H

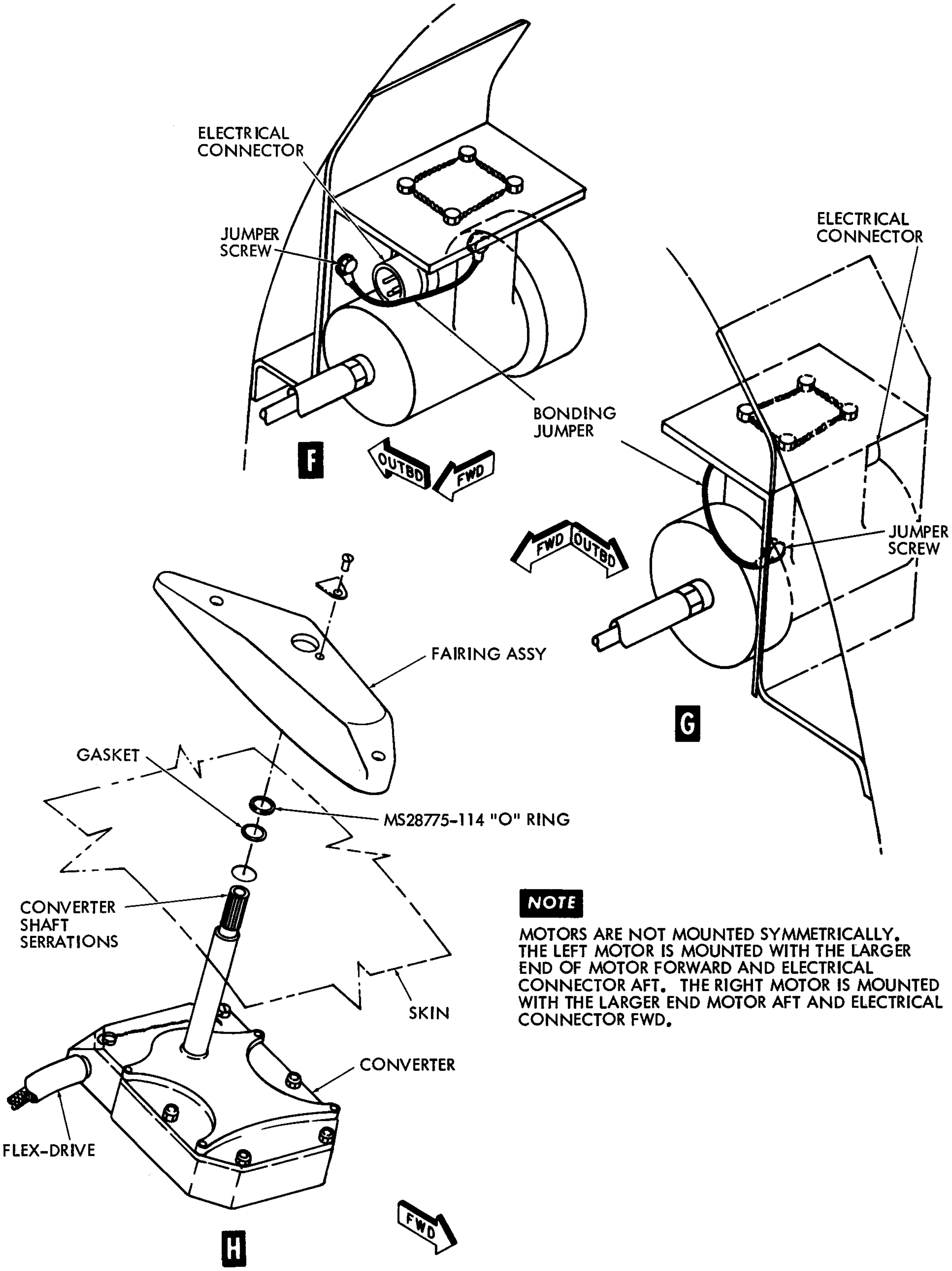
WINDSHIELD WIPER SYSTEM - FLIGHT STATION



W/S WIPER OPERATIONS LIMITATIONS DECAL

EXTERIOR LIGHTS CONTROL PANEL LOOKING UP, L/H SIDE

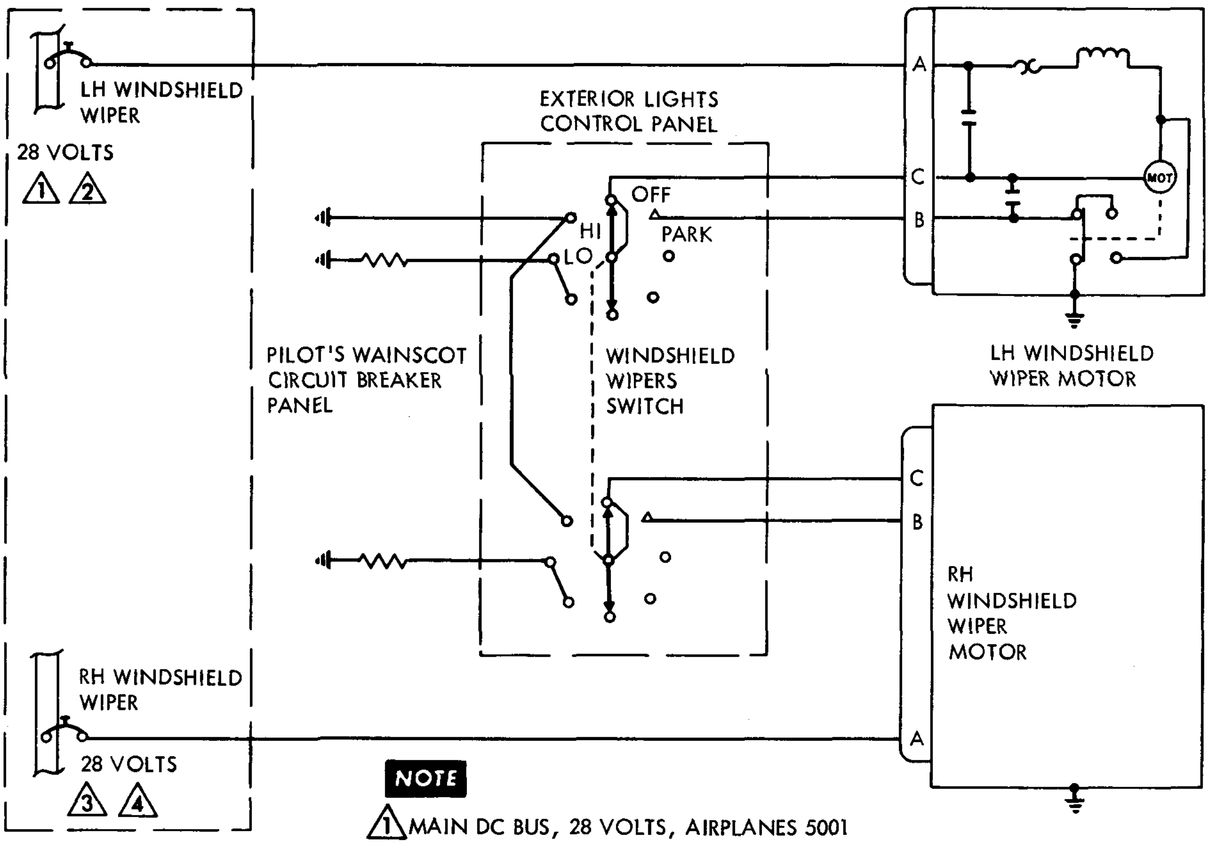
Figure 30-7. Windshield Wiper System Components (Sheet 1 of 2)



NOTE

MOTORS ARE NOT MOUNTED SYMMETRICALLY. THE LEFT MOTOR IS MOUNTED WITH THE LARGER END OF MOTOR FORWARD AND ELECTRICAL CONNECTOR AFT. THE RIGHT MOTOR IS MOUNTED WITH THE LARGER END MOTOR AFT AND ELECTRICAL CONNECTOR FWD.

Figure 30-7. Windshield Wiper System Components (Sheet 2 of 2)


LH WINDSHIELD WIPER
28 VOLTS
1 2
EXTERIOR LIGHTS CONTROL PANEL
OFF
HI
PARK
LO
A
C
B
MOT
PILOT'S WAINSCOT CIRCUIT BREAKER PANEL
WINDSHIELD WIPERS SWITCH
LH WINDSHIELD WIPER MOTOR
C
B
RH WINDSHIELD WIPER MOTOR
RH WINDSHIELD WIPER
A
28 VOLTS
3 4

NOTE

△1 MAIN DC BUS, 28 VOLTS, AIRPLANES 5001 THRU 5038 ONLY.

△2 ESSENTIAL DC BUS, 28 VOLTS, AIRPLANES 5039 AND UP.

△3 ESSENTIAL DC BUS, 28 VOLTS, AIRPLANES 5001 THRU 5038 ONLY.

△4 MAIN DC BUS, 28 VOLTS, AIRPLANES 5039 AND UP.

Figure 30-8. Windshield Wiper System Electrical Schematic Diagram

SECTION 30.4
PITOT TUBE AND STATIC PORT ANTI-ICING SYSTEM

Contents Page

DESCRIPTION AND OPERATION 30-35
General 30-35

OPERATIONAL CHECKOUT 30-35

FIGURES
30-9. Pitot Head Locations 30-37

30.4.1 DESCRIPTION AND OPERATION. (See figure 30-9.)

30.4.1.1 GENERAL. Provisions for heating the pitot tubes and the static ports include a two-position ("ON," "OFF") PITOT HEAT switch on the copilot's inclined panel and eight heating units. Each of the two pitot heads on the underside of the nose section contain a heating unit. Each of the nacelle pitot tubes contain a heating unit. A blanket type heating unit is installed at each of the four static port locations. The static port heating units contain a thermostat which removes power from the unit when the temperature increases to 110 (±8)° Fahrenheit and reapplies power when the temperature falls to 90 (±5)° Fahrenheit. All eight heating units are turned on simultaneously when the PITOT HEAT switch is placed in the "ON" position. Main DC power is supplied to the system through four PITOT HEATERS circuit breakers on the copilot's wainscot console circuit breaker panel. When the PITOT HEAT switch is placed in the "ON" position, 28-volt, DC electrical power passes directly from the four PITOT HEATERS circuit breakers to the eight heating units.

30.4.2 OPERATIONAL CHECKOUT.

TEST EQUIPMENT REQUIRED

EQUIPMENT	DESCRIPTION	USE AND APPLICATION
Auxiliary Power Unit	28 V DC electrical power	To energize heating units

CAUTION

Remove pitot head and pitot tube covers before turning heaters on.

a. Connect the auxiliary power unit to the airplane external 28-volt, DC electrical power receptacle.

b. Energize the MAIN DC BUS.

c. Engage the four PITOT HEATERS circuit breakers.

d. Position the PITOT HEAT switch to "ON."
RESULT: The two nacelle pitot tubes, two nose section pitot heads, and four static ports shall become warm to the touch.

e. Position the "PITOT HEAT" switch to "OFF."
RESULT: All pitot tubes, pitot heads, and static ports shall lose heat and return to ambient temperature.

f. Disconnect external electrical power and return the airplane to normal.

30.4.3 TROUBLESHOOTING.

a. Heaters do not operate when turned on.

   1. Check for 28 volts at heater terminals, switch terminals, and/or circuit breaker terminals. Replace defective parts or repair circuit.

30.4.4 REMOVAL AND REPLACEMENT.

30.4.4.1 Removal and Replacement of Pitot Head Heating Unit. Refer to Removal and Replacement of Pitot Head, 34.10.

30.4.4.2 Removal and Replacement of Static Port Heater Units. Refer to Removal and Replacement of Static Port, 34.10.

30.4.4.3. For removal and replacement of engine pressure ratio indicating probes, refer to Section 77.0.

1. COPILOT'S STATIC PORT HEATER (R.H.)
2. AFT STATIC PORT HEATER (R.H.)
3. AFT STATIC PORT HEATER (L.H.)
4. PILOT'S STATIC PORT HEATER (L.H.)
5. PILOT'S PITOT HEAD HEATER (L.H.)
6. COPILOT'S PITOT HEAD HEATER (R.H.)
7. NACELLE PITOTHEAD HEATER

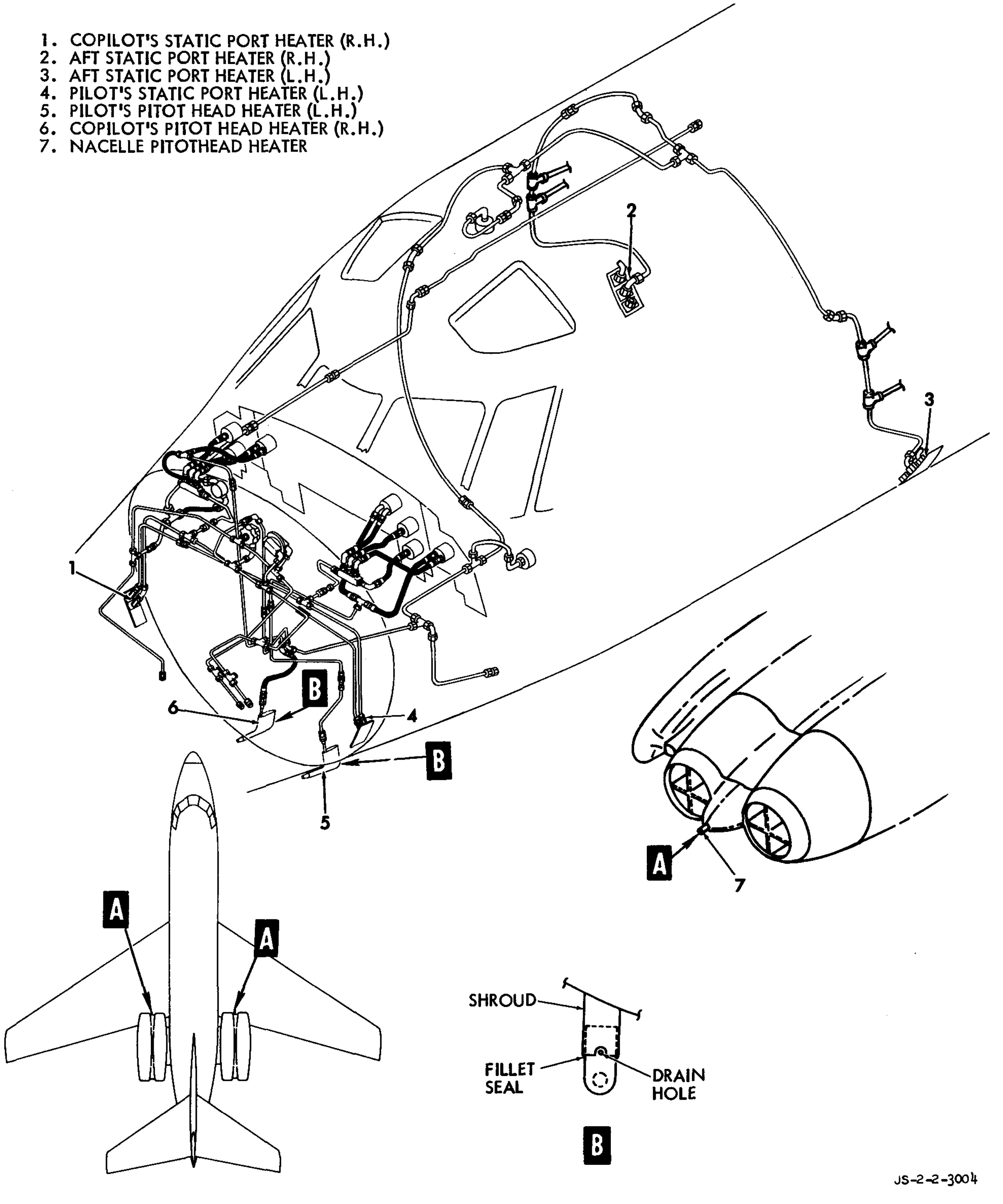


Figure 30-9. Pitot Head and Static Port Anti-Icing Locations

SECTION 30.5
WINDSHIELD DE-FOG SYSTEM

30.5.1 Air from the flight station air conditioning and pressurization system supply is used for windshield defogging. A discharge nozzle for each flat forward windshield directs a stream of air over the inner surface of the windshield glass. The system functions when air is flowing to the flight station through the supply ducts, either during normal system operation or while ram air ventilation is being supplied, provided the manual shutoff slides (located on the glare shield), one for the pilot and one for the copilot, are open. There is no air available for the windshield de-fog system while the emergency pressurization is being utilized.

SECTION 30.6

CABIN WINDOW DE-FOG SYSTEM

Contents Page

GENERAL 30-41

REPLACEMENT 30-41
  Replacement or Reactivation of Desiccant 30-41
  Replacement of Components 30-41
  Replacement of Window Plug 30-41

FIGURES
  30-9A. Cabin Window De-Fog System Components 30-42A

30.6.1 GENERAL. (See figure 30-9A.)

Five windows flank the cabin on each side. Beneath each of these cabin windows, behind the interior trim, is a small vial filled with 2 bags (10 grams each) of activated desiccant and connected to the window by a small tube. The desiccant absorbs the moisture from the air in the open space between the window laminations and prevents window misting. When the windows begin to mist, the desiccant should be replaced or reactivated. Also, connected to these windows is a flexible plastic air-tight bag which, by expanding and contracting, keeps the air pressure in the window air space at approximately the same value as the cabin air pressure. These bags are connected to the windows by tubes leading from the desiccant vial, and are located beneath the vial. Since the window, desiccant, and plastic bag form an air-tight system, the enclosed air is continuously reused; this increases the life of the desiccant.

30.6.2 REPLACEMENT.

30.6.2.1 REPLACEMENT OR REACTIVATION OF DESICCANT. When replacing desiccant, use MIL-D-3464 Type I and Type II, Filtrol Corp., Los Angeles, Calif. The bagged desiccant may be reactivated by drying in an oven as follows:

Reactivation temperature in bag 245-275°F
Reactivation time 12 hours (min)
Cool in airtight container

30.6.2.2 REPLACEMENT OF COMPONENTS. Should it be necessary to replace any of the components mentioned in paragraph 30.6.1, one of the following approved sealants (MIL-S-7502, Class B-1/2) may be used to ensure airtight connections:

a. PRO SEAL 707B-1/2
   Essex Chem. Corp.
   Color: Choc Brown

b. EC1239 Class B-1/2
   Minn. Mining & Mfg. Co.
   Color: Choc Brown

c. PR-1221-1/2
   Products Research Co.
   Color: Choc Brown

d. PRESSTRITE 1170 Class B-1/2
   Presstrite-Keystone Engr. Co.
   Color: Choc Brown

30.6.2.3 REPLACEMENT OF WINDOW PLUG. When replacing window plug, it is essential that airtight connections are made between window plug and window, and window plug and the polyvinyl chloride tubing (see figure 30-9A, detail C). Use one of the following reclaimed rubber adhesives:

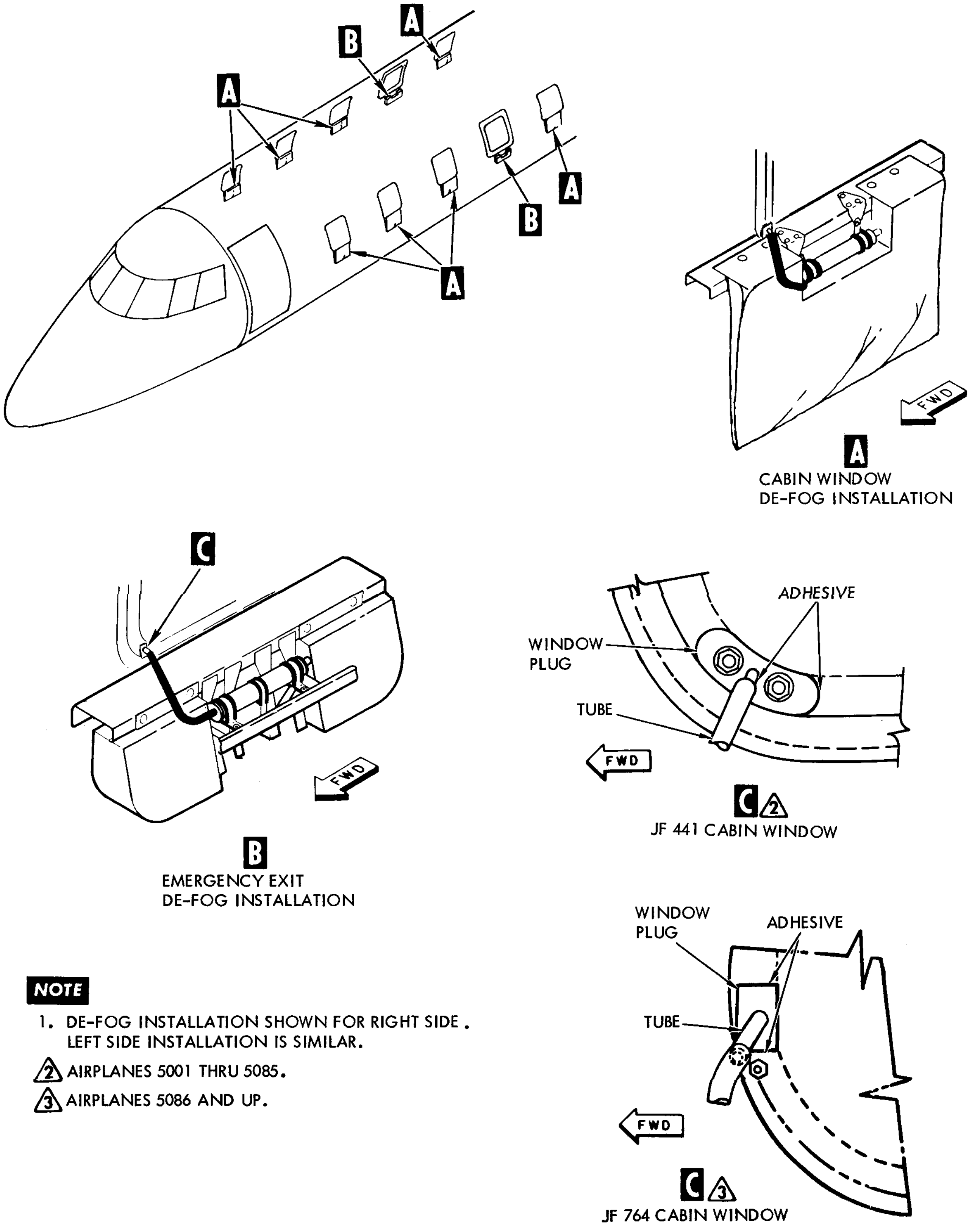
a. R-1134-T
   B.F. Goodrich Co.
   Adhesive Products

b. BOSTIK 3045
   USM Corporation
   Chemical Division

c. U.S. Royal M6314
   U.S. Rubber Company
   Adhesive and Coating Dept.

d. R-1053
   Shore Chemical Co., Inc.

e. LOXITE 103-486
   Xylos Rubber Co.
   Division of Firestone
   Tire & Rubber Co.



CABIN WINDOW DE-FOG INSTALLATION

EMERGENCY EXIT DE-FOG INSTALLATION

JF 441 CABIN WINDOW

JF 764 CABIN WINDOW

NOTE

1. DE-FOG INSTALLATION SHOWN FOR RIGHT SIDE. LEFT SIDE INSTALLATION IS SIMILAR.

△2 AIRPLANES 5001 THRU 5085.

△3 AIRPLANES 5086 AND UP.

JS-1-2-X0/2-3009A

Figure 30-9A. Cabin Window De-fog System Components

SECTION 30.7
ENGINE AND NACELLE ICE PROTECTION

Contents — Page

DESCRIPTION AND OPERATION 30-43

OPERATIONAL CHECKOUT 30-44

REMOVAL OF NACELLE ANTI-ICING SHUT-OFF VALVE 30-44

INSTALLATION OF NACELLE ANTI-ICING SHUT-OFF VALVE 30-44

FIGURES

30-10. Nacelle Anti-Icing Air Flow 30-45

30-11. Nacelle and Engine Inlet Duct Anti-Icing Schematic Diagram (Airplanes 5001, 5002, and 5004 through 5035) 30-46

30-12. Nacelle and Engine Inlet Duct Anti-Icing Schematic Diagram (Airplanes 5003, 5036 and Up) 30-47

30.7.1. DESCRIPTION AND OPERATION.
(See figures 30-10, 30-11, and 30-12.)

a. Each engine is protected from ice formation by a nacelle and engine inlet duct anti-icing system. This system is composed of two sub-systems that are turned on and off by two ENG ANTI-ICE switches at the pilot's instrument panel. One switch controls the circuits for engines No. 1 and 4 and the other controls the circuits for engines No. 2 and 3. Each switch has two positions: "OFF," and "ON." When either switch is placed "ON," it energizes the nacelle inlet duct anti-icing valve and de-energizes the engine inlet duct anti-icing valve for the two affected engines. Both valves open and remain open until the ENGINE ANTI-ICE switch is placed in the "OFF" position.

b. The nacelle section of the anti-icing systems obtains hot bleed air from the engine diffuser case. The hot air flows through a duct to a solenoid actuated pneumatic shutoff valve. Placing the ENG ANTI-ICE switch in the "ON" position opens the shutoff valve and admits bleed air to the anti-icing duct. The air divides here. One portion enters a nozzle tube on the bottom of the engine intake and is directed through the nozzle upward and around both sides of the intake. A second portion of air leaves the anti-icing duct through a recirculating ejector, to reduce the temperature and pressure, and enters the nacelle nose cowl. Here it circles the lip, the forward fairing between the nacelle openings, and then flows aft through the nacelle to join the first portion of air. These flow down and overboard through louvers in the lower nacelle cowling. A third portion of anti-icing air is diverted from the nozzle tube to channels inside the air splitter and is guided to all forward areas of the splitter. This air escapes through bleed holes in the leading edges and surface of the splitter, to flow into the inlet duct of the engine.

c. The engine section of the anti-icing system receives its hot bleed air from the 9th stage of the engine compressor section. This air flows through a duct on the left side of the engine to a solenoid-actuated shutoff valve at the air inlet housing. When the ENG ANTI-ICE switch is placed in the "ON" position, the valve opens to discharge the hot air through the hollow inlet guide vanes and the bullet nose, and then into the engine intake.

d. An ENG ANTI-ICING VALVE OPEN WHEN LIT indicator light for each engine is mounted on the pilot's instrument panel. Each light illuminates independently of the others when both the nacelle anti-icing valve and the engine anti-icing valve for its respective engine are open. Since one switch controls the anti-icing valves for two engines, the lights will normally go on and off in pairs. On airplanes 5003, 5036 and up, an indicator light relay is connected in the circuit to each pair of indicator lights. This relay causes an indicator light to illuminate if either the nacelle valve or the engine valve fails to close when the switch is turned "OFF."

CAUTION

In cases where an aircraft is to be inactive with power applied for extended periods, it is recommended that the four ENG ANTI-ICE PWR circuit breakers be pulled to protect the anti-icing valve solenoid from overheating.

30.7.2 OPERATIONAL CHECKOUT.

a. Start all four engines. (Individual systems may be checked by running one engine at a time.)

b. Close the four ENG ANTI-ICE PWR circuit breakers.

c. Place the ENG ANTI-ICING switch for engines No. 1 and 4 in the "ON" position.
RESULT: The ENG ANTI-ICING VALVE OPEN WHEN LIT light for engines No. 1 and 4 should go on.

d. Place the ENG ANTI-ICING switch for engine No. 1 and 4 in the "OFF" position.
RESULT: The ENG ANTI-ICING VALVE OPEN WHEN LIT lights should go off.

e. Repeat steps (c) and (d) for the No. 2 and 3 ENG ANTI-ICING switch.

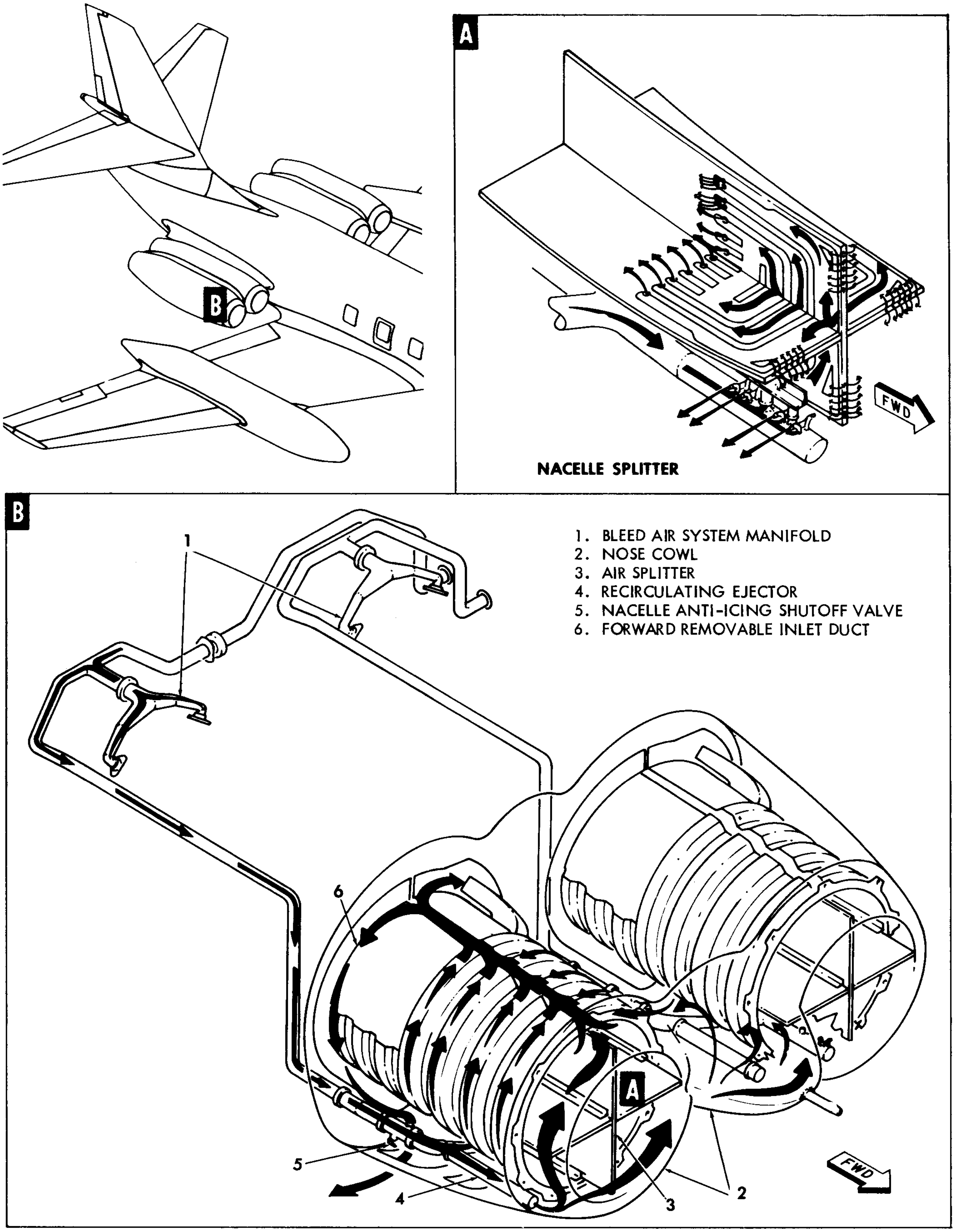
CAUTION

Prolonged engine operation at idle speed during severe icing conditions should be avoided. Even with the anti-icing system ON some ice will accumulate and can result in damage to the engine.

f. Shut down engines.

30.7.3 REMOVAL OF NACELLE ANTI-ICING SHUT-OFF VALVE. After removal of the lower forward nacelle access panel, disconnect electrical connector and remove the two V-Band couplings.

30.7.4 INSTALLATION OF NACELLE ANTI-ICING SHUT-OFF VALVE. Position valve between the tube flange ends and loosely attach V-Band couplings, using care to see that arrow, which shows the direction of flow, shown on the valve, faces forward. The portion of the electrical connector should be about the same as when the valve was removed so that the connection with the harness can be made without strain. The valve, when installed, should not touch any nacelle structure. V-Band installation torque is 30 to 40 inch pounds. Reinstall access panel.


A
B
NACELLE SPLITTER
FWD
1
2
3
4
5
6
1. BLEED AIR SYSTEM MANIFOLD
2. NOSE COWL
3. AIR SPLITTER
4. RECIRCULATING EJECTOR
5. NACELLE ANTI-ICING SHUTOFF VALVE
6. FORWARD REMOVABLE INLET DUCT

JS-2-2-3010

Figure 30-10. Nacelle Anti-Icing Air Flow

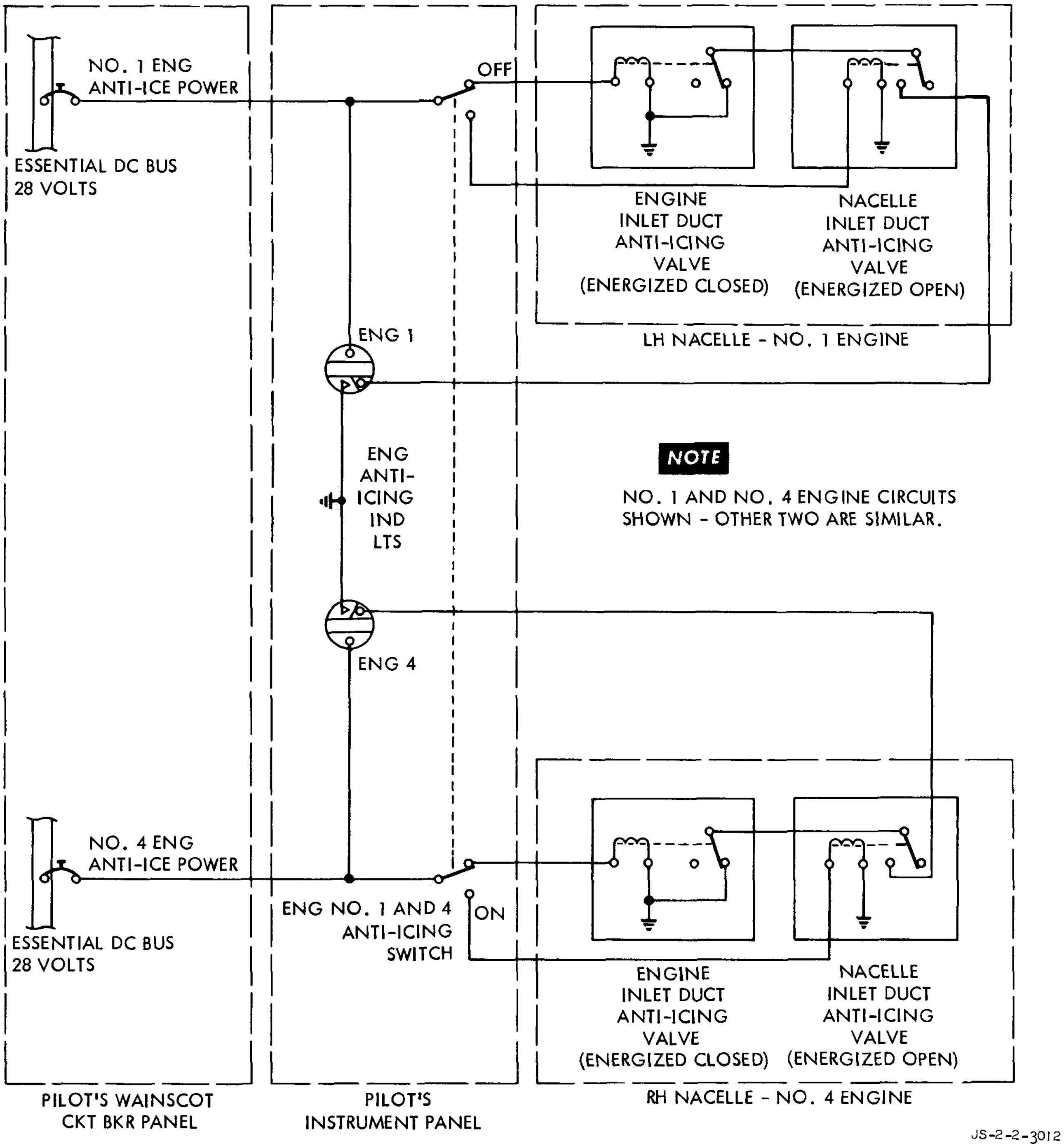


Figure 30-11. Nacelle and Engine Inlet Duct Anti-Icing Schematic Diagram (Airplanes 5001, 5002, and 5004 thru 5035)

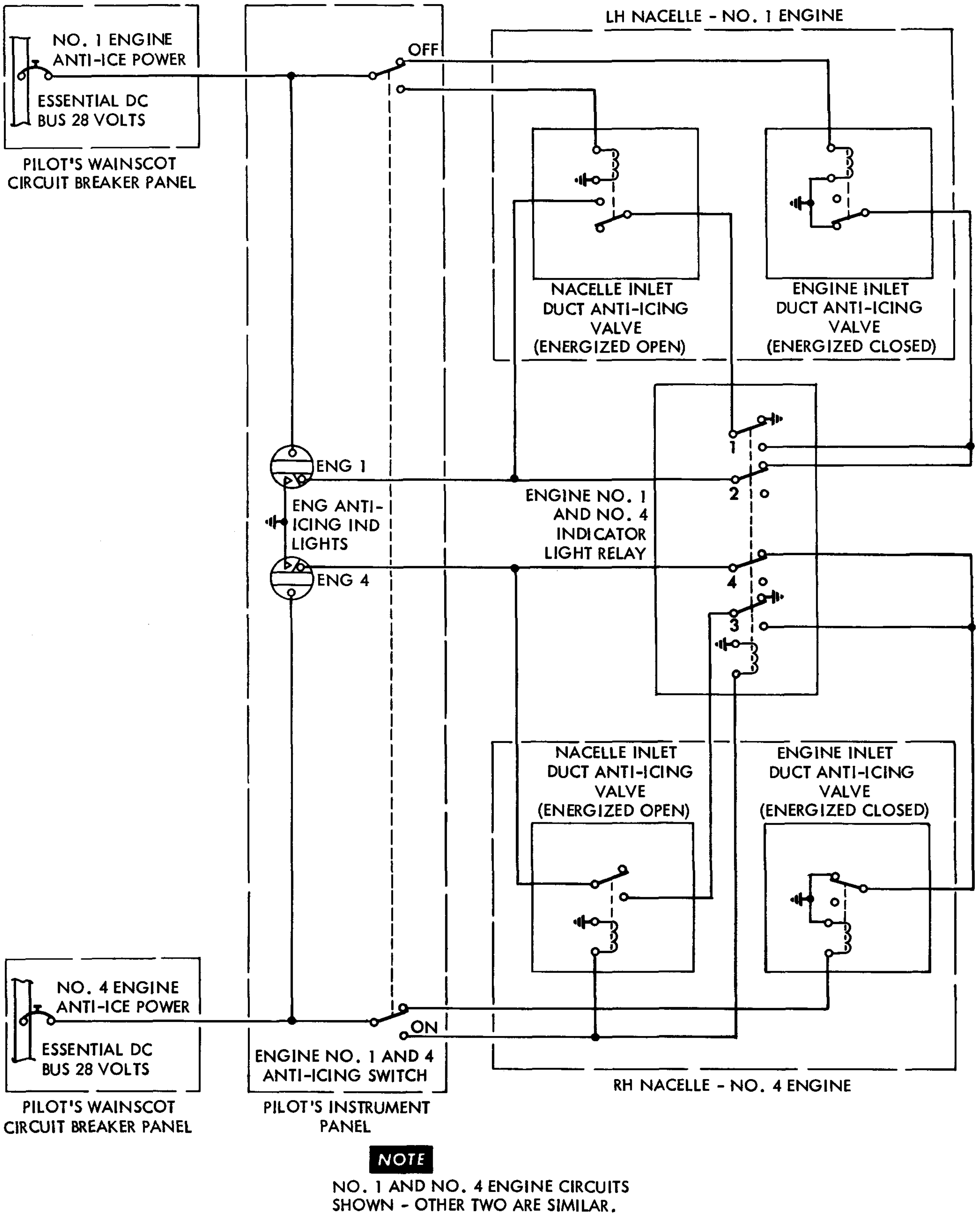


Figure 30-12. Nacelle and Engine Inlet Duct Anti-Icing Schematic Diagram (Airplanes 5003, 5036 and Up)

CHAPTER 31
INSTRUMENTS

TABLE OF CONTENTS

Contents Page

GENERAL 31-1

MAIN INSTRUMENT PANEL 31-2
  Pilot's Instrument Panel 31-2
  Center Instrument Panel 31-2
  Copilot's Instrument Panel 31-2

GLARE SHIELD 31-2

INCLINED PANELS 31-2
  Pilot's Inclined Panel 31-2
  Copilot's Inclined Panel 31-2

SIDE CONSOLE PANELS 31-2
  Pilot's Side Console 31-2
  Copilot's Side Console 31-2

CENTER CONSOLE 31-2

OVERHEAD CONSOLE PANEL 31-2

MISCELLANEOUS INSTRUMENTS 31-2
  Clocks 31-2
  Standby Compass 31-2
  Adjustment 31-2
    Standby Compass Swinging and Compensation 31-2
    Standby Compass Reswing 31-3

FIGURES
  31-1. Main Instrument Panel, Glare Shield, and Inclined Panels (Basic Configuration) (Airplanes 5001 through 5049, 5051, 5056, and 5059) 31-4
  31-1A. Main Instrument Panel, Glare Shield, and Inclined Panels (Basic Configuration) (Airplanes 5050, 5052 through 5055, 5057, 5058, 5060 through 5085) 31-6A
  31-1B. Main Instrument Panel, Glare Shield and Inclined Panels (Basic Configuration) (Airplanes 5086 and Up) 31-6E
  31-2 Side Console Panels (Basic Configuration) 31-7
  31-3 Center Console (Basic Configuration) (Airplanes 5001 through 5049, 5051, 5056, and 5059) 31-9
  31-3A. Center Console (Basic Configuration) (Airplanes 5050, 5052 through 5055, 5057, 5058, 5060 and Up) 31-10A
  31-4 Overhead Control Panel (Basic Configuration) (Airplanes 5001 through 5086) 31-11
  31-4A. Overhead Control Panel (Basic Configuration) (Airplanes 5086 and Up) 31-13

31.1 GENERAL. Instrumentation and controls found in the flight station are located on the main instrument panel, glare shield, inclined panels, side console panels, center console, and the overhead console panel. All panels are visible to either the pilot or copilot or both, and are secured to the airplane structure. With the exception of miscellaneous instruments, the following paragraphs refer to the applicable figure which locates the instrument, or control, and lists the chapter containing applicable

31.1 (Cont).

information for that component. Information concerning miscellaneous instruments is found within this chapter.

31.2 MAIN INSTRUMENT PANEL. The main instrument panel in the flight station consists of the pilot's, center, and copilot's instrument panels.

31.2.1 PILOT'S INSTRUMENT PANEL. See figure 31-1 for location of components and applicable information.

31.2.2 CENTER INSTRUMENT PANEL. See figure 31-1 for location of components and applicable information.

31.2.3 COPILOT'S INSTRUMENT PANEL. See figure 31-1 for location of components and applicable information.

31.3 GLARE SHIELD. The glare shield is located above the main instrument panel. See figure 31-1 for location of components and applicable information.

31.4 INCLINED PANELS. The inclined panels consist of two panels, a pilot's and a copilot's, one mounted to the left and one mounted to the right of the main instrument panel.

31.4.1 PILOT'S INCLINED PANEL. See figure 31-1 for location of components and applicable information.

31.4.2 COPILOT'S INCLINED PANEL. See figure 31-1 for locations of components and applicable information.

31.5 SIDE CONSOLE PANELS. The side console panels consist of two panels, a pilot's and a copilot's, one mounted to the left of the pilot and one mounted to the right of the copilot.

31.5.1 PILOT'S SIDE CONSOLE. See figure 31-2 for location of components and applicable information.

31.5.2 COPILOT'S SIDE CONSOLE. See figure 31-2 for location of components and applicable information.

31.6 CENTER CONSOLE. The center console is mounted between the pilot and copilot. See figure 31-3 for location of components and applicable information.

▌31.7. OVERHEAD CONSOLE PANEL. The overhead control panel is mounted between and above the pilot and copilot. See figure 31-4 for location of components and applicable information.

31.9 MISCELLANEOUS INSTRUMENTS.

31.9.1 CLOCKS. (See figure 31-1, item 3.) Both the pilot and the copilot are provided with a type A-13A clock on their respective sides of the main instrument panel. Each clock is an 8-day type, wound and set with a knob on the lower left-hand corner of the instrument. An elapsed time knob at the upper right-hand corner can be pushed to either (a) start, (b) stop, or (c) return to normal, an elapsed time pointer in the same sequence. The sweep second hand operates only with the elapsed time function.

31.9.2 STANDBY COMPASS. (See figure 31-1, item 61.) A magnetic standby compass is provided. The standby compass, on the center of the glareshield can be seen easily by both the pilot and the copilot. The ENGINE, FUEL PANEL & CENTER STAND INSTRUMENT LIGHTS rheostat controls the self-contained illumination of the compass. The magnetic standby compass is a self-contained instrument used if the electronic compass fails. The unit is basically a pivoted magnetic type having magnetic-type compensators. The magnet is attached to a drum-heading dial. Aircraft magnetic heading is displayed through a window-cursor type indicator. The case contains a liquid to dampen any rapid movements. Magnetic screw type compensators, one on the left for N and S and one the right for E and W, are located behind a cover plate on the front of the instrument. These provide for any compass compensation that may be required to correct for magnetic field distortion caused by airplane parts.

31.9.3 ADJUSTMENT.

31.9.3.1 STANDBY COMPASS SWINGING AND COMPENSATION. The Standby Compass should be compensated by the coefficient method as described below. This operation should be performed on an FAA approved compass rose.

Note

Newly installed standby compasses must be swung and compensated for possible magnetic field distortion caused by airplane parts and equipment.

a. Connect external power to the airplane (refer to Chapter 24).

b. The screw-type compensators should be removed or set to null effect with a non-magnetic screwdriver. Universal screw-type compensators may be set to null effect by turning both adjustment screws until the dots on the screws are matched with the dots on the compensator case.

c. With power on, tow the airplane around the rose approximately four or five times to remove any residual magnetism in any soft iron structure.

Note

Personnel engaged in compensating the compasses must be sure all magnetic tools and maintenance material are removed from their persons and from the airplane. A non-magnetic screwdriver must be used for compensating.

31.9.3.1 (Cont).

d. The compass swings should be performed in the following manner:

1. Place the airplane, by the most suitable method, in 0 (±1), 90 (±1), 180 (±1), and 270 (±) degree magnetic headings. Note and record each compass deviation reading.

   (a) Align the airplane within ±1.0 degree of each specified heading. Determine the actual heading of the airplane within ±0.05 degree, using a transit with the airplane's centerline as reference.

Note

A transit can be mounted along the top centerline of the airplane at fuselage station 398, using an adapter and screws that are longer than normal. Otherwise, the transit can be taped in position. Alignment can be accomplished by sighting aft on the screw at fuselage station 572 and forward on the screw at the trailing edge of the No. 2 VHF Communication antenna.

   (b) When taking readings on the standby compass, energize the following systems:

      (1) Battery
      (2) Windshield Inverter
      (3) Autopilot
      (4) VHF Communication
      (5) Flight Director Systems
      (6) Glideslope Systems
      (7) Windshield De-Icing
      (8) Air Traffic Control Transponder
      (9) Main Inverter(s)
      (10) Boost Pumps
      (11) All the Pilots' Compartments Lights
      (12) VOR-localizer Systems
      (13) Weather Radar
      (14) Automatic Direction Finder Systems
      (15) DME (Standby)

   (c) Be sure all the components of the airplane which could influence the standby compass are installed properly, otherwise errors might be introduced after compensation.

   (d) Algebraically calculate the coefficients A, B, and C from the following formulas. Note here that N, E, S, and W are deviations from the actual heading, and not the indicated headings. Deviations are considered positive when the indicated readings are less than the actual heading, and negative when greater.

$$A = 1/4\ (N+E+S+W) \quad B = 1/2\ (E-W) \quad C = 1/2\ (N-S)$$

   (e) If the coefficent A for the standby compass exceeds 1 degree, realign it with respect to the main instrument panel by means of non-magnetic washers and spacers.

2. Using the computed values of B and C, perform the following compensations:

   (a) While at 0 (±1) or 180 (±1) degrees, make a N-S compensation on the standby compass, according to the magnitude and direction of coefficient C.

   (b) While at 90 (±1) or 270 (±1) degrees, make an E-W compensation on the standby compass, according to the magnitude and direction of coefficient B.

   (c) While at 270 (±1) degrees, adjust the standby compass so as to change the readings according to the magnitude and direction of coefficient A.

3. Make another compass swing of all cardinal headings to ensure that the compensation is satisfactory. If it is not, repeat step (2), using new values of B and C, until all of the compass headings check out.

4. Turn the airplane to eight different magnetic headings, 45 degrees apart, beginning at 0 degrees. At each heading make the required entries on the standby compass correction card.

31.9.3.2 STANDBY COMPASS RESWING. After the original compass swinging and compensation, the standby compass can be compensated to the gyrosyn compass systems. This check can be done on or off a compass rose, provided there are no metal objects within a 250-foot radius.

ITEM	NOMENCLATURE	TEXT REFERENCE
1	EMERGENCY GEAR RELEASE TEE HANDLE.	SEE CHAPTER 32.
2	NOSE LANDING GEAR STEERING WHEEL.	SEE CHAPTER 32.
3	CLOCK.	SEE CHAPTER 31.
4	TURN AND SLIP INDICATOR.	SEE CHAPTER 34.
5	AIRSPEED INDICATOR.	SEE CHAPTER 34.
6	GYROSYN COMPASS (RMI) INDICATOR.	SEE CHAPTER 34.
7	FOOT WARMER HANDLE.	SEE CHAPTER 21.
8	TURN AND SLIP POWER INDICATOR.	SEE CHAPTER 34.
9	DISTANCE MEASURING INDICATOR.	SEE CHAPTER 34.
10	ENGINE ANTI-ICING SWITCHES.	SEE CHAPTER 30.
11	SLAVE COMPASS SWITCH.	SEE CHAPTER 34.
12	ENGINE ANTI-ICING VALVE OPEN WHEN LIT INDICATOR LIGHT.	SEE CHAPTER 30.
13	RADIO CALL NAMEPLATE.	---
14	AFT EQUIPMENT COMPARTMENT OVERHEAT INDICATOR LIGHT.	SEE CHAPTER 33.
15	APPROACH HORIZON INDICATOR.	SEE CHAPTER 34.
16	COURSE INDICATOR.	SEE CHAPTER 34.
17	SPEED LIMITATION PLACARD.	---
18	MASTER CAUTION INDICATOR LIGHT	SEE CHAPTER 33.
19	ALTIMETER INDICATOR.	SEE CHAPTER 34.
20	VERTICAL VELOCITY (RATE OF CLIMB) INDICATOR.	SEE CHAPTER 34.
21	MACH METER INDICATOR.	SEE CHAPTER 34.
22	WEATHER RADAR SCOPE.	SEE CHAPTER 34.
23	EMERGENCY LANDING GEAR SWITCH △2 LANDING GEAR △2	SEE CHAPTER 32.
24	DRAG CHUTE HANDLE.	SEE CHAPTER 39.
25	ENGINE FIRE EXTINGUISHING AGENT SWITCH.	SEE CHAPTER 26.
26	FIRE DETECTION WARNING LIGHT TEST SWITCH.	SEE CHAPTER 26.
27	FIRE WARNING PLACARD.	SEE CHAPTER 26.
28	ENGINE FIRE EXTINGUISHING PULL HANDLES (ENG 1, 2, 3, AND 4).	SEE CHAPTER 26.
29	ENGINE PRESSURE RATIO INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
30	ENGINE TACHOMETER INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
31	ENGINE EXHAUST GAS TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
32	ENGINE FUEL FLOW INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
33	ANNUNCIATOR TEST SWITCH.	SEE CHAPTER 33.
34	ANNUNCIATOR PANEL.	SEE CHAPTER 33.
35	HYDRAULIC PRESSURE INDICATOR.	SEE CHAPTER 29.
36	LEADING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
37	LANDING GEAR CONTROL PANEL.	SEE CHAPTER 32.
38	HYDRAULIC POWER SELECTOR SWITCH (FLAPS, BRAKES, AND STEERING).	SEE CHAPTER 29.
39	TRAILING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
40	CABIN INTERPHONE INDICATOR LIGHT.	SEE CHAPTER 23.
41	AUXILIARY HYDRAULIC SYSTEM CONTROL SWITCH.	SEE CHAPTER 29.
42	OUTSIDE AIR TEMPERATURE INDICATOR.	SEE CHAPTER 34.
43	ENGINE OIL TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
44	FUEL JETTISON SWITCH.	SEE CHAPTER 28.
45	ENGINE OIL PRESSURE INDICATORS (ENG 1 & 2, 3 & 4).	SEE CHAPTER 77.
46	FUEL TANK ISOLATION SWITCHES.	SEE CHAPTER 28.
47	WINDSHIELD ANTI-ICING POWER SELECTOR SWTICH.	SEE CHAPTER 30.
48	WINDSHIELD ANTI-ICING HEAT SELECTOR SWITCH.	SEE CHAPTER 30.
49	WINDSHIELD ANTI-ICING PUSH-TO-START SWITCH.	SEE CHAPTER 30.
50	ENGINE FUEL HEATER SWITCHES (ENG 1 & 4, 2 & 3).	SEE CHAPTER 73.
51	ENGINE WARNING LIGHTS (FUEL FILTER CLOGGING AND FUEL HEAT ON FOR ENG 1, 2, 3 AND 4).	SEE CHAPTER 73.
52	MACH TRIM COMPENSATOR OUT INDICATOR LIGHT.	SEE CHAPTER 27.
53	PITOT HEAT SWITCH.	SEE CHAPTER 30.

Figure 31-1. Main Instrument Panel, Glare Shield, and Inclined Panels (Basic Configuration) (Sheet 1 of 2)

(Airplanes 5001 thru 5049, 5051, 5056 & 5059)

ITEM	NOMENCLATURE.	TEXT REFERENCE
54	ENGINE OIL PRESSURE LOW ANNUNCIATOR LIGHT CUT-OFF SWITCHES (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
55	AUTOPILOT AND Q SWITCH PITOT SHUTOFF SWITCH.	SEE CHAPTER 34.
56	OVERSPEED WARNING TEST SWITCH.	SEE CHAPTER 34.
57	MARKER BEACON LIGHTS.	SEE CHAPTER 34.
58	SYSTEM NO. 2 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.
59	SYSTEM NO. 1 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.
60	THRUST REVERSERS EXTENDED INDICATOR LIGHTS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 78.
61	STANDBY COMPASS	SEE CHAPTER 31.
62	GASPER OUTLET	SEE CHAPTER 21

⚠1 INSTALLED BY LOCKHEED ON AIRPLANES 5001 THROUGH 5045. ON AIRPLANES 5046 AND UP THE AIRPLANE IS RELEASED TO THE CUSTOMER WITHOUT THIS EQUIPMENT.

⚠2 AIRPLANES 5001 THROUGH 5040 THAT HAVE NOT HAD SERVICE BULLETIN 329-25B INCORPORATED.

⚠3 AIRPLANES 5046 AND UP

Figure 31-1. Main Instrument Panel, Glare Shield and Inclined Panels (Basic Configuration)
(Sheet 2 of 2)
(Airplanes 5001 thru 5049, 5051, 5056 & 5059)

ITEM	NOMENCLATURE	TEXT REFERENCE
1	EMERGENCY GEAR RELEASE TEE HANDLE.	SEE CHAPTER 32.
2	NOSE LANDING GEAR STEERING WHEEL.	SEE CHAPTER 32.
3	CLOCK.	SEE CHAPTER 31.
4	TURN AND SLIP INDICATOR.	SEE CHAPTER 34.
5	AIRSPEED INDICATOR.	SEE CHAPTER 34.
6	GYROSYN COMPASS (RMI) INDICATOR. △1	SEE CHAPTER 34.
7	FOOT WARMER HANDLE.	SEE CHAPTER 21.
8	TURN AND SLIP POWER INDICATOR.	SEE CHAPTER 34.
9	GASPER OUTLET	SEE CHAPTER 21.
10	ENGINE ANTI-ICING SWITCHES.	SEE CHAPTER 30.
11	SLAVE COMPASS SWITCH.	SEE CHAPTER 34.
12	ENGINE ANTI-ICING VALVE OPEN WHEN LIT INDICATOR LIGHT.	SEE CHAPTER 30.
13	RADIO CALL NAMEPLATE.	---
14	AFT EQUIPMENT COMPARTMENT OVERHEAT INDICATOR LIGHT. △1	SEE CHAPTER 33.
15	APPROACH HORIZON INDICATOR. △1	SEE CHAPTER 34.
16	COURSE INDICATOR △1	SEE CHAPTER 34.
17	SPEED LIMITATION PLACARD.	---
18	MASTER CAUTION INDICATOR LIGHT.	SEE CHAPTER 33.
19	ALTIMETER INDICATOR.	SEE CHAPTER 34.
20	VERTICAL VELOCITY (RATE OF CLIMB) INDICATOR.	SEE CHAPTER 34.
21	MACH METER INDICATOR.	SEE CHAPTER 34.
22	WEATHER RADAR SCOPE. △1	SEE CHAPTER 34.
23	DRAG CHUTE HANDLE GUARD △2	SEE CHAPTER 39.
24	DRAG CHUTE HANDLE.	SEE CHAPTER 39.
25	ENGINE FIRE EXTINGUISHING AGENT SWITCH.	SEE CHAPTER 26.
26	FIRE DETECTION WARNING LIGHT TEST SWITCH.	SEE CHAPTER 26.
27	FIRE WARNING PLACARD.	SEE CHAPTER 26.
28	ENGINE FIRE EXTINGUISHING PULL HANDLES (ENG 1, 2, 3, AND 4).	SEE CHAPTER 26.
29	ENGINE PRESSURE RATIO INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
30	ENGINE TACHOMETER INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
31	ENGINE EXHAUST GAS TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
32	ENGINE FUEL FLOW INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
33	ANNUNCIATOR TEST SWITCH.	SEE CHAPTER 33.
34	ANNUNCIATOR PANEL.	SEE CHAPTER 33.
35	HYDRAULIC PRESSURE INDICATOR.	SEE CHAPTER 29.
36	LEADING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
37	LANDING GEAR CONTROL PANEL.	SEE CHAPTER 32.
38	HYDRAULIC POWER SELECTOR SWITCH (FLAPS, BRAKES, AND STEERING).	SEE CHAPTER 29.
39	TRAILING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
40	CABIN INTERPHONE INDICATOR LIGHT.	SEE CHAPTER 23.
41	AUXILIARY HYDRAULIC SYSTEM CONTROL SWITCH.	SEE CHAPTER 29.
42	OUTSIDE AIR TEMPERATURE INDICATOR.	SEE CHAPTER 34.
43	ENGINE OIL TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
44	FUEL JETTISON SWITCH.	SEE CHAPTER 28.
45	ENGINE OIL PRESSURE INDICATORS (ENG 1 & 2, 3 & 4).	SEE CHAPTER 77.
46	FUEL TANK ISOLATION SWITCHES.	SEE CHAPTER 28.
47	WINDSHIELD ANTI-ICING POWER SELECTOR SWTICH.	SEE CHAPTER 30.
48	WINDSHIELD ANTI-ICING HEAT SELECTOR SWITCH.	SEE CHAPTER 30.
49	WINDSHIELD ANTI-ICING PUSH-TO-START SWITCH.	SEE CHAPTER 30.
50	ENGINE FUEL HEATER SWITCHES (ENG 1 & 4, 2 & 3).	SEE CHAPTER 73.
51	ENGINE WARNING LIGHTS (FUEL FILTER CLOGGING AND FUEL HEAT ON FOR ENG 1, 2, 3 AND 4).	SEE CHAPTER 73.
52	MACH TRIM COMPENSATOR OUT INDICATOR LIGHT.	SEE CHAPTER 27.
53	PITOT HEAT SWITCH.	SEE CHAPTER 30.

Figure 31-1A. Main Instrument Panel, Glare Shield, and Inclined Panels (Basic Configuration) Airplanes 5050,5052 Thru 5055,5057, 5058,5060 thru 5085) (Sheet 1 of 2)

ITEM	NOMENCLATURE	TEXT REFERENCE
54	ENGINE OIL PRESSURE LOW ANNUNCIATOR LIGHT CUT-OFF SWITCHES (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
55	AUTOPILOT AND Q SWITCH PITOT SHUTOFF SWITCH.	SEE CHAPTER 34.
56	OVERSPEED WARNING TEST SWITCH.	SEE CHAPTER 34.
57	MARKER BEACON LIGHTS.	SEE CHAPTER 34.
58	SYSTEM NO. 2 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.
59	SYSTEM NO. 1 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.
60	THRUST REVERSERS EXTENDED INDICATOR LIGHTS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 78.
61	STANDBY COMPASS	SEE CHAPTER 31.

⚠1 AIRPLANE IS RELEASED TO CUSTOMER WITHOUT THIS EQUIPMENT

⚠2 AIRPLANES 5065 THRU 5085

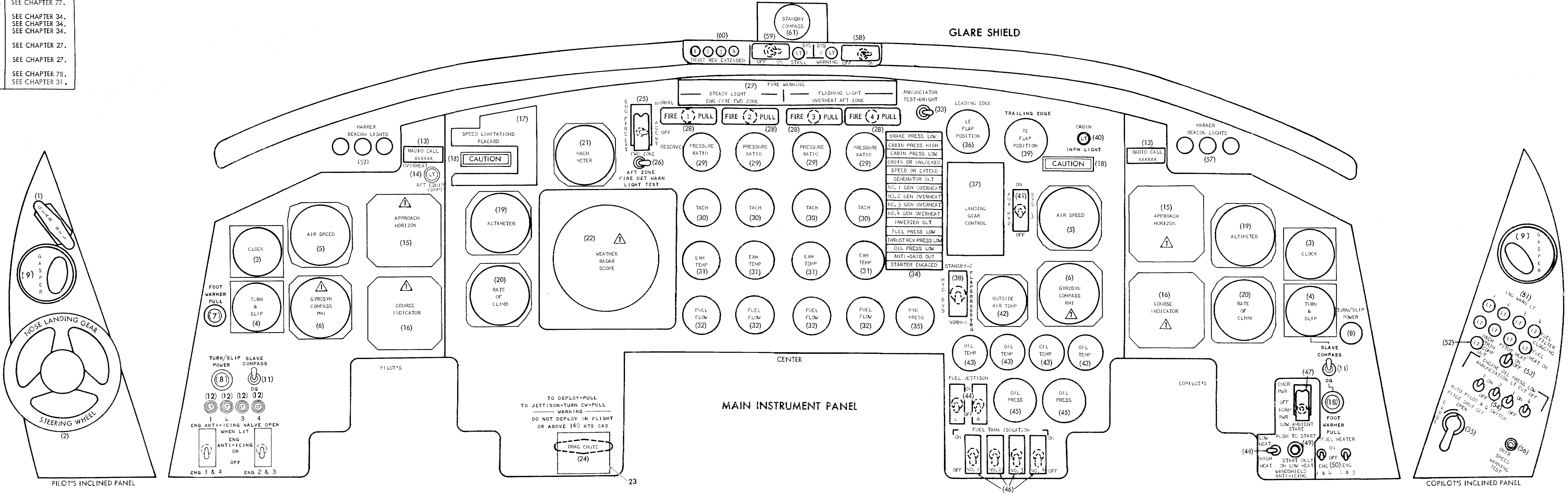


Figure 31-1A. Main Instrument Panel, Glare Shield Inclined Panels (Basic Configuration) Airplanes 5050, 5052 Thru 5055, 5057, 5058, 5060 thru 5085 (Sheet 2of 2)

Changed 1 June 1966

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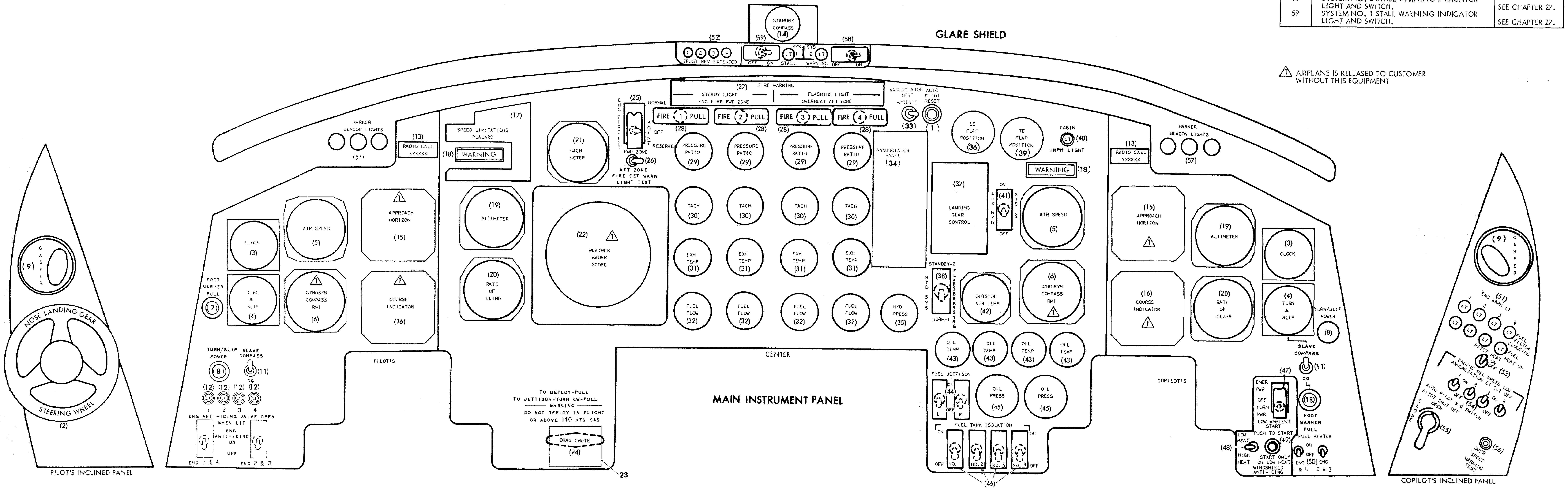
1	AUTO PILOT RESET	SEE CHAPTER
2	NOSE LANDING GEAR STEERING WHEEL.	SEE CHAPTER 32.
3	CLOCK.	SEE CHAPTER 31.
4	TURN AND SLIP INDICATOR.	SEE CHAPTER 34.
5	AIRSPEED INDICATOR.	SEE CHAPTER 34.
6	GYROSYN COMPASS (RMI) INDICATOR.△1	SEE CHAPTER 34.
7	FOOT WARMER HANDLE.	SEE CHAPTER 21.
8	TURN AND SLIP POWER INDICATOR.	SEE CHAPTER 34.
9	GASPER OUTLET	SEE CHAPTER 21.
10	ENGINE ANTI-ICING SWITCHES.	SEE CHAPTER 30.
11	SLAVE COMPASS SWITCH.	SEE CHAPTER 34.
12	ENGINE ANTI-ICING VALVE OPEN WHEN LIT INDICATOR LIGHT.	SEE CHAPTER 30.
13	RADIO CALL NAMEPLATE.	---
14	STANDBY COMPASS.	SEE CHAPTER 31.
15	APPROACH HORIZON INDICATOR.△1	SEE CHAPTER 34.
16	COURSE INDICATOR △1	SEE CHAPTER 34.
17	SPEED LIMITATION PLACARD.	---
18	MASTER CAUTION INDICATOR LIGHT.	SEE CHAPTER 33.
19	ALTIMETER INDICATOR.	SEE CHAPTER 34.
20	VERTICAL VELOCITY (RATE OF CLIMB) INDICATOR.	SEE CHAPTER 34.
21	MACH METER INDICATOR.	SEE CHAPTER 34.
22	WEATHER RADAR SCOPE. △1	SEE CHAPTER 34.
23	DRAG CHUTE HANDLE GUARD	SEE CHAPTER 39.
24	DRAG CHUTE HANDLE.	SEE CHAPTER 39.
25	ENGINE FIRE EXTINGUISHING AGENT SWITCH.	SEE CHAPTER 26.
26	FIRE DETECTION WARNING LIGHT TEST SWITCH.	SEE CHAPTER 26.
27	FIRE WARNING PLACARD.	SEE CHAPTER 26.
28	ENGINE FIRE EXTINGUISHING PULL HANDLES (ENG 1, 2, 3, AND 4).	SEE CHAPTER 26.
29	ENGINE PRESSURE RATIO INDICATORS (ENG 1, 2, 3, AND 4)	SEE CHAPTER 77.
30	ENGINE TACHOMETER INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
31	ENGINE EXHAUST GAS TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
32	ENGINE FUEL FLOW INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
33	ANNUNCIATOR TEST SWITCH.	SEE CHAPTER 33.
34	ANNUNCIATOR PANEL.	SEE CHAPTER 33.
35	HYDRAULIC PRESSURE INDICATOR.	SEE CHAPTER 29.
36	LEADING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
37	LANDING GEAR CONTROL PANEL.	SEE CHAPTER 32.
38	HYDRAULIC POWER SELECTOR SWITCH (FLAPS, BRAKES, AND STEERING).	SEE CHAPTER 29.
39	TRAILING EDGE FLAP POSITION INDICATOR.	SEE CHAPTER 27.
40	CABIN INTERPHONE INDICATOR LIGHT.	SEE CHAPTER 23.
41	AUXILIARY HYDRAULIC SYSTEM CONTROL SWITCH.	SEE CHAPTER 29.
42	OUTSIDE AIR TEMPERATURE INDICATOR.	SEE CHAPTER 34.
43	ENGINE OIL TEMPERATURE INDICATORS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 77.
44	FUEL JETTISON SWITCH.	SEE CHAPTER 28.
45	ENGINE OIL PRESSURE INDICATORS (ENG 1 & 2, 3 & 4).	SEE CHAPTER 77.
46	FUEL TANK ISOLATION SWITCHES.	SEE CHAPTER 28.
47	WINDSHIELD ANTI-ICING POWER SELECTOR SWITCH.	SEE CHAPTER 30.
48	WINDSHIELD ANTI-ICING HEAT SELECTOR SWITCH.	SEE CHAPTER 30.
49	WINDSHIELD ANTI-ICING PUSH-TO-START SWITCH.	SEE CHAPTER 30.
50	ENGINE FUEL HEATER SWITCHES (ENG 1 & 4, 2 & 3).	SEE CHAPTER 73.
51	ENGINE WARNING LIGHTS (FUEL FILTER CLOGGING AND FUEL HEAT ON FOR ENG 1, 2, 3, AND 4).	SEE CHAPTER 73.
52	THRUST REVERSERS EXTENDED INDICATOR LIGHTS (ENG 1, 2, 3, AND 4).	SEE CHAPTER 78.
53	PITOT HEAT SWITCH.	SEE CHAPTER 30.

Figure 31-1B. Main Instrument Panel Glare Shield and Inclined Panels (Basic Configuration) Airplanes 5086 and Up (Sheet 1 of 2)

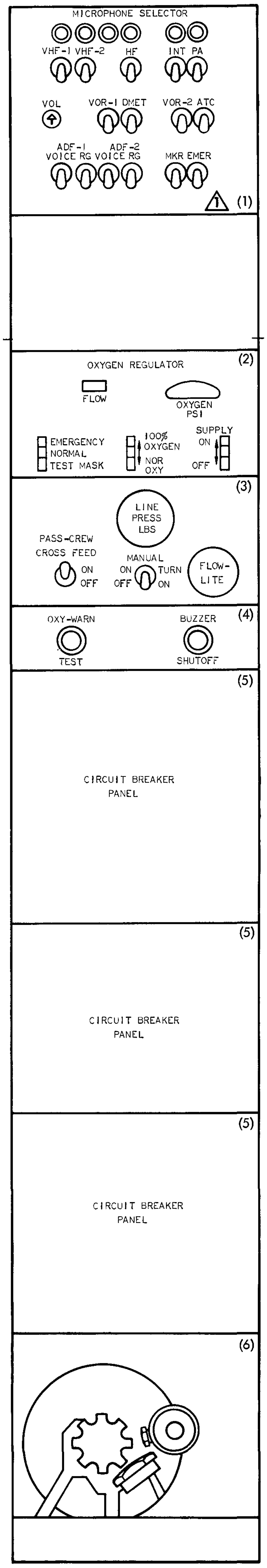
ITEM	NOMENCLATURE	TEXT REFERENCE
54	ENGINE OIL PRESSURE LOW ANNUNCIATOR LIGHT CUT-OFF SWITCHES (ENG 1, 2, 3, AND 4)	SEE CHAPTER 77.
55	AUTOPILOT AND Q SWITCH PITOT SHUTOFF SWITCH.	SEE CHAPTER 34.
56	OVERSPEED WARNING TEST SWITCH.	SEE CHAPTER 34.
57	MARKER BEACON LIGHTS.	SEE CHAPTER 34.
58	SYSTEM NO. 2 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.
59	SYSTEM NO. 1 STALL WARNING INDICATOR LIGHT AND SWITCH.	SEE CHAPTER 27.



⚠1 AIRPLANE IS RELEASED TO CUSTOMER WITHOUT THIS EQUIPMENT

Figure 31-1B. Main Instrument Panel, Glare Shield Inclined Panels (Basic Configuration) Airplanes 5086 and Up (Sheet 2 of 2)

Changed 1 January 1967



ITEM	NOMENCLATURE	TEXT REFERENCE
1	AUDIO SELECTOR PANEL.	SEE CHAPTER 23.
2	CREW OXYGEN CONTROL PANEL.	SEE CHAPTER 35.
3	PASSENGER OXYGEN CONTROL PANEL.	SEE CHAPTER 35.
4	OXYGEN WARNING TEST PANEL.	SEE CHAPTER 35.
5	CIRCUIT BREAKER PANEL.	SEE CHAPTER 24.
6	PORTABLE OXYGEN CYLINDER AND STORAGE COMPARTMENT.	SEE CHAPTER 35.
7	DE-ICING TIMER.	SEE CHAPTER 30.

⚠1 INSTALLED BY LOCKHEED ON AIRPLANES 5001 THRU 5045. ON AIRPLANES 5046 AND UP THE AIRPLANE IS RELEASED TO THE CUSTOMER WITHOUT THIS EQUIPMENT.

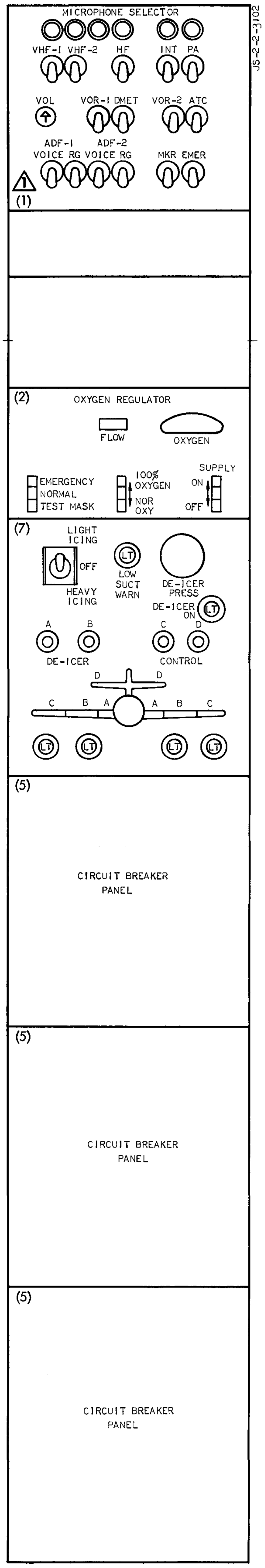
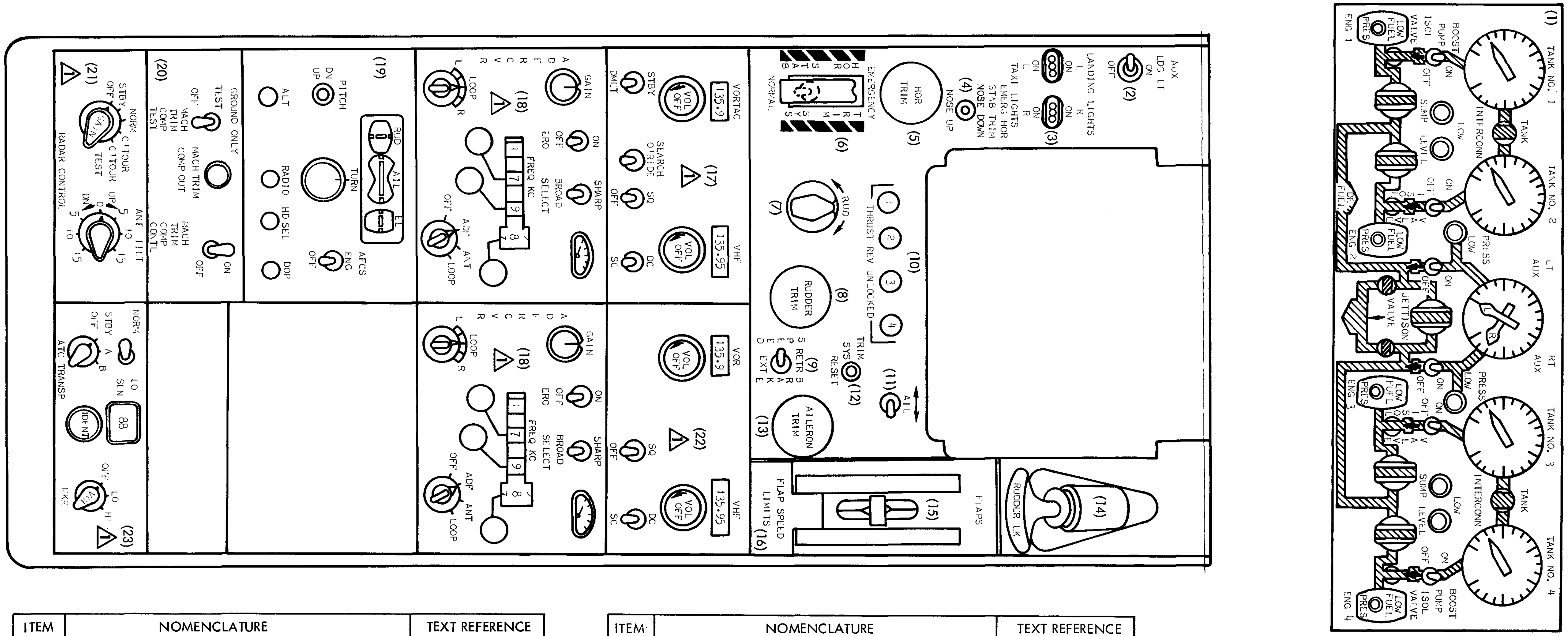


Figure 31-2. Side Console Panels (Basic Configuration)

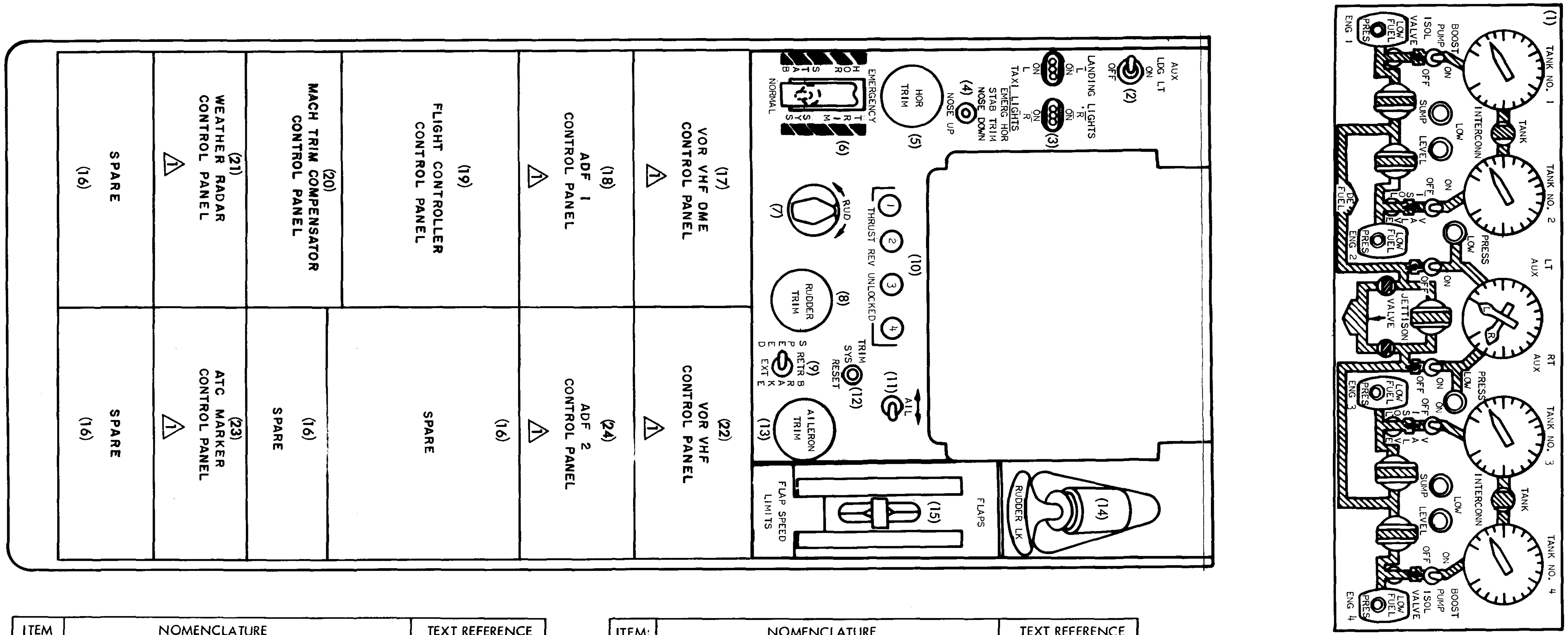


ITEM	NOMENCLATURE	TEXT REFERENCE
1	FUEL MANAGEMENT PANEL.	SEE CHAPTER 28.
2	AUXILIARY LANDING LIGHT SWITCH.	SEE CHAPTER 33.
3	LANDING TAXI LIGHTS SWITCH.	SEE CHAPTER 33.
4	EMERGENCY HORIZONTAL STABILIZER TRIM CONTROL SWITCH.	SEE CHAPTER 27.
5	HORIZONTAL STABILIZER POSITION INDICATOR.	SEE CHAPTER 27.
6	EMERGENCY HORIZONTAL STABILIZER TRIM SYSTEM SWITCH.	SEE CHAPTER 27.
7	RUDDER TRIM TAB CONTROL SWITCH.	SEE CHAPTER 27.
8	RUDDER TRIM TAB INDICATOR.	SEE CHAPTER 27.
9	SPEED BRAKE SWITCH.	SEE CHAPTER 27.
10	THRUST REVERSE UNLOCKED LIGHTS.	SEE CHAPTER 78.
11	AILERON TRIM TAB CONTROL SWITCH.	SEE CHAPTER 27.

ITEM	NOMENCLATURE	TEXT REFERENCE
12	TRIM SYSTEM RESET BUTTON.	SEE CHAPTER 27.
13	AILERON TRIM TAB INDICATOR.	SEE CHAPTER 27.
14	RUDDER LOCK HANDLE.	SEE CHAPTER 27.
15	FLAP CONTROL LEVER.	SEE CHAPTER 27.
16	FLAP SPEED LIMITS PLACARD.	
17	VORTAC - VHF CONTROL PANEL.	SEE CHAPTER 34.
18	ADF RECEIVER CONTROL PANEL.	SEE CHAPTER 34.
19	AUTOPILOT FLIGHT CONTROLLER.	SEE CHAPTER 22.
20	MACH TRIM COMPENSATOR CONTROL PANEL.	SEE CHAPTER 27.
21	WEATHER RADAR CONTROL PANEL.	SEE CHAPTER 34.
22	VOR-VHF CONTROL PANEL.	SEE CHAPTER 34.
23	ATC TRANSPONDER - MARKER CONTROL PANEL.	SEE CHAPTER 34.

⚠1 INSTALLED BY LOCKHEED ON AIRPLANES 5001 THROUGH 5045. ON AIRPLANES 5046 AND UP THE AIRPLANE IS RELEASED TO THE CUSTOMER WITHOUT THIS EQUIPMENT.

Figure 31-3. Center Console (Basic Configuration)
(Airplanes 5001 thru 5049, 5051, 5056 & 5059)

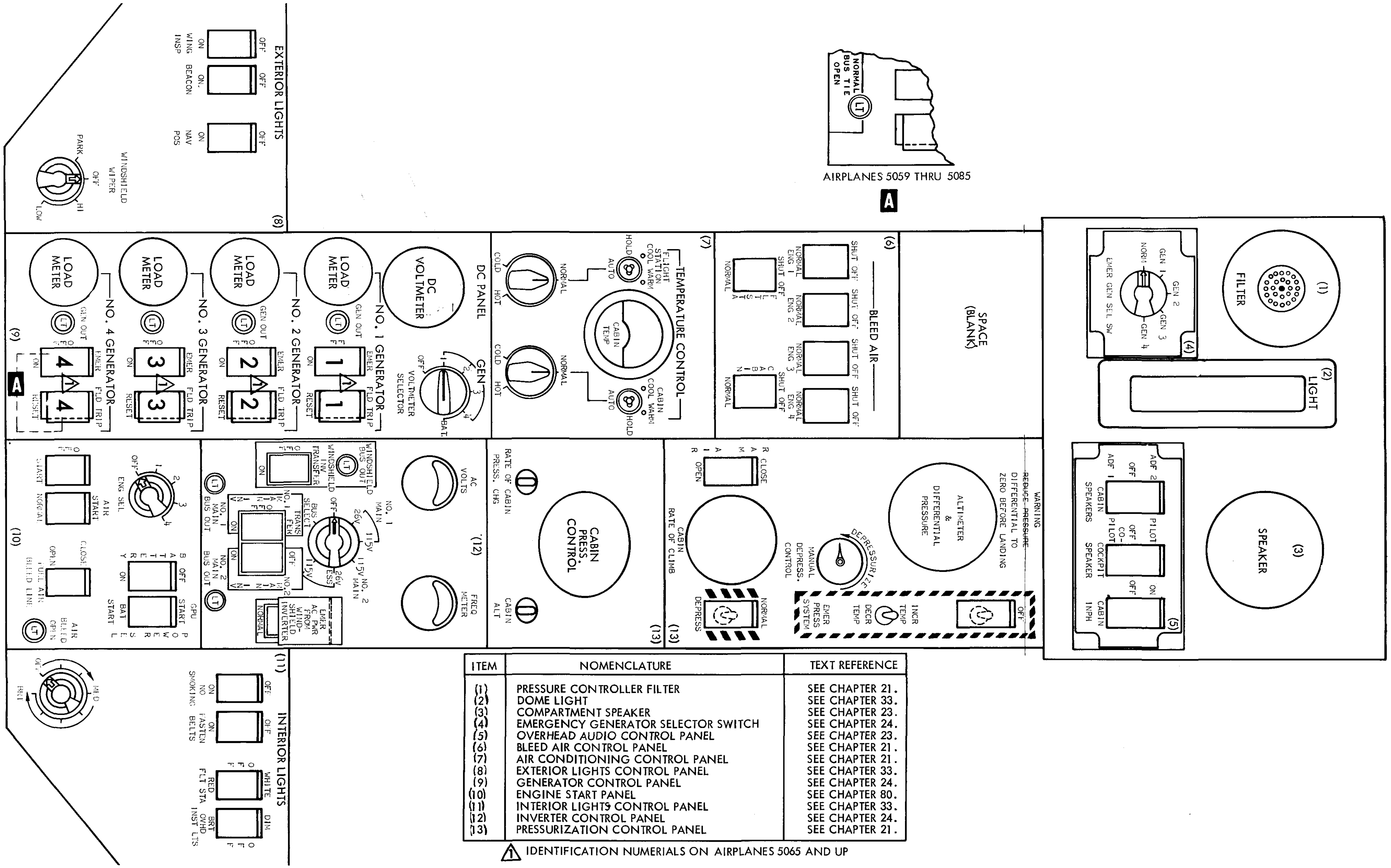


ITEM	NOMENCLATURE	TEXT REFERENCE
1	FUEL MANAGEMENT PANEL.	SEE CHAPTER 28.
2	AUXILIARY LANDING LIGHT SWITCH.	SEE CHAPTER 33.
3	LANDING TAXI LIGHTS SWITCH.	SEE CHAPTER 33.
4	EMERGENCY HORIZONTAL STABILIZER TRIM CONTROL SWITCH.	SEE CHAPTER 27.
5	HORIZONTAL STABILIZER POSITION INDICATOR.	SEE CHAPTER 27.
6	EMERGENCY HORIZONTAL STABILIZER TRIM SYSTEM SWITCH.	SEE CHAPTER 27.
7	RUDDER TRIM TAB CONTROL SWITCH.	SEE CHAPTER 27.
8	RUDDER TRIM TAB INDICATOR.	SEE CHAPTER 27.
9	SPEED BRAKE SWITCH.	SEE CHAPTER 27.
10	THRUST REVERSE UNLOCKED LIGHTS.	SEE CHAPTER 78.
11	AILERON TRIM TAB CONTROL SWITCH.	SEE CHAPTER 27.

ITEM	NOMENCLATURE	TEXT REFERENCE
12	TRIM SYSTEM RESET BUTTON.	SEE CHAPTER 27.
13	AILERON TRIM TAB INDICATOR.	SEE CHAPTER 27.
14	RUDDER LOCK HANDLE.	SEE CHAPTER 27.
15	FLAP CONTROL LEVER.	SEE CHAPTER 27.
16	SPARE	
17	VOR VHF DME CONTROL PANEL	SEE CHAPTER 34.
18	ADF1 CONTROL PANEL	SEE CHAPTER 34.
19	AUTOPILOT FLIGHT CONTROLLER.	SEE CHAPTER 22.
20	MACH TRIM COMPENSATOR CONTROL PANEL.	SEE CHAPTER 27.
21	WEATHER RADAR CONTROL PANEL.	SEE CHAPTER 34.
22	VOR VHF CONTROL PANEL	SEE CHAPTER 34.
23	ATC TRANSPONDER - MARKER CONTROL PANEL.	SEE CHAPTER 34.
24	ADF 2 CONTROL PANEL	SEE CHAPTER 34.

⚠ AIRPLANE IS RELEASED TO CUSTOMER WITHOUT THIS EQUIPMENT

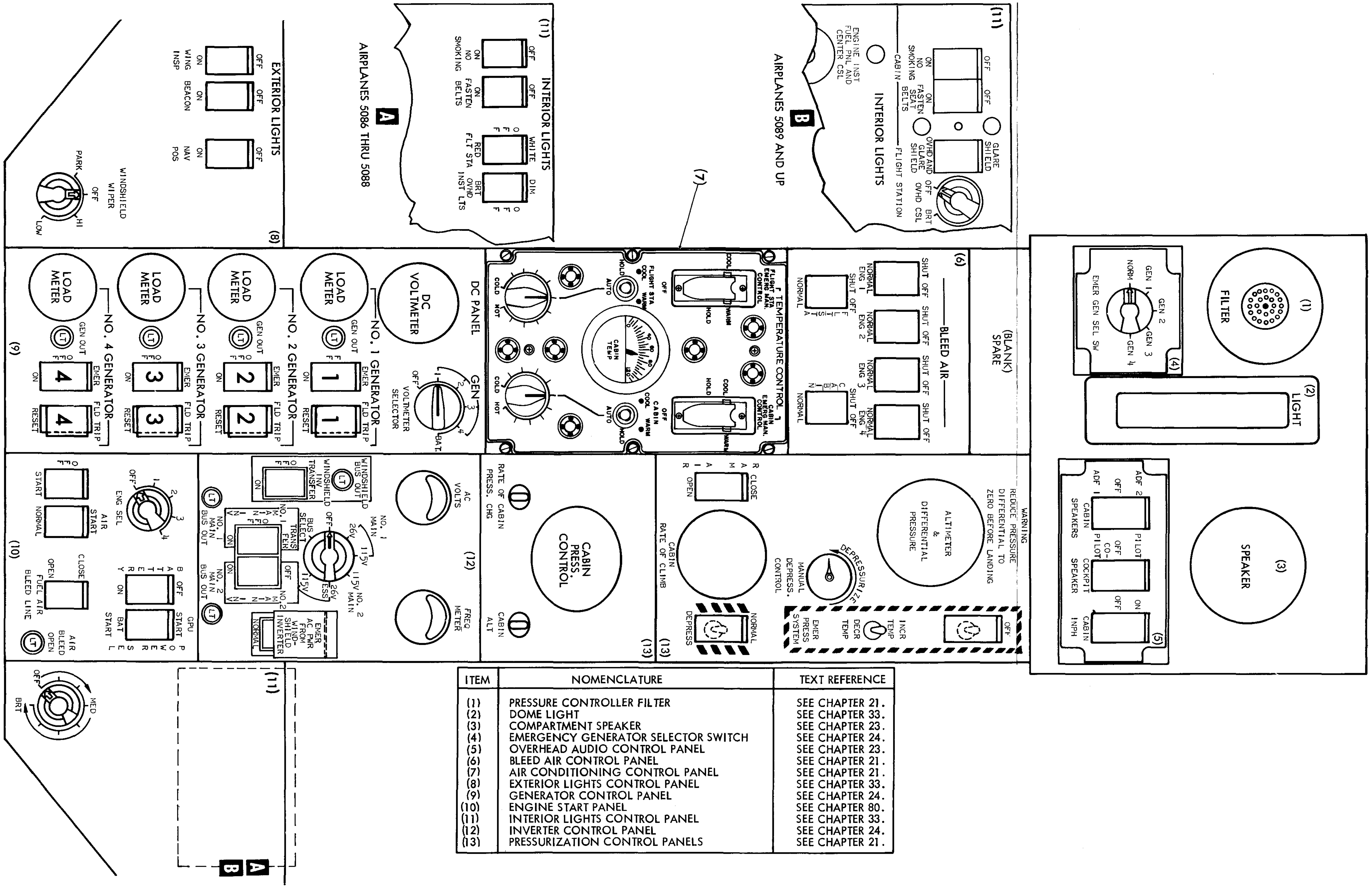
Figure 31-3A. Center Console (Basic Configuration)
Airplanes 5050, 5052 Thru 5055, 5057, 5058, 5059, 5060 and Up



ITEM	NOMENCLATURE	TEXT REFERENCE
(1)	PRESSURE CONTROLLER FILTER	SEE CHAPTER 21.
(2)	DOME LIGHT	SEE CHAPTER 33.
(3)	COMPARTMENT SPEAKER	SEE CHAPTER 23.
(4)	EMERGENCY GENERATOR SELECTOR SWITCH	SEE CHAPTER 24.
(5)	OVERHEAD AUDIO CONTROL PANEL	SEE CHAPTER 23.
(6)	BLEED AIR CONTROL PANEL	SEE CHAPTER 21.
(7)	AIR CONDITIONING CONTROL PANEL	SEE CHAPTER 21.
(8)	EXTERIOR LIGHTS CONTROL PANEL	SEE CHAPTER 33.
(9)	GENERATOR CONTROL PANEL	SEE CHAPTER 24.
(10)	ENGINE START PANEL	SEE CHAPTER 80.
(11)	INTERIOR LIGHTS CONTROL PANEL	SEE CHAPTER 33.
(12)	INVERTER CONTROL PANEL	SEE CHAPTER 24.
(13)	PRESSURIZATION CONTROL PANEL	SEE CHAPTER 21.

△1 IDENTIFICATION NUMERIALS ON AIRPLANES 5065 AND UP

Figure 31-4. Overhead Control Panel (Basic Configuration)(Airplanes 5001 thru 5085)



ITEM	NOMENCLATURE	TEXT REFERENCE
(1)	PRESSURE CONTROLLER FILTER	SEE CHAPTER 21.
(2)	DOME LIGHT	SEE CHAPTER 33.
(3)	COMPARTMENT SPEAKER	SEE CHAPTER 23.
(4)	EMERGENCY GENERATOR SELECTOR SWITCH	SEE CHAPTER 24.
(5)	OVERHEAD AUDIO CONTROL PANEL	SEE CHAPTER 23.
(6)	BLEED AIR CONTROL PANEL	SEE CHAPTER 21.
(7)	AIR CONDITIONING CONTROL PANEL	SEE CHAPTER 21.
(8)	EXTERIOR LIGHTS CONTROL PANEL	SEE CHAPTER 33.
(9)	GENERATOR CONTROL PANEL	SEE CHAPTER 24.
(10)	ENGINE START PANEL	SEE CHAPTER 80.
(11)	INTERIOR LIGHTS CONTROL PANEL	SEE CHAPTER 33.
(12)	INVERTER CONTROL PANEL	SEE CHAPTER 24.
(13)	PRESSURIZATION CONTROL PANELS	SEE CHAPTER 21.

Figure 31-4A. Overhead Control Panel (Basic Configuration) (Airplanes 5086 and Up)

CHAPTER 32
LANDING GEAR

TABLE OF CONTENTS

Section		Page
32.0	LANDING GEAR AND LANDING GEAR DOORS	32-3
32.1	EMERGENCY GEAR RELEASE SYSTEM (AIRPLANES 5001 THRU 5085)	32-51
32.2	LANDING GEAR INDICATING AND WARNING SYSTEM	32-63
32.3	BRAKE AND ANTI-SKID SYSTEMS	32-73
32.4	NOSE STEERING SYSTEM	32-87
32.5	EMERGENCY GEAR EXTENSION SYSTEM (AIRPLANES 5086 AND UP)	32-97

SECTION 32.0
LANDING GEAR AND LANDING GEAR DOORS

Contents Page

DESCRIPTION AND OPERATION 32-5
- General 32-5
- Main Gear Assembly 32-5
- Main Gear Doors 32-5
- Nose Gear Assembly 32-5
- Nose Gear Doors 32-5
- Normal Gear-Up Actuation (Airplanes 5001 thru 5040) 32-5
- Normal Gear-Up Actuation (Airplanes 5041 and up) 32-6
- Normal Gear-Down Actuation (Airplanes 5001 thru 5040) 32-6
- Normal Gear-Down Actuation (For Airplanes 5001 thru 5040 Modified by Service Bulletin 329-291) . . 32-6
- Normal Gear-Down Actuation (Airplanes 5001 thru 5040 Modified by Service Bulletin 329-25B and Airplanes 5041 and Up) 32-7
- Normal Gear-Down Actuation (For Airplanes 5001 thru 5040 When Modified by Service Bulletins 329-25B and 329-291 and Airplanes 5041 and Up When Modified by Service Bulletin 329-291) 32-7
- Landing Gear Control Panel 32-7
- Landing Gear Switches and Relays 32-7
- Main Gear Uplock Assembly 32-7
- Main Gear Downlocks 32-7
- Nose Gear Uplock Assembly 32-7
- Nose Gear Downlock Assembly 32-7
- Main Gear Inboard Door Actuating Cylinders 32-7
- Nose Gear Forward Door Actuating Cylinder 32-7
- Nose Gear Actuating Cylinder 32-7
- Main Gear Actuating Cylinders 32-8
- Landing Gear Selector Valve 32-8
- Landing Gear Door Selector Valve 32-8
- Landing Gear Isolation Valve (Airplanes 5041 and Up) 32-8
- Nose Landing Gear and Landing Gear Door Sequencing Valves 32-8
- Main Gear Uplock Accumulators (Airplanes 5041 and Up) 32-8
- Main Gear Wheels 32-8
- Main Gear Tires 32-8
- Nose Gear Wheels 32-8
- Nose Gear Tires 32-8
- Tire Interchangeability 32-8
- Shuttle Valves 32-8A
- MLG Door Pressure Reducer 32-8A
- Restrictors 32-8A
- Check Valves 32-8A
- Swivel Joints 32-8A

OPERATIONAL CHECKOUT 32-8A
- Landing Gear and Doors 32-8A

TROUBLESHOOTING 32-9
- Landing Gear Control System 32-9

REMOVAL AND REPLACEMENT 32-11
- Removal of the Nose Landing Gear Assembly 32-11
- Replacement of the Nose Landing Gear Assembly 32-11
- Removal of the Main Gear Shock Strut Assembly 32-12
- Replacement of Main Gear Shock Strut Assembly 32-12
- Removal of the Main Gear Actuating Cylinders 32-12
- Replacement of the Main Gear Actuating Cylinders 32-12
- Removal of the Main Gear Uplock Assemblies 32-12
- Replacement of the Main Gear Uplock Assemblies 32-12
- Removal of the Main Gear Landing Downlock Switches 32-13
- Replacement of the Main Gear Landing Downlock Switches 32-13
- Nose Wheel Removal 32-13

Contents | Page

Nose Wheel Replacement 32-13
Main Wheel Removal 32-13
Main Wheel Replacement 32-13
Nose and Main Wheel Tire Removal and Replacement 32-13
Installation of Nose and Main Landing Gear Honeycomb Doors 32-14
ADJUSTMENT 32-14
  Main Gear Door Rigging Instructions 32-14
  Nose Gear Door Rigging Instructions 32-14
  Landing Gear and Door Switch Adjustment 32-14
  Main Gear Uplock Cylinder Linkage Adjustment (Aircraft 5001 thru 5055) 32-14
  Main Gear Uplock Cylinder Linkage Adjustment (Aircraft 5056 and Up) 32-14
  Main Gear Downlock Actuator Adjustment 32-14
MINOR REPAIR, PRESERVATION AND DEPRESERVATION, CLEANING, AND INSPECTION 32-14
  Strut Packing 32-14
  Preservation of the Landing Gear 32-14A
  Main and Nose Gear Strut Preservation 32-14A
  Main and Nose Gear Tire Preservation 32-15
  Main and Nose Gear Strut and Tire Depreservation 32-15
  Nose Wheel Disassembly 32-15
  Nose Wheel Assembly 32-15
  Main Wheel Disassembly 32-16
  Main Wheel Assembly 32-16
  Tire Maintenance, Wear, and Inspection 32-17
  Cleaning Wheel Components 32-17
  Inspecting Wheel Halves 32-17
  Retouching Wheels 32-17
  Auxiliary Landing Light Clearance Check 32-18
  Landing Gear Bushing Wear 32-18
  Nose Wheel and Tire Balancing 32-18
  Main and Nose Gear Cylinder Inspection Methods 32-18
  Inspection of Wiring in Area of Trunnion Joint 32-18
  Methods for Crack Detection in Landing Gear Wheels 32-18
  Corrosion Repair and/or Prevention Under Steering Collar Sleeves 32-19

FIGURES

32-1. Main Wheel Assembly 32-20
32-2. Nose Wheel Assembly 32-21
32-3. Removal of the Nose Gear Shock Strut, Drag Link, and Actuating Cylinder 32-22
32-4. Removal of the Main Gear Strut Assembly 32-24
32-5. Main Gear Downlock Switch and Adapter Replacement 32-26A
32-5A. Main Gear Downlock Actuator Adjustments 32-27
32-6. Main Gear Door Rigging Instruction 32-28
32-7. Nose Gear Door Rigging Instructions 32-32
32-8. Landing Gear and Door Switch Adjustment 32-36
32.9 Main Gear Uplock Cylinder Linkage Adjustment (Airplanes 5001 thru 5055) 32-41
32-9A. Main Gear Uplock Cylinder Linkage Adjustment (Airplanes 5056 and Up) 32-42
32-10. Nose Gear Strut Packing Instructions 32-43
32-11. Main Gear Strut Packing Instructions 32-44B
32-12. Landing Gear Bushing Wear 32-45
32-13. Nose Landing Gear Wheel and Tire Balancing Instructions 32-46
32-14. Landing Gear Control Schematic Diagram (Airplanes 5001 thru 5040) 32-47
32-15. Landing Gear Control Schematic Diagram (Airplanes 5041 thru 5055) 32-49
32-15A. Landing Gear Control Schematic Diagram (Airplanes 5056 and Up, and 5041 thru 5055 Modified by Service Bulletin 329-184) 32-50A
32-15B. Landing Gear Control Schematic Diagram (Airplanes 5001 thru 5040 Modified by Service Bulletin 329-291) 32-50C
32-15C. Landing Gear Control Schematic Diagram (Airplanes 5041 thru 5055 Modified by Service Bulletin 329-291) 32-50E
32-15D. Landing Gear Control Schematic Diagram (Airplanes 5056 thru 5162 Modified by Service Bulletin 329-291) 32-50G
32-15E. Landing Gear Control Schematic Diagram (Airplanes 5001 thru 5040 Modified by Service Bulletin 329-25B) 32-50J

32.0.1 DESCRIPTION AND OPERATION.

32.0.1.1 GENERAL. (See figures 32-14 and 32-15.) The JetStar airplanes have retractable tricycle landing gear made up of two main strut assemblies and one nose strut assembly. Each strut assembly has dual wheels equipped with tubeless tires. Each strut is actuated by a double-acting hydraulic actuating cylinder. Hydraulic power for gear actuation is normally supplied by the normal hydraulic system with the auxiliary hydraulic pump automatically supplementing normal system power during gear-up actuation. The auxiliary hydraulic pump can be manually selected to actuate the landing gear independently of the normal hydraulic system pump. In the absence of both normal and auxiliary hydraulic power, the gear can be extended by using the emergency gear release system.

a. The gear actuating cylinders extend to retract the landing gear and retract to extend the landing gear. The nose strut door and the main gear outboard doors are mechanically linked to their respective strut assemblies, and follow the movement of the strut. The nose gear forward doors and the main gear inboard doors are actuated by hydraulic cylinders.

32.0.1.2 MAIN GEAR ASSEMBLY. Each main gear strut has dual wheels equipped with full-circle, multiple-disk wheel brakes. A trunnion at the top of each strut is connected to the wing structure by two self-aligning bearing assemblies. One end of each main gear actuating cylinder attaches to its strut, and the other end attaches to a fitting which is attached to the wing structure. Extending the main gear actuators pivots the gear inboard and up.

a. Main gear uplocks are spring loaded to the locked position, but are hydraulically actuated to the unlocked position. Main gear downlocks are internal locking devices inside the main gear actuating cylinders.

b. Main gear struts are combination air-oil shock absorbing and load carrying struts. Internal metering pins control shock absorbing and damping characteristics according to the loading factors.

32.0.1.3 MAIN GEAR DOORS. Each main gear has four doors: three outboard doors, and one inboard door. The outboard doors are mechanically linked to the strut assemblies and follow strut movement. The inboard doors are actuated by hydraulic cylinders, and are sequenced to gear actuation by the landing gear control circuits. See figure 52-11 and 52-13 for detail component and identification figures.

WARNING

When working around open main gear inboard doors, make sure that the ground safety locks are installed. Application of power to the gear control circuit with these doors open could actuate the doors and injure personnel critically.

32.0.1.4 NOSE GEAR ASSEMBLY. The nose gear assembly has dual wheels. The strut is connected to the fuselage structure by two self-aligning bearing assemblies attached to the sides of the wheel well. One end of a two-piece drag link is connected to the strut, and the other end is connected to a fulcrum which in turn attaches to the airplane structure. Extending the nose gear actuating cylinder forces the two-piece drag link to fold and retracts the nose strut assembly. A latching mechanism engages the nose strut assembly in the retracted position. When in the gear-down position, a latch and anvil mechanism locks the two-piece drag link in the extended position, thereby locking the nose strut assembly in the down position.

32.0.1.5 NOSE GEAR DOORS. There are three main doors on the nose gear and one small flapper door. Two forward doors are hydraulically actuated. An aft door is attached to the strut and follows strut movement. A small flapper door on the aft door is spring loaded closed, but opens to prevent steering cylinder interference with the strut door during steering operations. See figure 52-11 and 52-12 for detail component and identification figures.

32.0.1.6 NORMAL GEAR-UP ACTUATION (Airplanes 5001 thru 5040). (See figure 32-14.) Squat switches on the main gear torque arms break the up-actuating circuit to prevent gear retraction when the airplane is on the ground.

a. With the landing gear down and the airplane off the ground, placing the landing gear control handle in the "LG UP" position completes a circuit from the essential bus, through the switch contacts in the landing gear control unit, to energize the landing gear auxiliary hydraulic pump relay. This relay sends power to the auxiliary hydraulic system No. 3 relay to energize the auxiliary hydraulic pump, as long as the control handle is in the "LG UP" position and the main gear doors are not up and locked. At the same time as the preceding circuit is completed, another circuit is completed through the main gear squat switches, through the nose gear uplock switch and/or the nose gear door uplock switch, and through the left main gear uplock switch to the No. 1 solenoid of the landing gear door selector valve, to open the main gear inboard doors. When Service Bulletin 329-25B or Service Bulletin 329-291 has been incorporated, the circuit to the No. 1 solenoid of the landing gear door selector valve is through the left main gear downlock switch instead of through the uplock switch.

b. As the main gear inboard doors reach the fully open position, a circuit is completed through the right main gear door down switch, through the left main gear door down switch, and through the switch contacts in the landing gear control unit to the No. 1 solenoid of the gear selector valve, to retract the landing gear.

32.0.1.6 (Cont).

c. As the main gear reaches the fully retracted position, the left main gear uplock switch removes the power from the open solenoid of the door selector valve. At the same time, the left and right main gear uplock switches also complete a circuit through the door uplock switches to the No. 2 solenoid of the door selector valve, to close the main gear inboard doors. A circuit is also completed to the up solenoid of the gear selector valve through all of the gear door uplock switches, the nose gear uplock switch, the right main gear uplock switch and the left main gear uplock relay.

d. As the last door locks in the closed position, the circuits to the close solenoid of the door selector valve and to the up solenoid of the gear selector valve are broken.

32.0.1.7 NORMAL GEAR-UP ACTUATION (FOR AIRPLANES 5041 AND UP). (See figure 32-15.) Gear-up actuation of airplanes 5041 and up is the same as that of airplanes 5001 thru 5040 except for auxiliary hydraulic pump actuation and one portion of the gear control circuit. On airplanes 5056 and up and airplanes 5041 thru 5055 that have been modified by Service Bulletin 329-184 or Service Bulletin 329-291, the circuit to the No. 1 solenoid of the landing gear door selector valve is through the left main gear downlock switch instead of through the uplock switch. (See figure 32-15A.)

a. On airplanes 5041 and up, the auxiliary hydraulic pump control circuit energizes the auxiliary hydraulic system No. 3 relay directly, instead of going through the landing gear auxiliary hydraulic pump relay.

b. During gear-up actuation, as the main gear inboard doors reach the fully-open position, a circuit is completed through the right main gear door down switch, and through the left main gear door down switch directly to the No. 1 solenoid of the gear selector valve, instead of through the gear control unit. This circuit also energizes the landing gear relay to hold the circuit to the No. 1 solenoid of the gear selector valve while the main gear doors are closing.

32.0.1.8 NORMAL GEAR-DOWN ACTUATION (FOR AIRPLANES 5001 THRU 5040). (See figure 32-14.) With the landing gear up and locked, placing the gear control handle in the "LG DOWN" position completes a circuit from the essential bus through the switch contacts in the landing gear control unit, through either or both of the main gear uplock switches, through either or both of the main gear door down switches, back through the switch contacts in the landing gear control unit to the No. 1 solenoid of the gear selector valve, to actuate the gear upwards and thus remove main gear weight from the uplocks until the uplocks are released. As the preceding circuit is completed, a circuit is also completed through the left main gear uplock switch to the No. 1 solenoid of the door selector valve, to open the main gear inboard doors.

a. As both main gear inboard doors reach the fully-open position, the door down switches break the circuit to the up solenoid of the gear selector valve, and at the same time make a circuit to the No. 2 solenoid of the gear selector valve, to unlock the main gear uplocks and extend the main gear.

b. As the main gear uplocks release, the left main gear uplock switch breaks the circuit to the open solenoid of the door selector valve. A spring detent in the door selector valve holds the valve in the door-down position until the close solenoid is energized.

c. As the main gear reaches the fully-down position, the main gear downlock switches complete a circuit through either or both of the main gear door uplock switches and the left main gear uplock switch to energize the close solenoid of the door selector valve and thus close the main gear inboard doors.

d. As the main gear inboard doors close fully, the main gear door uplock switches break the circuit to the close solenoid of the door selector valve. The gear selector valve remains energized in the gear-down position.

32.0.1.9 NORMAL GEAR-DOWN ACTUATION (FOR AIRPLANES 5001 THROUGH 5040 MODIFIED BY SERVICE BULLETIN 329-291). (See figure 32-15B.) With the landing gear up and locked, placing the gear control handle in the "LG DOWN" position completes a circuit from the essential bus through the switch contacts in the landing gear control unit through either or both of the main gear uplock switches, through either or both of the main gear door down switches. From there a circuit is completed through the left main gear downlock switch, back through the switch contacts in the landing gear control unit to the No. 1 solenoid of the gear selector valve, to actuate the gear upwards and thus remove main gear weight from the uplocks until the uplocks are released. As the preceding circuit is completed, a circuit is also completed through the left main gear downlock switch to the No. 1 solenoid of the door selector valve to open the main gear inboard doors.

a. As both main gear inboard doors reach the fully open position, the door down switches break the circuit to the up solenoid of the gear selector valve and at the same time make a circuit to the No. 2 solenoid of the gear selector valve to unlock the main gear uplock and extend the main gear.

b. As the main gear reaches the fully down position, the left main gear downlock switch breaks the circuit to the open solenoid of the door selector valve and completes a circuit through both main gear downlock switches, both main gear uplock switches, and either of the main gear door uplock switches. This energizes the close solenoid of the door selector valve and thus closes the main gear inboard doors.

c. As the main gear inboard doors close fully, the main gear door uplock switches break the circuit to the close solenoid of the door selector valve. The gear selector valve remains energized in the gear down position by a circuit through both main gear uplock switches and also a circuit through the right main gear downlock switch.

32.0.1.10 NORMAL GEAR-DOWN ACTUATION (FOR AIRPLANES 5001 THROUGH 5040 THAT HAVE BEEN MODIFIED BY SERVICE BULLETIN 329-25B AND AIRPLANES 5041 AND UP.) (See figure 32-15.) Normal gear-down actuation on airplanes 5041 and up is similar to that on airplanes 5001 thru 5040, except that the main gear selector valve does not receive a gear-up signal while the main gear uplocks release, and the valve has a spring detent to hold it in the gear-down position when the No. 2 solenoid is de-energized. This spring detent releases the valve only after the No. 1 solenoid is energized. On airplanes 5001 through 5040, when modified by Service Bulletin 329-25B, airplanes 5041 thru 5055 when modified by Service Bulletin 329-184 and airplanes 5056 and up the circuit to the No. 1 solenoid of the landing gear door selector valve is through the left main gear downlock switch instead of through the uplock switch. (See figure 32-15A.)

32.0.1.11 NORMAL GEAR-DOWN ACTUATION (FOR AIRPLANES 5001 THROUGH 5040 WHEN MODIFIED BY SERVICE BULLETINS 329-25B AND 329-291 AND AIRPLANES 5041 AND UP WHEN MODIFIED BY SERVICE BULLETIN 329-291.) (See figures 32-15A, 32-15B, and 32-15E.) Normal gear-down actuation is similar to that of airplanes 5001 through 5040 when modified by Service Bulletin 329-291, except that the main gear selector valve does not receive a gear-up signal while the main gear uplocks release.

32.0.1.12 LANDING GEAR CONTROL PANEL. The landing gear control panel has a landing gear control handle, three position indicating lights, a WARNING SILENCER pushbutton, and a HANDLE LIGHT TEST pushbutton. This panel is attached to the copilot's side of the main instrument panel. The control handle operates the gear control switch. The handle contains two bulbs which make the handle glow red for gear warning. A green indicator light for each gear goes on as its corresponding gear locks in the down position. Pressing the WARNING SILENCER pushbutton silences the gear warning signal. Pressing the HANDLE LIGHT TEST button completes a circuit directly to the bulbs in the control handle to test them.

32.0.1.13 LANDING GEAR SWITCHES AND RELAYS. Landing gear switches and relays change the landing gear control, warning, and indicating circuits according to component positions to properly sequence gear actuation and to control hydraulic power to the nose steering and the wheel brake systems. Electronic equipment power is sometimes routed through gear squat switches, to operate only during flight or when the gear is off the ground.

32.0.1.14 MAIN GEAR UPLOCK ASSEMBLY. Each main gear has an uplock assembly located in the roof of the wheel well box. This assembly is a latching mechanism that is spring-loaded to remain either open or closed. The latches engage a hub cap on the main gear axle to lock the gear in the fully up position. Routing hydraulic power to the uplock actuating cylinder unlocks the gear by releasing the latching mechanism.

32.0.1.15 MAIN GEAR DOWNLOCKS. The main gear downlock is part of the main gear actuating cylinder. This locking mechanism locks the actuating cylinder in the retracted position to lock the gear in the down position. The lock is spring loaded to lock in the absence of gear-up hydraulic pressure. When gear-up hydraulic pressure is applied to the cylinder, the pressure forces the lock to release the actuating cylinder.

32.0.1.16 NOSE GEAR UPLOCK ASSEMBLY. The nose gear uplock assembly is a mechanical latching device that engages the nose gear strut when it reaches the fully up position. This uplock assembly is entirely mechanical in operation, and locks the gear in the up position by partially wrapping an arm around the nose strut.

32.0.1.17 NOSE GEAR DOWNLOCK ASSEMBLY. The nose gear downlock assembly is made up of a latch and anvil mechanism which locks the two sections of the drag link in the extended position to hold the nose gear in the down-and-locked position. When hydraulic pressure is applied to the actuating cylinder for the nose gear, the actuating cylinder movement as it extends releases the downlock and allows the drag link to fold normally.

32.0.1.18 MAIN GEAR INBOARD DOOR ACTUATING CYLINDER. One double-acting hydraulic cylinder actuates each main gear inboard door. Hydraulic power to these cylinders is supplied by the normal or auxiliary hydraulic system through the sequence action of the landing gear door selector valve.

32.0.1.19 NOSE GEAR FORWARD DOOR ACTUATING CYLINDER. Both nose gear forward doors are actuated by a single actuating cylinder. This cylinder is attached to door linkages, and is mounted laterally in the front of the nose wheel well. This cylinder receives its hydraulic power from the normal or auxiliary hydraulic system through the sequencing action of the controllable check valves operated by the nose gear and nose gear forward door movements.

32.0.1.20 NOSE GEAR ACTUATING CYLINDER. A double acting hydraulic cylinder actuates the nose gear assembly. Retracting the cylinder extends the landing gear, and extending the cylinder retracts the landing gear. The cylinder actuates the gear by acting on the nose gear drag link. Hydraulic power to this cylinder is controlled by the landing gear selector valve.

32.0.1.21 MAIN GEAR ACTUATING CYLINDERS. The main gear actuating cylinders extend to retract the gear and retract to extend the gear. An internal locking device in each cylinder serves as a main gear downlock. Main gear downlock switches are positioned by these downlocks within the cylinders. Once the downlock is set and the cylinder is fully retracted, it can be released only by applying hydraulic pressure to the extension side of the cylinder.

32.0.1.22 LANDING GEAR SELECTOR VALVE. The landing gear selector valve is a four-way, three-position, two-solenoid valve. This valve controls normal hydraulic power to the main gear uplock cylinders, main gear actuating cylinders, the nose gear steering system, the nose gear actuating system, the nose gear door actuating system, and the wheel brake system, via the anti-skid valves. Energizing the No. 2 solenoid of this valve positions the valve to extend the gear and to supply hydraulic power to the wheel brake and nose steering systems. Energizing the No. 1 solenoid positions the valve to retract the gear and block hydraulic power from the wheel brake and nose gear steering systems. On airplanes 5041 and up and on all previous airplanes equipped with a valve manufactured by the Parker Aircraft Co., solenoid No. 1 is designated solenoid A and solenoid No. 2 is designated solenoid B. An internal spring detent holds the valve in the gear down position after solenoid B is de-energized. To release this detent, solenoid A must be energized.

32.0.1.23 LANDING GEAR DOOR SELECTOR VALVE. The landing gear door selector valve is a four-way three-position, two-solenoid valve. This valve controls fluid flow between the normal hydraulic system and the main gear inboard door actuating cylinders. Energizing the No. 2 solenoid positions the valve to open the doors; energizing the No. 1 solenoid positions the valve to close the doors. When this valve is positioned to open the doors, a spring detent within the valve holds the valve in the door open position after the No. 2 solenoid is deenergized. To release this detent, the No. 1 solenoid must be energized. On airplanes 5041 and up, the landing gear door selector valve is identical to the landing gear selector valve (see paragraph 32.0.1.22.)

32.0.1.24 LANDING GEAR ISOLATION VALVE. A landing gear isolation valve on airplanes 5041 and up can be manually positioned to bypass the landing gear and landing gear door selector valves.

32.0.1.25 NOSE LANDING GEAR AND LANDING GEAR DOOR SEQUENCING VALVES. Two controllable check valves sequence nose gear and nose gear door actuation. Each valve has a control arm which is attached to a spring to keep the valve in the normal position. The control arm of one valve is actuated by contact with the nose strut, and the control arm of the other valve is actuated by contact with the nose gear forward doors. When in the normal position, the valves allow fluid to flow in either direction. When actuated, the valves allow fluid flow in one direction only.

32.0.1.26 MAIN GEAR UPLOCK ACCUMULATORS. Airplanes 5041 and up have two main gear uplock accumulators, one in the hydraulic line to each main gear uplock. The normal precharge on these accumulators is 500 PSI. An air filler valve and on airplanes 5066 and up a pressure gauge facilitate accumulator servicing. One accumulator is located outboard of each main wheel well in the wing trailing edge structure.

32.0.1.27 MAIN GEAR WHEELS. (See figure 32-1.) Each main gear wheel is a split, reconvertible wheel made of forged aluminum and weighing approximately 29 pounds without the tire. These wheels are normally used with tubeless tires, but can be reconverted to use tires with tubes. The periphery of the inner-wheel-half parting surface is recessed to allow installation of an O-ring. This O-ring prevents air leakage when the wheel assembly is used with a tubeless tire. Roller bearings in each wheel half support the wheel on the axle. Steel inserts in the inboard wheel half engage and rotate the brake carrier and lining assemblies of the wheel brake assemblies. Three thermal relief plugs in the inboard wheel half melt at high temperatures to relieve excessive tire pressures caused by heat.

32.0.1.28 MAIN GEAR TIRES. Main gear tires for airplanes 5001 thru 5092, 5094 thru 5096, are 26 X 6.6, Type VII, 14 ply rating, tubeless tires. For airplanes 5093, 5097 and up tires are 26 X 6.6, Type VII, 14 ply rating, reinforced tread, tubeless tires.

32.0.1.29 NOSE GEAR WHEELS. (See figure 32-2.) Each nose gear wheel is a split, reconvertible wheel made of forged aluminum and weighing approximately 12 pounds without the tire. It is used with tubeless tires, but can be reconverted to use tires with tubes. The periphery of the inner-wheel-half parting surface is recessed to allow installation of an O-ring. This O-ring prevents air leakage when using tubeless tires. Both halves carry the same serial number. Roller bearings on each wheel half support the wheel on the axle. A mounting flange on the hub of the outer wheel half allows installation of a torque tube and splined plate to provide co-rotation between nose wheels.

32.0.1.30 NOSE GEAR TIRES. Nose gear tires for airplanes 5001 through 5092 and 5094 through 5096 are 18 X 4.4, Type VII, 10-ply rating, tubeless tires. For airplanes 5093, 5097 and up nose gear tires are 18 X 4.4, Type VII, 12-ply rating, reinforced tread tubeless tires.

32.0.1.31 TIRE INTERCHANGEABILITY. Dash 8 main gear tires (26x6.6-14PRTLT7RT) may be used interchangeably and intermixed on a 6A JetStar and tire pressures of 205 psi (6A) or 220 psi (Dash 8) are optional as long as all four main gear tire pressures are the same. 6A main gear tire (26x6.6-14PRTLT7) pressures may be raised to 220 psi (Dash 8 pressures) if an operator desires.

32.0.1.31 (Cont).

It is important that tire pressures never be reduced below the specified figures as tire sidewall integrity is jeopardized.

Dash 8 nose gear tires (18x4.4-12PRTLT7RT chine) may be used on the 6A JetStar but they should not be mixed with the 6A tire (18x4.4-10PRTLT7 chine) on the same gear because of the different ply-rating. Nose tire pressure on the 6A is optional as long as both tire pressures are the same. 180 psi (6A) may be used in Dash 8 nose tires on a 6A JetStar, or the standard Dash 8 pressure of 220 psi may be used.

If an operator owned a JetStar of each type (6A and Dash 8), he could standardize using Dash 8 tires and Dash 8 tire pressures, thus eliminating the possibility of improper tire installation and the use of improper tire pressures.

Lockheed does not recommend use of recapped tires on the JetStar. Experience has shown that those operators that have used recapped tires have had extensive damage to wheel well doors, trailing edge flaps and other structures.

32.0.1.32 SHUTTLE VALVES. Three shuttle valves in the landing gear system permit utilization of the emergency release air bottles: one shuttle valve for the nose landing gear actuator and one each for the main landing gear actuators.

32.0.1.33 MLG DOOR PRESSURE REDUCER. On airplanes 5041 and up, a pressure reducing valve is installed in the main landing gear inboard door operating system so as to reduce door open operating pressure (1500 (±50) PSI). The reduced pressure is sufficient to overcome air loads on the door assembly but does not produce excessive loads in the door hinge backup structure.

32.0.1.34 RESTRICTORS. Several restrictor valve assemblies are installed in the landing gear and landing gear door operating systems to control or meter hydraulic flow.

a. A two-way restrictor is installed on the head side of each main landing gear door actuator to control the speed of operation of the doors.

b. A one-way restrictor is installed on the head side of each main landing gear actuating cylinder to regulate the landing gear down operating speed.

c. A one-way restrictor is installed on the rod side of each main landing gear actuating cylinder to regulate the landing gear up operating speed. On airplanes 5041 and up, a two-way restrictor is used in lieu of the one-way restrictor; the location, downstream of the pressure source for the uplock assembly, and upstream of the pressure source for the accumulator, ensures that the uplock is pressurized before the landing gear actuator for landing gear down operation.

d. A one-way restrictor is installed in the plumbing line to the head side of the nose landing gear door actuator and to the rod side of the nose landing gear actuator. This restrictor controls the closing speed of the nose landing gear doors and the retraction speed of the nose landing gear.

e. A two-way restrictor is installed in the plumbing line to the rod side of the nose landing gear door actuator and to the head side of the nose landing gear actuator. This restrictor controls the opening speed of the nose landing gear doors and the extension speed of the nose landing gear, and also protects against overpressurization of the nose landing gear actuating cylinder.

32.0.1.35 CHECK VALVES. Several check valve assemblies are installed in landing gear and landing gear door plumbing, positioned to prevent undesirable hydraulic fluid flow.

a. On airplanes 5041 and up, a check valve is installed in the plumbing associated with the main landing gear door pressure reducer. It is used to control a bypass line around the pressure reducer so that full hydraulic system pressure is utilized to close the inboard doors.

b. Check valves are located in the return lines downstream of the main landing gear selector valve and downstream of the main landing gear door selector valve. These check valves prevent inadvertent unlocking of the gear uplocks or unlatching of the doors by transient surges in the normal system (No. 1) return line.

32.0.1.36 SWIVEL JOINTS. Two swivel assemblies are used in the nose landing gear door actuator plumbing. These swivels permit the position of the nose landing gear door actuator to translate without damaging the plumbing.

32.0.2 OPERATIONAL CHECKOUT.

32.0.2.1 LANDING GEAR AND DOORS.

a. Jack the airplane according to the instructions in Chapter 7 using wing and fuselage jacks. Wheels must clear the ground.

b. Remove the landing gear ground safety locks, and make sure that the following circuit breakers are engaged:

Landing Gear Position Indicator
Landing Gear Down and Locked Indicator
Landing Gear Control
Hydraulic Pump Control
Hydraulic Pressure, System No. 1
Warning Lights
Main Inverter Control
Inverter Out Lights
Landing Gear Warning

c. Connect 28-volt, DC power to the airplane.

32.0.2.1 (Cont).

d. Connect external hydraulic power to the normal hydraulic system external hydraulic connections. Adjust the power source to deliver no flow at 3,000 PSI, and 8 gallons per minute at 2,700 PSI.

Note

If there is any indication of air in the system during this test, discontinue the test, bleed the system, and start the test again. Allow 30 seconds between each gear actuation.

e. Position all throttles in the fully forward position.

f. Place the landing gear control handle in the "LG UP" position and observe the following: the red warning lights in the gear control handle, the green indicator lights on the landing gear control panel, all gear and door actuation, the auxiliary hydraulic pump, and the time required for complete gear-up actuation from start to finish.
RESULT: The auxiliary hydraulic pump should energize and remain energized from the time that either main gear inboard door unlocks until both inboard doors are closed and locked. The landing gear control handle should glow from the time the control handle is placed in the "LG UP" position, until the gear is up and locked. Each green indicator light should go out as soon as its respective gear leaves the down and locked position. The total time required for actuation from the fully-down position to the fully-up position should not exceed 11 seconds.

g. Retard the throttles to below minimum cruise one at a time, and then return them one at a time to the fully-forward position. When a warning tone is heard, press the WARNING SILENCER button on the gear control panel, and then disengage and engage the landing gear control circuit breaker. Make sure all throttles are returned to the fully-forward position upon completing this step.
RESULT: The warning lights in the landing gear control handle should go on each time a throttle is retarded below minimum cruise, and a warning tone should be heard. When the WARNING SILENCER button is pressed, the warning tone should cease. When the landing gear control circuit breaker is disengaged and then engaged, the warning tone should start again. Returning the throttles to the fully-forward position should silence the warning and extinguish the warning lights in the control handle.

Note

Airplanes 5001 through 5045 are equipped with plunger type squat switches, while airplanes 5046 and up are fitted with rotary squat switches. The plunger type switch can be held in the "on the ground" position by the use of a simple wedge, but in the case of the rotary switch, it is preferable to remove the torque arm pivot pin (by performing Step 1 in Fig. 32-11) so that the switch arms complete with attendant linkage may be moved into the "on the ground" position, without disturbing the linkage settings. Tying the switch arms in the "on the ground" position with cord or wire can then be carried out.

h. Place the landing gear control handle in the "LG DOWN" position, and observe the following: the red warning lights in the gear control handle, the green indicator lights on the gear control panel, the gear and gear door operation, and the amount of time required for complete gear-down actuation.
RESULT: The gear control handle should glow red as soon as the gear control handle is placed in the "LG DOWN" position, and should remain on until the gear is down and locked. The green indicator lights should go on as soon as their respective landing gears reach the down and locked positions. The time required for the gear to actuate from the up and locked position to the down and locked position should not exceed 15 seconds.

i. Hold the main gear squat switches in the "on the ground" position, and then place the gear control handle in the "LG UP" position.
RESULT: There should be no effect on the landing gear actuating system.

j. Return the gear control handle to the "LG DOWN" position, and return the squat switch to the "in the air" position.

k. Operate the right main gear squat switch to the "on the ground" position, and then place the gear control handle in the "LG UP" position.
RESULT: There should be no effect on the landing gear actuating system.

l. Return the gear control handle to the "LG DOWN" position, return the right main gear squat switch to the "in the air" position, and reinstall the landing gear ground safety locks.

m. Disengage the following circuit breakers:

Landing Gear Control.
LG Down and Lock Ind.
LG Position Ind.

Remove the landing gear control panel, and disconnect the connector. Check continuity between pin V of the plug and aircraft ground.
RESULT: The meter should read continuity.

n. Connect the meter between pins V and W on the receptacle on the back of the control panel.
RESULT: Meter show an open circuit with the handle in the "LG DOWN" position and continuity with the handle in the "LG UP" position.

32.0.2.1 (Cont).

o. Replace the connector and the landing gear control panel. Place the handle in the "LG DOWN" position, and engage the LG Control, LG Down and Lock Ind., and LG Position Ind. circuit breakers.

p. Lower the airplane to the ground in accordance with the instructions in Chapter 7, and return the landing gear control system to normal.

32.0.3 TROUBLESHOOTING.

32.0.3.1 LANDING GEAR CONTROL SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
GEAR BEGINS TO RETRACT WHEN THE GEAR CONTROL HANDLE IS PLACED IN THE "LG UP" POSITION, EVEN THOUGH THE MAIN GEAR SQUAT SWITCHES ARE IN THE "ON-THE-GROUND" POSITION.		
Defective or improperly adjusted squat switches.	Check the operation and the adjustment of the switches.	Replace or adjust the switches as needed.
EXCESSIVE TIME REQUIRED FOR GEAR ACTUATION.		
Insufficient hydraulic power available.	Check the hydraulic power source.	Restore sufficient power.
Defective gear or gear door actuating cylinders.	Check operation of the actuating cylinders.	Repair or replace as needed.
Defective main gear door or main gear selector valves.	Check operation of the main gear door and the main gear selector valves.	Restore operation of valves to normal.
Defective or improperly adjusted controllable check valves in nose gear actuating system.	Check the operation and adjustment of the controllable check valves.	Adjust or replace the valves as needed.
Defective or improperly adjusted gear uplocks.	Check the operation and the adjustment of the gear uplocks.	Replace or adjust the uplocks as needed.
AUXILIARY HYDRAULIC PUMP FAILS TO ENERGIZE DURING GEAR-UP ACTUATION.		
Disengaged or defective landing gear control circuit breaker or hydraulic pump control circuit breaker.	Check that the circuit breakers are engaged and serviceable.	Engage disengaged circuit breakers; replace defective circuit breakers.
Disengaged or defective hydraulic pump circuit breaker.	Check that the circuit breaker is engaged and serviceable.	Engage or replace circuit breaker as needed.
Defective auxiliary hydraulic system No. 3 relay.	Check the operation of the relay.	Replace the defective relay as needed.
Defective landing gear auxiliary hydraulic pump relay (airplanes 5001 thru 5040 only).	Check the operation of the relay.	Replace the defective relay as needed.
Defective auxiliary hydraulic pump assembly.	Check the operation of the auxiliary pump assembly by trying manual control using the AUX HYD SYS-3 switch.	Replace the pump assembly as needed.

32.0.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
AUXILIARY HYDRAULIC PUMP FAILS TO ENERGIZE DURING GEAR-UP ACTUATION (Cont).		
Defective landing gear control unit.	Check the wiring and operation of the landing gear control unit.	Repair or replace the unit as needed.
Defective or improperly adjusted main gear door uplock switches.	Check the operation and adjustment of the switches.	Replace or adjust switches as needed.
MAIN GEAR INBOARD DOORS FAIL TO OPEN WHEN THE LANDING GEAR CONTROL HANDLE IS PLACED IN THE "LG UP" POSITION WITH THE MAIN GEAR SQUAT SWITCHES IN THE "IN-THE-AIR" POSITION.		
Defective or improperly adjusted left main gear uplock switch. (5001-5055)	Check the operation and the adjustment of the switch.	Replace or adjust the switch as needed.
Defective or improperly adjusted left main gear downlock switch. (5056 and up and 5001-5055 when modified by Service Bulletin 329-291)	Check the operation and the adjustment of the switch.	Replace or adjust the switch as needed.
Defective main gear door selector valve.	Check the operation of the valve.	Repair the circuit or replace the valve as needed.
Defective or improperly adjusted squat switches.	Check the operation and adjustment of the switches.	Replace or adjust switches as needed.
Defective or improperly adjusted nose gear uplock switch or nose gear door uplock switch.	Check the operation and adjustment of the switches.	Replace or adjust switches as needed.
DURING GEAR-UP ACTUATION, MAIN GEAR INBOARD DOORS OPEN, BUT GEAR FAILS TO RETRACT.		
Defective or improperly adjusted main gear inboard door down switches.	Check the operation and the adjustment of the switches.	Replace or adjust the switches as needed.
Defective main gear selector valve.	Check the operation of the valve.	Replace the valve or repair the circuit as needed.
DURING GEAR-UP ACTUATION, THE GEAR RETRACTS, BUT THE INBOARD DOORS FAIL TO CLOSE AND LOCK.		
Defective or improperly adjusted gear door or gear uplock switches.	Check the operation and the adjustment of the switches.	Replace or adjust the switches as needed.
Defective main gear door selector-valve.	Check the operation of the valve.	Repair the circuit or replace the valve as needed.
DURING GEAR-UP ACTUATION, GEAR RETRACTS AND DOORS CLOSE, BUT THE SYSTEM REMAINS ENERGIZED.		
Defective or improperly adjusted main gear inboard door down switches.	Check the operation and the adjustment of the switches.	Replace or adjust the switches as needed.

32.0.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
DURING GEAR-UP ACTUATION, GEAR RETRACTS AND DOORS CLOSE, BUT THE SYSTEM REMAINS ENERGIZED (Cont).		
Defective or improperly adjusted door uplock switches.	Check the operation and the adjustment of all gear door uplock switches.	Replace or adjust the switches as needed.
Defective or improperly adjusted nose gear uplock switch.	Check the operation and adjustment of the switch.	Replace or adjust switch as needed.

32.0.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
DURING GEAR-UP ACTUATION,GEAR RETRACTS AND DOORS CLOSE, BUT THE SYSTEM REMAINS ENERGIZED (Cont).		
Defective or improperly adjusted nose gear door uplock switch.	Check the operation and adjustment of the switch.	Replace or adjust switch as needed.
MAIN GEAR DOORS DO NOT OPEN WHEN THE LANDING GEAR CONTROL HANDLE IS PLACED IN THE "LG DOWN" POSITION (STARTING FROM THE GEAR UP POSITION).		
Defective or improperly adjusted left main gear uplock switch. (5001-5055)	Check the operation and adjustment of the switch.	Replace or adjust switch as needed.
Defective or improperly adjusted left main gear downlock switch. (5056 and up and 5001-5055 when modified by Service Bulletin 329-291)	Check the operation and adjustment of the switch.	Replace or adjust switch as needed.
Defective main gear door selector valve.	Check the operation of the valve.	Replace the valve or repair the circuit as needed.
DURING GEAR-DOWN ACTUATION, THE MAIN GEAR INBOARD DOORS OPEN, BUT THE GEAR FAILS TO EXTEND.		
Defective or improperly adjusted main gear inboard door down switches.	Check the operation and adjustment of the switches.	Replace or adjust the switches as needed.
Defective main gear selector valve.	Check the operation of the valve.	Repair the circuit or replace the valve as needed.
DURING GEAR-DOWN ACTUATION, THE MAIN GEAR EXTENDS, BUT THE MAIN GEAR INBOARD DOORS FAIL TO CLOSE.		
Defective or improperly adjusted main gear downlock switches, main gear door uplock switches and/or left main gear uplock switch.	Check the operation and the adjustment of the switches.	Replace or adjust the switches as needed.
Defective main gear door selector valve.	Check the operation of the valve.	Repair the circuit or replace the valve as needed.

32.0.4 REMOVAL AND REPLACEMENT.

32.0.4.1 REMOVAL OF THE NOSE LANDING GEAR SHOCK STRUT ASSEMBLY, DRAG LINK, AND ACTUATING CYLINDER. For these procedures, refer to figure 32-3.

32.0.4.2 REPLACEMENT OF THE NOSE LANDING GEAR SHOCK STRUT ASSEMBLY, DRAG LINK, AND ACTUATING CYLINDER. To install these components, reverse the removal procedures of figure 32-3, shimming the drag link and shock strut assemblies to center them. If after installing the drag link and shock strut assemblies they do not align properly, rearrange the shims as needed.

32.0.4.2 (Cont).

Note

Maximum shimming on either side of the shock strut is 0.086 inch; maximum shimming on either side of the drag link is 0.05 inch. Maximum side play .005-.010 after shimming.

After installing the nose gear assembly, cycle the gear to bleed the system.

32.0.4.3 REMOVAL OF THE MAIN GEAR SHOCK STRUT ASSEMBLY. For these procedures, refer to figure 32-4.

Note

The wheel and brake assemblies may be removed before removing the gear to lighten the assembly and thus make it easier to handle.

32.0.4.4 REPLACEMENT OF THE MAIN GEAR SHOCK STRUT ASSEMBLY. To install the shock strut assembly, reverse removal procedure. (Refer to Paragraph 32.0.4.3.) Check end play of installation after strut or bearing replacement as follows: (See figure 32-4.)

Note

Seals are required around the four bolts (22) on the forward side of the forward bearing cap (20). New seals are recommended with each change. The bolts (22) come in two different lengths. The two longer bolts go in the two inboard holes.

a. Check the forward trunnion bearing installation for proper clamping of the outer bearing race. There should be a gap of 0.001 inch minimum between the outer bearing race support fitting and the forward bearing cap assembly (20) prior to installation of bolts (22).

b. If no gap exists, shim the forward bearing outer race using shims P/N JL301-4 or shims fabricated from 0.020 to 0.025 inch thick stainless steel stock (annealed stock is satisfactory). The shims should have an inside diameter of 3.5 inches and an outside diameter of 3.9 inches. Install the shims as needed between shoulder on the bearing retainer hole and the forward face of the outer race of the forward trunnion bearing.

c. After shimming, check the strut fore and aft play by fastening a dial gage to the landing gear brake line bracket at the top of the strut and pull the strut by hand as far aft as possible.

d. With the strut in the position of step c, adjust the dial gage against the flat face of the forward bearing cap assembly (20).

e. Move the strut as far forward as possible by hand and record the gage reading. End play as shown by the gage should not exceed 0.020 inch.

f. Replace the forward trunnion bearing if the end play exceeds the limit of step e. Pre-fit the new trunnion bearing in the forward bearing cap assembly per steps a and b before installing the strut. Recheck per steps c, d and e.

Note

After the anti-skid and squat switch wiring has been reconnected, inspect in accordance with instructions outlined in this chapter. (Refer to Paragraph 32.0.6.18.)

Note

After gear replacement, it may be necessary to adjust the uplock to compensate for differences in gear length.

32.0.4.5 REMOVAL OF THE MAIN GEAR ACTUATING CYLINDERS.

WARNING

Depressurize the normal and standby hydraulic systems, and de-energize the auxiliary hydraulic pump circuits. Make sure that the landing gear ground locks are installed.

a. Jack the airplane clear of the ground using wing and fuselage jacks.

b. Remove the access panel on top of the wing and over the strut, and disconnect the outboard doors from the actuating cylinder being removed.

c. Disconnect the hydraulic hose from the actuating cylinder. Plug or cap all openings.

d. Disconnect the main gear downlock switch wiring from the terminal strip over the strut.

e. Disconnect the actuating cylinder from the strut by removing the retaining nut and bolt and sliding the pin forward.

f. Remove the access panel aft of the cylinder-to-wing attach point.

g. Remove the angle attached to the wing structure aft of the cylinder-to-wing attach pin.

h. Remove the cotter pin from the retaining pin in the cylinder-to-wing attach pin, and slide the retaining pin up enough to clear the locking hole in the fitting.

i. Remove the cylinder by pulling the cylinder-to wing attach pin aft. A tool to pull the pin can be made as follows: Obtain a pipe approximately 18 inches long with a diameter small enough to fit inside the attach pin. Notch the end of the pipe to fit over the retaining pin, and make cutouts near the bottoms of the notches so that the tool will hook the retaining pin when it is rotated and then moved aft. Bend the notched end of the pipe to

32.0.4.5 (Cont).

fit into the working area while allowing the maximum working area for pulling the pin. Be sure the notches are at the correct angle to the handle.

32.0.4.6 REPLACEMENT OF THE MAIN GEAR ACTUATING CYLINDERS.

a. Reverse the removal procedures.

b. Secure connections as required, bleed the landing gear system, and then check gear and door operation.

32.0.4.7 REMOVAL OF THE MAIN GEAR UPLOCK ASSEMBLIES.

a. Disconnect the emergency gear release cable from the uplock assembly.

b. Disconnect the outboard uplock claw from the unlock cylinder.

c. Disconnect the uplock switch from the uplock assembly.

d. Remove the bolts and nuts holding the uplock assembly to the top of the wheel well, and lower the uplock assembly being careful not to lose shims or serrated plates.

32.0.4.8 REPLACEMENT OF THE MAIN GEAR UPLOCK ASSEMBLIES.

a. Reverse the removal procedures, adjusting the uplock using the serrated plates for up-and-down adjustment, and shims as needed for inboard-and-outboard adjustment.

b. After installation, bleed the landing gear system, secure components as required, and then perform an operational checkout of the landing gear system.

c. Make sure that the emergency gear release system is rigged properly.

d. Adjust switch in accordance with figure 32-8.

32.0.4.9 REMOVAL OF THE MAIN LANDING GEAR DOWNLOCK SWITCHES. The main landing gear downlock switches are part of the main gear actuating cylinders, and the cylinders must be removed to remove the switches. No special instructions are required for the removal of the switches themselves.

32.0.4.10 REPLACEMENT OF THE MAIN LANDING GEAR DOWNLOCK SWITCHES. For switch and adapter replacement, refer to figure 32-5.

32.0.4.11 NOSE WHEEL REMOVAL.

a. Jack the airplane according to the instructions in Chapter 7. Wheels must clear the ground.

b. Deflate the tire and then remove the valve core.

CAUTION

Do not deflate tire by removing the valve core. With high air pressure in the tire, the valve core will be expelled with explosive force.

c. Remove the splined hub plate.

d. Remove the axle nut.

e. Install axle thread protection, and then remove the outer bearing cone.

f. Remove the wheel and tire assembly from the axle.

32.0.4.12 NOSE WHEEL REPLACEMENT.

a. Install axle thread protection.

b. With the tire mounted, install the wheel and tire assembly on the axle.

c. Pack the outer bearing cone with grease and install it in the outer-wheel half. Use grease as specified in Chapter 12.

d. Remove the axle thread protection, and then install the axle nut and tighten to 60 ft.-lbs. torque while rotating the wheel to properly seat the bearings.

e. Loosen nut to zero torque, and while rotating the wheel, retighten the axle nut to 2 ft.-lbs. Then check for alignment of locking holes. If the locking holes are not aligned, continue tightening to the next locking hole. Do not exceed 25 ft.-lbs. of torque. Safety the axle nut.

f. Install the splined hub plate.

g. Service the tire according to the instructions in Chapter 12.

32.0.4.13 MAIN WHEEL REMOVAL.

a. Jack the airplane according to the instructions in Chapter 7. Wheels must clear the ground.

b. Deflate the tire and then remove the valve core.

CAUTION

Do not deflate tire by removing the valve core. With high air pressure in the tire, the valve core will be expelled with explosive force.

c. Remove the skid detector drive.

d. Remove the axle nut.

e. Remove the outer bearing cone, install axle thread protection, and remove the tire and wheel assembly from the axle.

32.0.4.14 MAIN WHEEL REPLACEMENT.

a. Install axle thread protection.

b. With the tire mounted on the wheel, and with the bearing spacer properly positioned on the axle, install the tire and wheel assembly, making sure that the lugs on the rotating brake disks are positioned in the keyway inserts of the wheel assembly. Visually check, with the aid of a flashlight, that all liner tangs are in the wheel assembly keyways.

CAUTION

It is extremely important that the brake liner tangs be in the wheel assembly keyways or the brakes will not function properly.

c. Pack the outer wheel bearing cone with grease, and install it in the outer wheel half. Use grease as specified in Chapter 12.

d. Remove the axle thread protection, and then install the axle nut and tighten to 100 ft.-lbs. torque, while rotating the wheel to properly seat the bearings.

e. Loosen nut to zero torque, and while rotating the wheel, retighten the axle nut to 25 ft.-lbs. Check for alignment of the locking holes. If the locking holes are not aligned, continue tightening to the next locking hole. Do not exceed 60 ft.-lbs. of torque. Safety the axle nut.

f. Service the tire according to the instructions in Chapter 12 and then replace the valve cap.

32.0.4.15 NOSE AND MAIN WHEEL TIRE REMOVAL AND REPLACEMENT. The wheels must be partially disassembled to remove the tires, therefore refer to

32.0.4.15 (Cont).

the applicable wheel disassembly instructions for tire removal instructions. For replacing the tires, refer to the wheel assembly instructions. See B.F. Goodrich, Engineered Systems Division manuals, listed on page Introduction-5, for more detailed instructions.

WARNING

Failure to completely deflate tires before loosening tie bolts could allow the wheel halves to separate with explosive force, thereby injuring personnel and damaging equipment. To make sure that tires are deflated, remove the valve cores from the valve stems. If valve stem extensions are used, make sure that the extension valve core and the valve stem valve core are both removed.

32.0.4.16 INSTALLATION OF NOSE AND MAIN LANDING GEAR HONEYCOMB DOORS. After final trim and fit, rivets should be installed using a pneumatic squeezer at approximately 1.00-inch center-to-center spacing; countersink both sides with manufactured head outside. The rivet type, size, and edge distance (ED) are as shown for each door.

a. MLG Outboard Wheel Cover Door Assembly, JL270.
NAS 1097AD4 rivets at 0.28 ED along the forward and aft edges of the door.

b. MLG Outboard Strut Cover Door Assembly, JL271.
NAS 1097AD4 rivets at 0.28 ED along the forward, aft and outboard edges of the door.

c. MLG Inboard Door Assembly, JL371.
NAS 1097AD4 rivets at 0.28 ED along the forward, aft, inboard and outboard edges of the door.

d. NLG Strut Door, JL426.
MS 20426AD3 rivets at 0.25 ED around the outer edges of the door. On this door these rivets are usually installed during fabrication of door. When this is the case, additional rivets are not required.

32.0.5 ADJUSTMENT.

32.0.5.1 MAIN GEAR DOOR RIGGING INSTRUCTIONS. For these instructions, refer to figure 32-6.

32.0.5.2 NOSE GEAR DOOR RIGGING INSTRUCTIONS. For these instructions, refer to figure 32-7.

32.0.5.3 LANDING GEAR AND DOOR SWITCH ADJUSTMENT. For these instructions, refer to figure 32-8.

32.0.5.4 MAIN GEAR UPLOCK CYLINDER LINKAGE ADJUSTMENT (AIRCRAFT 5001 THRU 5055). (See figure 32-9 for component identification.) If the slots (6) in the links (5) do not bottom against the screw (7) when the uplock and uplock actuating cylinder are in the locked positions, adjust the linkage according to these instructions.

a. Disconnect the adapter (4) from the links (5).

b. Straighten the tabs of the tabwasher (2), and loosen the jamnut (3).

c. Remove the adapter (4) and replace the tabwasher (2) with a new one.

d. Reinstall the adapter (4) so that when the links (5) are connected, the inboard ends of the slots (6) will bottom against the screw (7). Both the claw and the actuation rod should be in the locked position for this step.

f. Bend one tab of the new washer (2) against the flat of the actuator rod (1), and bend another tab against one of the flats of the nut (3).

g. Make sure that all components are safetied as required, and then perform an operational checkout of the landing gear system.

32.0.5.5 MAIN GEAR UPLOCK CYLINDER LINKAGE ADJUSTMENT (AIRCRAFT 5056 AND UP.) (See figure 32-9A for component identification.) If the link assembly (6) is not offset 0.20 (±0.14) inch in center when the uplock and uplock actuating cylinder are in the unlocked position. adjust the linkage according to these instructions:

a. Disconnect the link assembly (6) from the adapter (4).

b. Straighten the tabs of the tabwasher (2), and loosen jamnut (3).

c. Remove adapter (4), and replace tabwasher (2).

d. Reinstall adapter (4), and adjust to make 0.20 (±0.14) inch offset as shown in figure 32-9A.

e. Bend one tab of the washer (2) against the flat on actuator rod (1) and bend one tab against flat on nut (3).

32.0.5.6 MAIN GEAR DOWNLOCK ACTUATOR ADJUSTMENT. (See figure 32-5A.)

32.0.6 MINOR REPAIR, PRESERVATION AND DEPRESERVATION, CLEANING, AND INSPECTION.

32.0.6.1 STRUT PACKING AND SERVICING. For nose strut packing, refer to figure 32-10. For main strut packing, refer to figure 32-11. See Chapter 12 for servicing.

32.0.6.3 (Cont)

Tools Required For Packing Strut

Nomenclature	Description	Use And Application
Wrench	11035-WR (Lockheed)	To loosen and tighten the main gear gland nut.
Wrench	11044-WR (Lockheed)	To loosen and tighten the nose steering collar nut.
Wrench	11045-WR (Lockheed)	To loosen and tighten the gland nut on the nose strut.

32.0.6.2 PRESERVATION OF THE LANDING GEAR DURING PERIODS OF INACTIVITY. When the airplane is inactive for prolonged periods, care must be taken to protect the landing gear components from deterioration and damage. Perform the following procedures to protect the landing gear of airplanes subjected to prolonged inactivity.

32.0.6.3 MAIN AND NOSE GEAR STRUT PRESERVATION. For periods of inactivity of not more than sixty days, maintain the proper strut inflation and wipe the strut piston thoroughly with Specification MIL-L-7870 lubricating oil or MIL-H-5606 hydraulic fluid often enough to maintain a light film of oil on the piston.

Note

The strut wiping intervals should not exceed 30 days.

For periods of inactivity of more than 60 days, follow these procedures:

a. Wipe the exposed portion of the strut pistons thoroughly with Specification MIL-L-7870 lubricating oil or MIL-H-5606 hydraulic fluid.

b. Completely deflate the struts according to the instructions in Chapter 12.

c. Remove the air valve assembly, insert a rubber tube (Specification MIL-R-6855, Class I) in the opening left by the removal of the valve until

32.0.6.3 (Cont).

it bottoms in the strut. Siphon the oil from the strut using a siphon pump. Siphon all of the oil and put half of it back in the strut.

WARNING

Be sure that the strut is completely deflated before loosening the valve assembly. If the valve is loosened with the strut pressurized, the valve could be expelled with explosive force.

d. Refill the strut with Specification MIL-H-6083 preservative oil, and then replace the valve assembly.

CAUTION

With the struts deflated, it is permissible to tow the airplane only enough to rotate the tires for prevention of flat spots.

32.0.6.4 MAIN AND NOSE GEAR TIRE PRESERVATION.

a. Apply two heavy coats of rubber preservative compound to the tires. Use Type R-8-N-C-186, Willard Tire and Rubber Company, Bell, California, or equivalent.

b. Keep the tires clean and free of all materials other than the preservative compound.

c. If the airplane is not jacked, move it enough to rotate the tires at least one-third turn every 72 hours. This prevents flat spots on the tires.

d. Keep the tires properly inflated by checking the inflation each time the tires are rotated. Add air as necessary.

Note

If the airplane is on jacks, tire pressure may be allowed to drop 45 PSI before inflation to normal pressure is required.

32.0.6.5 MAIN AND NOSE GEAR STRUT AND TIRE DEPRESERVATION. To restore the struts and tires to normal after periods of inactivity, service the struts and tires according to the instructions in Chapter 12. Preservative oil may be left in the struts when returning the airplane to service.

32.0.6.6 NOSE WHEEL DISASSEMBLY. Refer to figure 32-2 for identification of components called out in this procedure.

Note

Disassembly through step (c) is sufficient for tire removal.

a. Remove the grease seal (1) and inner bearing cone (3).

b. Make sure that the tire is fully deflated, and then remove the nuts (20), tie bolts (32), and washers (21) and (31).

WARNING

Failure to completely deflate the tire before loosening the tie bolts could allow the wheel assembly to separate with explosive force thereby injuring personnel and damaging equipment. To make sure that the tire is deflated, remove the valve core from the valve stem. If valve stem extensions are used, make sure that the extension valve core and the valve core are both removed.

c. Separate the wheel halves (24) and (28), and remove the tire and the packing (25) from the inner wheel half (28).

Note

Use a bead breaker instead of a tire iron to loosen the tire.

d. Remove the grommet (14) from the outer wheel half (24).

e. Remove the nut (13), spacer (12), grommet (11), and valve stem (8) from the outer wheel half (24).

32.0.6.7 NOSE WHEEL ASSEMBLY. Refer to figure 32-2 for identification of components called out in this procedure.

Note

Refer to paragraph 5.5.21 for wheel inspection requirements.

Note

Assembly starting at step (c) is sufficient for tire replacement.

a. Position the tubeless valve stem (8) in the outer wheel half (24).

b. Position the grommet (11) and spacer (12) on the valve stem (8).

c. Screw the nut (13) on the valve stem (8) and tighten it.

Note

Hold the valve stem firmly in place while tightening the nut.

32.0.6.7 (Cont).

d. Install the grommet (14) on the outer wheel half (24).

e. Wipe the wheel bead seat, wheel register areas, and packing groove with a cloth dampened with solvent to remove foreign matter.

f. Make sure that the tire bead area is free of foreign matter and is not damaged.

g. Place the outer wheel half (24) on a clean, flat surface, and install the tire on the outer wheel half with the tire balance mark adjacent to the valve assembly.

h. Lubricate the packing (25) with a thin coat of Specification MIL-S-8660 grease, and assemble it to the inner wheel half (28).

i. Assemble the inner wheel half (28) to the outer wheel half (24), making sure that the half-moon clearance hole in the register area of the inner wheel half is adjacent to the valve assembly.

CAUTION

Wheel halves must be assembled in the proper relative positions to prevent air leaks.

j. Lubricate the threads and bearing surfaces of the nuts (20), bolt head (32), and washers (21) and (31) with Specification MIL-T-83483 grease.

k. Compress the wheel halves together enough to allow installation of the bolts (32), washers (21) and (31), and nuts (20); and then alternately tighten them until the wheel halves seat.

CAUTION

The countersunk washer faces should be toward the heads of the bolts and nuts.

l. Alternately tighten the nuts in a criss-cross manner, and in equal increments of torque, until all nuts are adjusted to a torque of 100 to 120 inch-pounds.

CAUTION

Incorrect bolt tightening can unseat the packing, and can also cause unequal tension on the bolts, resulting in bolt failure.

m. Inflate the tire in accordance with the instructions in Chapter 12 and check for air leakage.

n. Pack the inner bearing cone with grease and assemble it to the inner wheel half. Use grease as recommended in Chapter 12.

32.0.6.8 MAIN WHEEL DISASSEMBLY. Refer to figure 32-1 for identification of components called out in this procedure.

Note

Disassembly through step (e) is sufficient for tire removal.

a. Remove valve cap (19) from tubeless tire valve assembly, deflate tire and then remove the valve core (20).

CAUTION

Do not deflate tire by removing the valve core. With high air pressure in the tire, the valve core will be expelled with explosive force.

b. Remove the retaining ring (1), retainer (2), grease seal (3), and inner roller bearing (4) from the inner wheel half (6).

c. Make sure that the tire is completely deflated, and remove nuts (23), bolts (10), washers (11) and (24) and separate wheel halves (6) and (7).

WARNING

Failure to deflate the tire completely before loosening the tie bolts could allow the wheel halves to separate with explosive force thereby injuring personnel and damaging equipment. To make sure that tires are deflated, remove the valve cores from the valve stems. If valve stem extensions are used, make sure that the extension valve core and the valve stem valve core are both removed.

d. Remove valve stem (21) and grommet (22) from inner wheel half (6).

e. Dismount tire and remove packing (13) from inner wheel half (6) and packing (14) from outer wheel half (7).

Note

Use a bead breaker instead of a tire iron to loosen the tire.

f. Remove fuse plugs (8) and packings (9) only when damaged.

32.0.6.9 MAIN WHEEL ASSEMBLY. Refer to figure 32-1 for identification of components called out in this procedure.

Note

Refer to paragraph 5.5.21 for wheel inspection requirements.

Note

Assembly starting at step (f) is sufficient for tire replacement.

a. Place the packings (9) on the relief plugs (8) and install the plugs in the inner wheel half (6). Adjust the torque on the plugs to 54 inch-pounds on airplanes indicated in Paragraph 12.6.1.1, or to 12 inch-pounds for all other airplanes covered by this manual. Plugs should be secured with safety wire.

b. Wipe the wheel bead seat, wheel register areas, and gasket groove with a cloth dampened with solvent to remove foreign matter.

c. Ensure that the tire bead area is clean and free from damage.

d. Place inner wheel half (6) on a clean flat surface and install tubeless tire on the inner wheel half with the tire balance mark adjacent to the valve stem threaded hole.

e. Lubricate the packing (13) with a thin coat of Specification MIL-S-8660 grease, and install it on the inner wheel half (6).

f. Apply a small amount of B.F. Goodrich adhesive No. A963B (or equivalent) to one side of packing (14) and attach to the register at the hub of outer wheel half (7).

g. Assemble the outer wheel half (7) onto the inner wheel half (6), making sure that the valve stem clearance hole in the outer wheel half is in alignment with the valve stem threaded hole.

h. Lubricate the threads and bearing surfaces of the tie bolts (10), washers (11) and (24) with Specification MIL-T-83483 grease.

i. Compress the wheel halves together enough to allow installation of the bolts (10), washers (11) and (24), and nuts (23); and then alternately tighten the nuts until the wheel halves seat.

CAUTION

Countersunk washer faces should face the bolt heads and nuts.

j. Alternately tighten the nuts in equal increments of torque and in a criss-cross manner until all nuts are adjusted to a torque of 800 to 850 inch-pounds.

CAUTION

Incorrect tightening of the nuts can cause the packing to unseat, and may also cause unequal tension on the bolts, resulting in failure.

k. To install tubeless valve stem assembly, place grommet (22) on the valve stem (21) and position valve stem in the inner wheel half (6). Torque the valve to 54 inch-pounds. Screw valve core (20) into the stem and tighten.

l. Inflate the tire according to the instructions in Chapter 12 and check for air leakage around the relief plugs (8), the valve stem (21), and the bead seat area.

m. Pack the inner bearing cone (4) with grease, and assemble it to the inner wheel half.

n. Assemble grease seal parts (2) and (3) in the inner wheel half (6) and secure in place with retaining ring (1). Use grease as recommended in Chapter 12.

32.0.6.10 TIRE MAINTENANCE, WEAR AND INSPECTION. Refer to Chapter 12 for general instructions for cuts and wear.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

32.0.6.11 CLEANING METAL WHEEL COMPONENTS. Clean metal wheel components in solvent, Federal Specification P-D-680, by wiping, dipping, or spraying.

32.0.6.12 INSPECTING WHEEL HALVES. (Refer to figures 32-1 and 32-2.) See Chapter 58 for wheel inspection methods.

32.0.6.13 RETOUCHING WHEELS.

a. Blend nicks, scratches, and lightly corroded areas smooth with fine emery cloth.

b. Replace the protective coating with two coats of Specification TT-P-1757 Comp. G, Color 34151, zinc chromate primer. Let the primer dry for 30 minutes between coats.

c. Finish the area with one coat of Specification TT-L-32 aluminum lacquer.

32.0.6.14 AUXILIARY LANDING LIGHT CLEARANCE CHECK. To make sure the landing light is installed with enough clearance, perform the following check:

a. Disconnect the nose gear forward doors, and open them by hand (refer to the nose gear door rigging instructions, figure 32-7).

b. Jack the airplane and retract the nose gear.

c. With the nose gear in the up-and-locked position, close the left door and check that there is 0.38-inch clearance around the auxiliary landing light.

d. Extend the landing gear, and reconnect the door linkage.

32.0.6.15 LANDING GEAR BUSHING WEAR.

a. For landing gear bushing wear tolerances, refer to figure 32-12.

b. If vertical play in nose gear steering collar assembly, JL 1605 exceeds 0.016 inch, remove JL 1614 lock and tighten JL 1613 nut assembly in one quarter turn increments until vertical play is reduced to 0.016 inch or less.

32.0.6.16 NOSE WHEEL AND TIRE BALANCING. (See figure 32-13.) When nose wheel tires are unbalanced, airframe vibrations are increased and become particularly noticeable during takeoff and landing. An increase in balance problems is anticipated with use of the chine tire because of the design and contour. Therefore, it is recommended that dynamic balancing be accomplished each time a tire is changed or whenever a tire is suspected of being unbalanced.

32.0.6.17 MAIN AND NOSE GEAR CYLINDER INSPECTION METHODS. See Chapter 58 for main and nose gear cylinder inspection methods.

32.0.6.18 INSPECTION OF WIRING IN AREA OF TRUNNION JOINT. When maintenance work, or replacements, in the main gear trunnion joint area is likely to disturb the anti-skid or squat switch wiring, make the following inspection:

a. Jack the airplane according to instructions in Chapter 7. Wheels must clear the ground.

b. Remove and retain the access doors and attaching hardware, and the upper and lower wing surfaces at approximately wing station 75, to gain access to the trunnion area of both gears.

c. Inspect the anti-skid and squat switch electrical wiring, routed up the main landing gear strut (both right-hand and left-hand) and across the trunnion joint, for frayed or broken wires.

d. Slowly retract the main landing gear, preferably with a manually operated hydraulic pump, and observe movements of the anti-skid and squat switch wire bundles for sufficient slack, and for interference with adjacent structure.

e. If no damage or interference exists and there is sufficient slack for the wire bundle to rotate without strain, steps (f) and (g) may be omitted.

f. Replace or repair all frayed or damaged wires.

g. Relocate the anti-skid and squat switch wire bundles, to provide sufficient slack and prevent abrasion of wires during normal operation of the main landing gear.

h. Reinstall access doors and attaching hardware, retained in step (b), and remove the airplane from jacks.

32.0.6.19 METHODS FOR CRACK DETECTION IN LANDING GEAR WHEELS. See Chapter 58 for landing gear wheel crack detection methods.

32.0.6.20 CORROSION REPAIR AND/OR PREVENTION UNDER STEERING COLLAR SLEEVES. If corrosion occurs under the steering collar sleeves, repair the aluminum surfaces as follows:

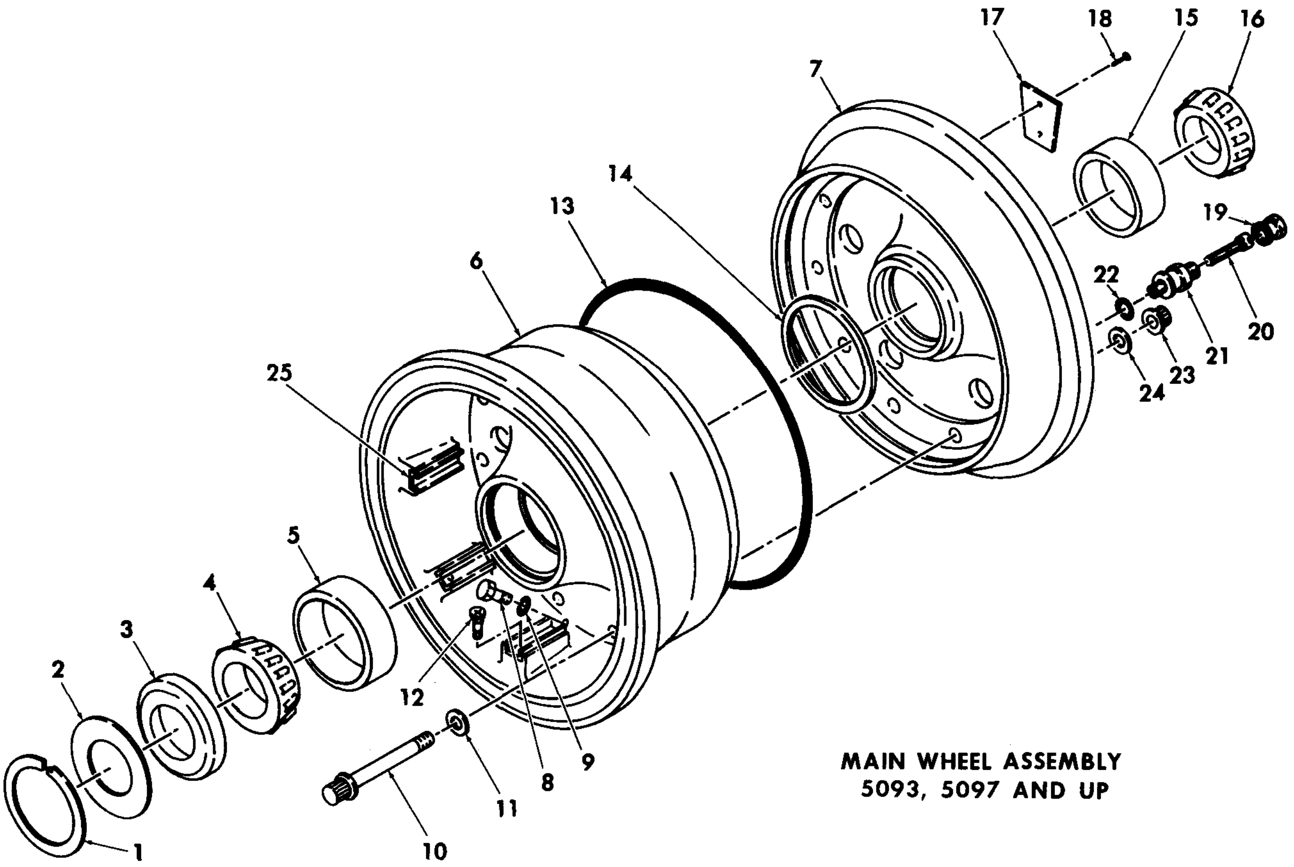
a. Mechanically clean all corrosion pits and surfaces with soft rubber bonded abrasive wheels no coarser than medium alumina (aluminum oxide). Blend the reworked areas into the surrounding surfaces.

b. Polish this reworked area with 280 grit abrasive paper, followed by 400 grit abrasive paper or a soft rubber-bonded abrasive wheel. The abrasive in the wheel shall not be coarser than fine alumina.

c. Apply Alodine 1200 conversion coating per section 51.7.

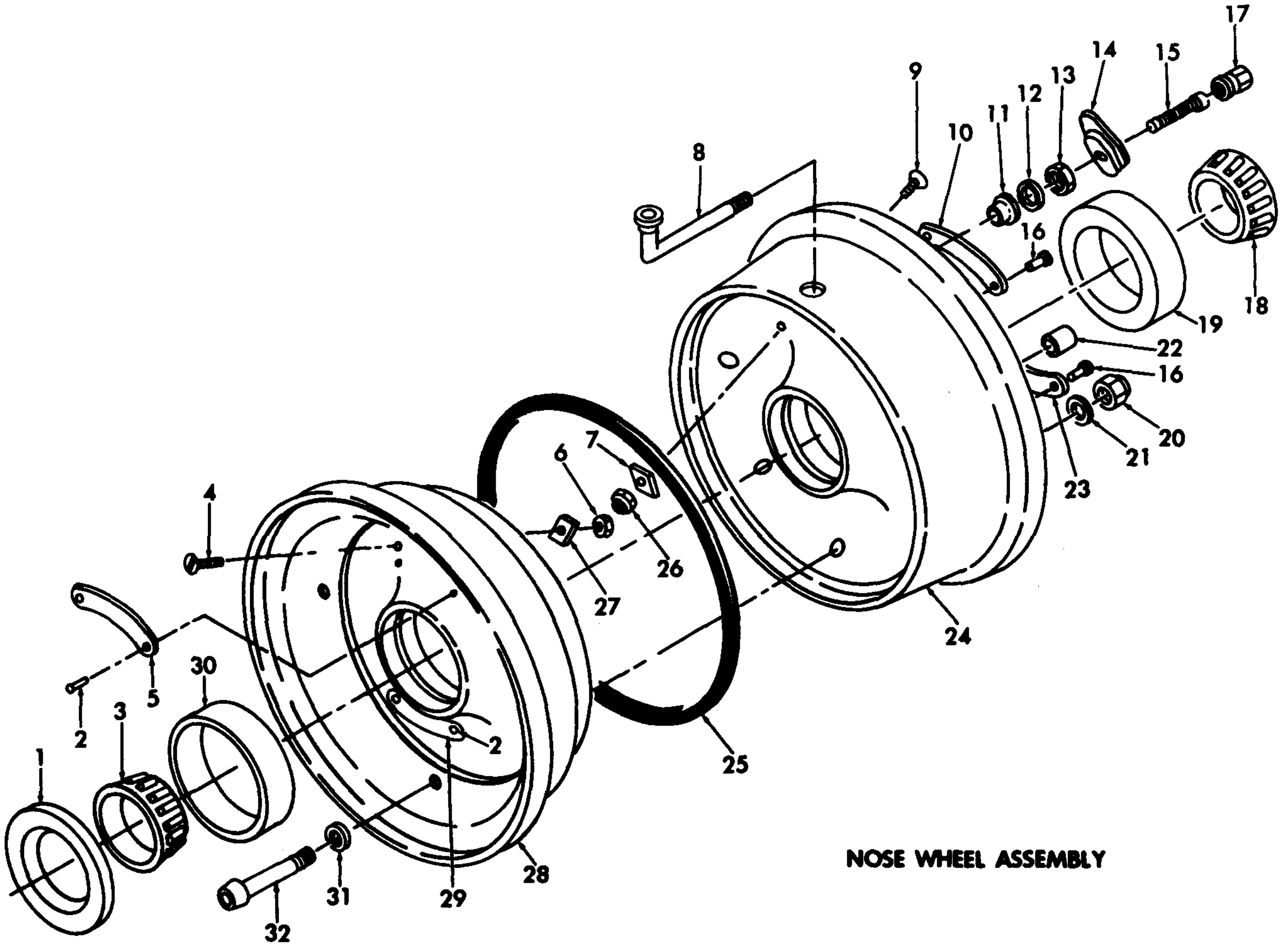
d. Apply a bead of inhibited sealing compound, MIL-S-81733, Type II, Class B1/2 in the radius of the cylinder where the top sleeve fits. MIL-S-8802, Class B1/2, sealing compound may be substituted but does not contain a corrosion preventative. (See figure 51-6.)

e. Coat the surface of the sleeves in contact with the cylinder with MIL-C-16173D, Grade 2 or Grade 3, rust preventative. (Ashland Petroleum Co., Valvoline TECTYL 502C or 894.) These corrosion preventative compounds will aid in preventing corrosion of the cylinder at the contact surfaces of the sleeves and cylinders.



MAIN WHEEL COMPONENTS LIST	
1. SNAP RING	14. PREFORMED PACKING
2. RETAINER	15. BEARING CUP
3. GREASE SEAL	16. ROLLER BEARING
4. ROLLER BEARING	17. INSTRUCTION PLATE
5. BEARING CUP	18. ALUMINUM DRIVE RIVET
6. INNER WHEEL HALF	19. VALVE CAP
7. OUTER WHEEL HALF	20. VALVE CORE
8. MACHINE THREAD FUSE PLUG	21. VALVE STEM
9. PREFORMED PACKING	22. GROMMET
10. MACHINE BOLT	23. SELF-LOCKING NUT
11. RECESSED WASHER	24. RECESSED WASHER
12. MACHINE SCREW	25. KEYWAY INSERT
13. PREFORMED PACKING	

Figure 32-1. Main Wheel Assembly



NOSE WHEEL COMPONENTS LIST

1. SEAL
2. ALUMINUM DRIVE RIVET
3. BEARING CONE
4. MACHINE SCREW
5. IDENTIFICATION PLATE
6. SELF LOCKING NUT
7. BALANCE WEIGHT
8. VALVE STEM
9. MACHINE SCREW
10. IDENTIFICATION PLATE
11. TIRE AND RIM GROMMET
12. TIRE AND RIM SPACER
13. HEX NUT
14. GROMMET
15. VALVE CORE
16. ALUMINUM DRIVE RIVET
17. VALVE CAP
18. BEARING CONE
19. BEARING CUP
20. SELF LOCKING NUT
21. COUNTERSUNK WASHER
22. BUSHING
23. INSTRUCTION PLATE
24. OUTER WHEEL HALF
25. PREFORMED PACKING
26. SELF LOCKING NUT
27. BALANCE WEIGHT
28. INNER WHEEL HALF
29. INSTRUCTION PLATE
30. BEARING CUP
31. COUNTERSUNK WASHER
32. INTERNAL WRENCHING BOLT

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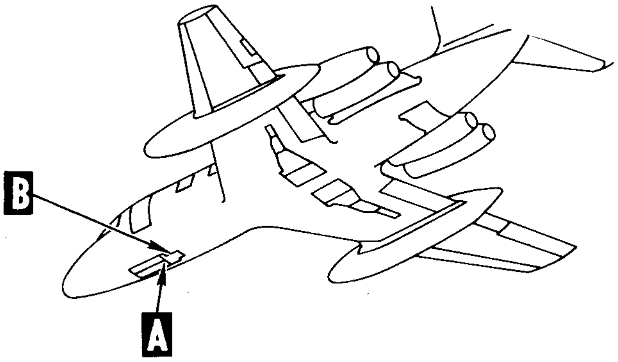
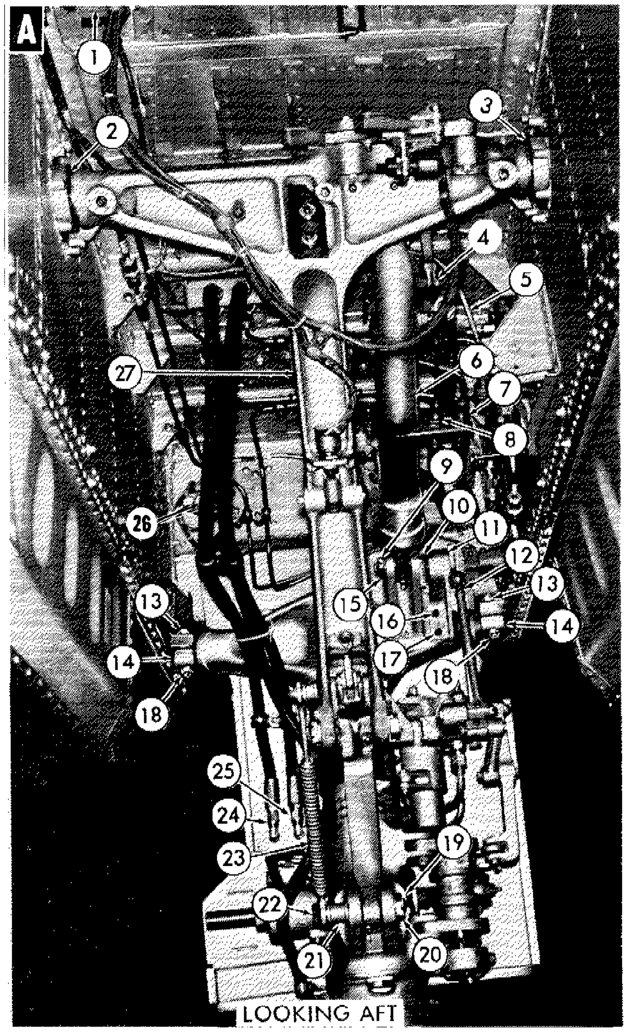
Figure 32-2. Nose Wheel Assembly



STEP 1

JACK THE AIRPLANE UNTIL THE NOSE WHEELS CLEAR THE GROUND, THEN DISCONNECT THE AFT NOSE GEAR DOOR FROM THE STRUT.



STEP 2

REMOVE THE FAIRING (NOT SHOWN) UNDER THE FULCRUM FITTINGS (14).

STEP 3

DISCONNECT THE LANDING GEAR SWITCH WIRING FROM TERMINAL STRIPS N-2 (1) AND N-4 (26).

STEP 4

DISCONNECT THE SPRING (23) THAT RUNS FROM THE DRAG LINK FOLDING AREA TO THE BOLT (22).

STEP 5

REMOVE THE COTTER PIN (20), NUT (19), BOLT (22), AND BUSHING (21), AND DISCONNECT THE DRAG LINK (27) FROM THE STRUT.

STEP 6

DISCONNECT THE HYDRAULIC HOSE AT POINTS (7), (8), (24), AND (25). PLUG THE OPEN ENDS OF THE HOSE.

STEP 7

REMOVE THE ATTACHING BOLT THAT CONNECTS THE ROD (12) TO THE STEERING ENGAGING MECHANISM, AND DISCONNECT THE ROD (12).

STEP 8

SWING THE STRUT FORWARD TO DISENGAGE THE STEERING MECHANISM, THEN REMOVE THE TWO BOLTS (16) AND (17) AND THE BRACKET (11). RETURN THE STRUT TO THE FULLY DOWN POSITION AFTER REMOVING THE BRACKET (11).

STEP 9

REMOVE THE COTTER PIN (9), NUT (15), AND BOLT (10) FROM THE ACTUATING CYLINDER (6) AT THE STRUT ATTACHING POINT.

STEP 10

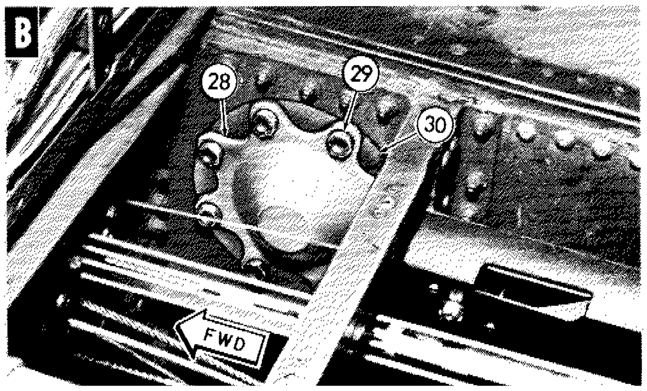
DISCONNECT THE EMERGENCY GEAR RELEASE CABLE (5) FROM THE EMERGENCY RELEASE BRACKET (4).

STEP 11

REMOVE THE FLOORING DIRECTLY ABOVE THE LEFT DRAG LINK FULCRUM FITTING (3), AND WHILE SUPPORTING THE DRAG LINK (27), REMOVE THE END CAP (28) BY REMOVING THE EIGHT BOLTS (29).

CAUTION

DO NOT LOSE SHIMS (30) INSIDE THE FUSELAGE FLOOR AREA WHEN REMOVING THE END CAP (28).



JS-1-2-X0/2-3203-1

Figure 32-3. Removal of Nose Gear Shock Strut, Drag Link, and Actuating Cylinder (Sheet 1 of 2)

STEP 12

MOVE THE DRAG LINK ASSEMBLY TO THE LEFT AND INTO THE HOLE LEFT BY THE REMOVAL OF THE END CAP (28) UNTIL THE OTHER END OF THE FULCRUM PULLS FREE OF THE FITTING (2). TWIST THE DRAG LINK SLIGHTLY, PULL IT OUT OF THE HOLE, AND LOWER IT TO THE GROUND.

STEP 13

USE AN AXLE JACK TO SUPPORT THE WEIGHT OF THE NOSE GEAR SHOCK STRUT, THEN REMOVE THE FULCRUM FITTING CAPS (14) BY REMOVING THE FOUR NUTS (18) AND BOLTS (13).

STEP 14

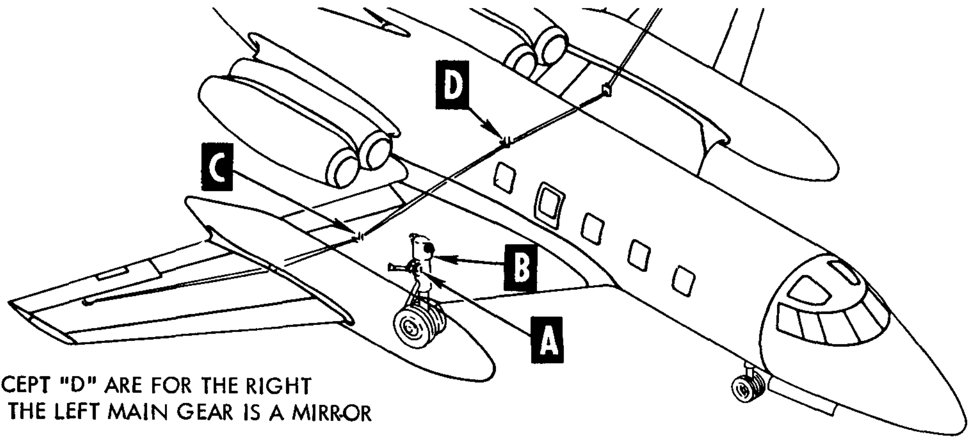
BALANCE THE STRUT ASSEMBLY ON THE AXLE JACK AND SLOWLY RELEASE THE JACK TO LOWER THE STRUT TO THE GROUND.

CAUTION

OBSERVE THE STEERING MECHANISM CLOSELY SO THAT IT WILL NOT BE DAMAGED DURING REMOVAL OF THE STRUT ASSEMBLY.

JS-2-2-3205-2

Figure 32-3. Removal of the Nose Gear Shock Strut, Drag Link, and Actuating Cylinder (Sheet 2 of 2)



NOTE

ALL VIEWS EXCEPT "D" ARE FOR THE RIGHT MAIN GEAR. THE LEFT MAIN GEAR IS A MIRROR OPPOSITE

STEP 1

CHECK THAT THE LANDING GEAR GROUND SAFETY LOCKS ARE INSTALLED.

STEP 2

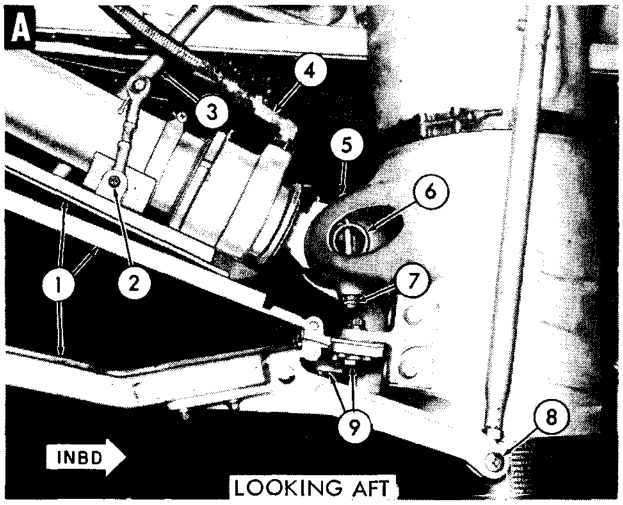
REMOVE THE ACCESS PANEL ON TOP OF WING AND OVER THE TRUNNION OF THE STRUT BEING REMOVED; AND THEN REMOVE THE ACCESS PANELS AROUND THE WING JACK PAD AREAS, INSTALL THE JACK PADS, AND JACK THE AIRPLANE UNTIL THE WHEELS CLEAR THE GROUND.

STEP 3

DISCONNECT THE MAIN GEAR OUTBOARD DOORS (1) AT POINTS (2), (8), AND (9), AND DISCONNECT ROD (3).

WARNING

IF THE MAIN GEAR INBOARD DOOR IS OPENED, INSTALL DOOR SAFETY LOCK.



STEP 4

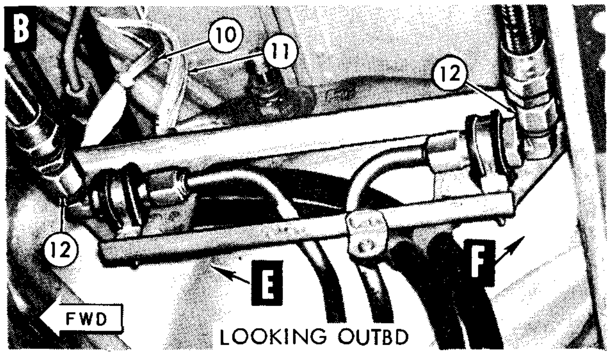
DISCONNECT AND CAP THE HOSE (4), THEN REMOVE THE BOLT (5) AND NUT (7) FROM THE STRUT END OF THE ACTUATOR, AND REMOVE THE PIN (6).

STEP 5

REMOVE THE ACCESS PANEL AT THE REAR OF THE TRUNNION.

STEP 6

DISCONNECT THE HYDRAULIC HOSE FROM THE STRUT BY REMOVING THE NUTS (12). PLUG THE OPEN LINES.



STEP 7

ON AIRPLANES 5001 THRU 5065 LOCATE A SPOT ON THE SKID-DETECTOR WIRING (11) WHERE EACH WIRE CAN BE IDENTIFIED EASILY AND WHERE THERE WILL BE ROOM ENOUGH FOR A SPLICE DURING INSTALLATION. CUT THE WIRES AT THIS SPOT. ON AIRPLANES 5066 AND UP, DISCONNECT SKID-DETECTOR WIRING AT CONNECTOR (NOT SHOWN ALTHOUGH LOCATED IN THE AREA OF PICTURE).

CAUTION

BE SURE EACH WIRE CAN BE IDENTIFIED.

STEP 8

ON AIRPLANES 5001 THRU 5065 DISCONNECT THE SQUAT SWITCH WIRING (10) FROM THE TERMINAL STRIP OVER THE STRUT TRUNNION, AND PULL THE WIRES THROUGH THE CONDUIT. ON AIRPLANES 5066 AND UP, DISCONNECT THE SQUAT SWITCH WIRING AT CONNECTOR (NOT SHOWN ALTHOUGH LOCATED IN THE AREA OF PICTURE).

JS-1-2-X0/2-3204-1

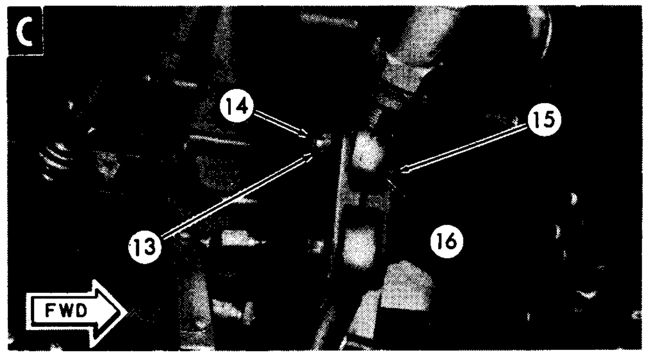
Figure 32-4. Removal of the Main Gear Strut Assembly (Sheet 1 of 2)

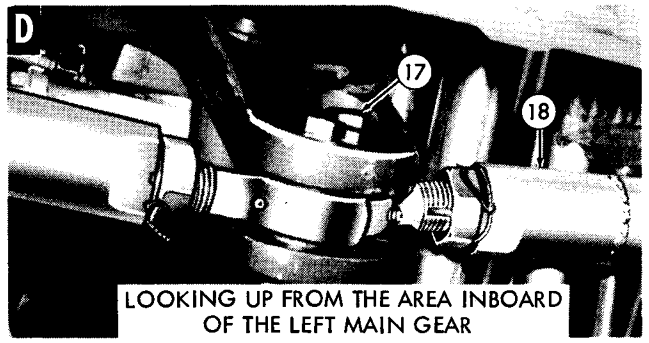
STEP 9

PLACE AN AXLE JACK UNDER THE STRUT JACK PAD, AND JACK THE STRUT UNTIL THE STRUT WEIGHT IS SUPPORTED BY THE AXLE JACK.

STEP 10

WHEN REMOVING THE RIGHT MAIN GEAR, SUPPORT THE RIGHT AILERON AND REMOVE THE COTTER PIN (14), NUT (13), WASHER (NOT VISIBLE), AND BOLT (15); AND DISCONNECT THE RIGHT AILERON PUSHROD (16) TO OBTAIN CLEARANCE BEHIND THE TRUNNION. (SEE VIEW "C".) WHEN REMOVING THE LEFT MAIN GEAR, SUPPORT THE LEFT AILERON AND REMOVE THE COTTER PIN, NUT, WASHER, AND BOLT (17); AND DISCONNECT THE LEFT AILERON PUSHROD (18) TO OBTAIN CLEARANCE. (THE NUT AND WASHER ARE NOT VISIBLE IN VIEW "D".)





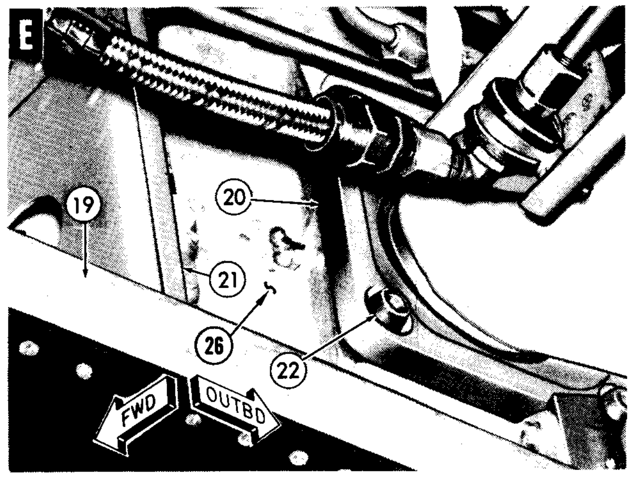
LOOKING UP FROM THE AREA INBOARD OF THE LEFT MAIN GEAR

STEP 11

REMOVE THE FORWARD ATTACH ANGLE (19), AND LOOSEN THE FOUR BOLTS (22) THAT HOLD THE FORWARD BEARING CAP (20) TO THE WING STRUCTURE (26). ON AIRPLANES 5001 - 5040 THE LOWER PORTION OF THE STIFFENER (21) CAN BE REMOVED TO PROVIDE ADDITIONAL WORKING SPACE.

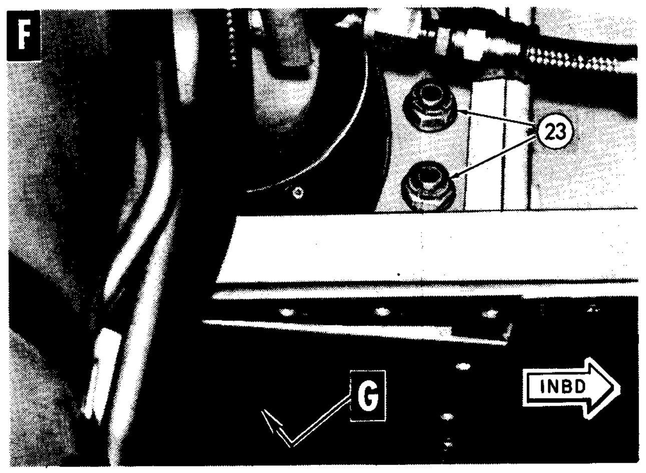
NOTE

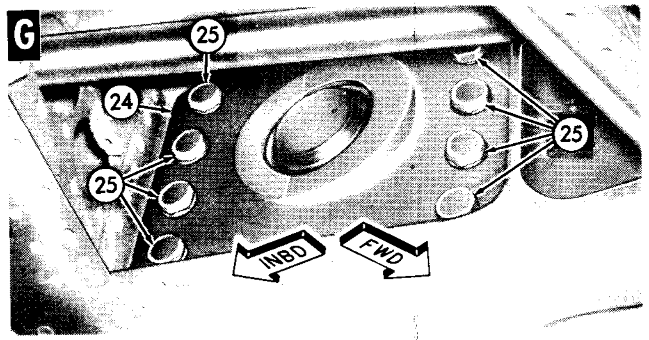
IT IS NOT NECESSARY TO REMOVE RIVETS WHEN REMOVING THE LOWER PORTION OF THE STIFFENER (21).



STEP 12

LOOSEN THE EIGHT NUTS (23) AND BOLTS (25) THAT HOLD THE AFT BEARING CAP (24) TO THE WING STRUCTURE.





STEP 13

CHECK THAT THE AXLE JACK SUPPORTS THE STRUT WEIGHT; THEN BALANCE THE STRUT ASSEMBLY ON THE JACK AND REMOVE THE FOUR BOLTS (22) FORWARD OF THE TRUNNION AND THE EIGHT NUTS (23) AND BOLTS (25) AFT OF THE TRUNNION.

STEP 14

CHECK CLEARANCE BETWEEN THE REAR BEARING CAP (24) AND THE BEARING. IF THERE IS MORE THAN 0.010 INCH CLEARANCE BETWEEN THE BEARING OUTER DIAMETER AND THE REAR BEARING CAP (24) INNER DIAMETER, REPLACE THE REAR BEARING CAP (24).

STEP 15

SLIDE THE REAR BEARING CAP ASSEMBLY (24) OFF THE TRUNNION, AND TILT THE STRUT BACKWARDS SO THAT THE AFT TRUNNION END GOES THROUGH THE HOLE LEFT BY THE REMOVAL OF THE BEARING CAP ASSEMBLY (24). THE FRONT BEARING ASSEMBLY SHOULD SLIDE OUT OF ITS MOUNTING STRUCTURE.

CAUTION

AVOID DAMAGING BEARING SURFACES DURING REMOVAL OF STRUT ASSEMBLY.

STEP 16

AS THE FRONT BEARING ASSEMBLY PULLS FREE, TWIST THE STRUT ASSEMBLY SO THAT THE FORWARD END OF THE TRUNNION SWINGS INBOARD.

CAUTION

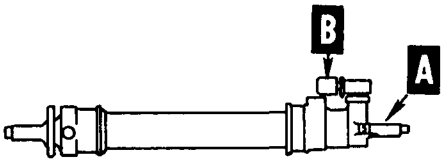
TAKE CARE NOT TO DAMAGE THE AILERON PUSH RODS OR AIRPLANE STRUCTURE DURING GEAR REMOVAL.

STEP 17

AS THE FORWARD END OF THE TRUNNION CLEARS THE WING STRUCTURE, SLIDE THE AFT END OF THE TRUNNION OUT OF THE AFT BEARING CAP HOLE, AND LOWER THE STRUT ASSEMBLY TO THE GROUND.

JS-1-2-X0/3-3206-2

Figure 32-4. Removal of the Main Gear Strut Assembly (Sheet 2 of 2)



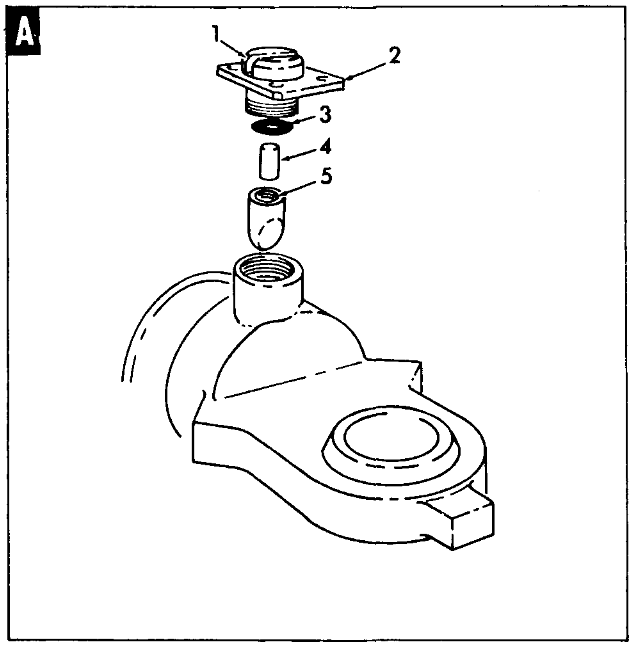
NOTE

THE MAIN GEAR ACTUATING CYLINDER MUST BE REMOVED FROM THE AIRPLANE IN ORDER TO REMOVE, INSTALL, OR ADJUST THE MAIN GEAR DOWNLOCK SWITCH.

ADAPTER REPLACEMENT

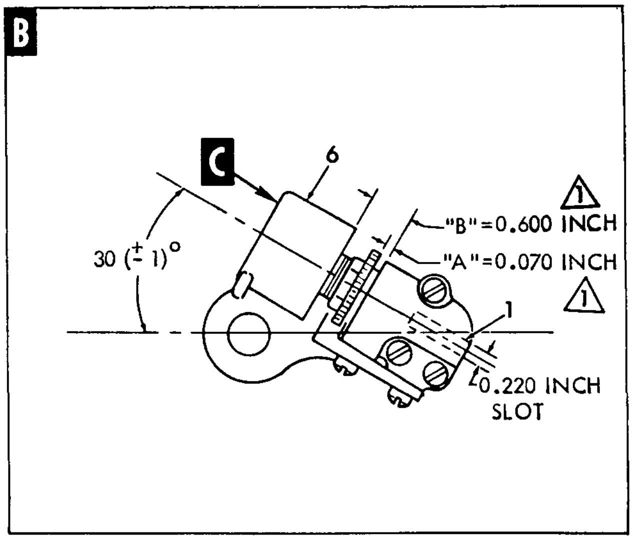
STEP 1

APPLY A LIBERAL AMOUNT OF MIL-G-23827 GREASE TO THE INSIDE OF THE GUIDE (5) AND TO THE ADAPTER POINT (4).



STEP 2

INSTALL THE PACKING RING (3) ON THE PLUG (2) AS SHOWN IN VIEW "A"; THEN INSTALL THE ADAPTER AS SHOWN IN VIEW "A" AND TIGHTEN THE PLUG (2) TO A TORQUE OF 60 INCH-POUNDS. IF THE SLOT (1) DOES NOT MAKE AN ANGLE OF 30 (± 1) DEGREES WITH THE CENTERLINE OF THE ACTUATING CYLINDER, BACK THE PLUG (2) OFF UNTIL IT DOES. (SEE VIEW "B".)



NOTE

△1 "A" AND "B" ARE NOT FINAL DIMENSIONS; THESE DIMENSIONS CHANGE DURING SWITCH ADJUSTMENT.

2. SWITCH AND ADAPTER POSITION FOR LEFT MAIN GEAR SHOWN; SET SWITCH AND ADAPTER 30 (±1) DEGREES TO OTHER SIDE OF CYLINDER CENTERLINE FOR RIGHT MAIN GEAR.

SWITCH REPLACEMENT

STEP 3

INSTALL THE SWITCH (6) AS SHOWN IN VIEWS "B" AND "C"; THEN ADJUST THE SWITCH ACCORDING TO THE INSTRUCTIONS FOR LANDING GEAR SWITCH ADJUSTMENT IN FIGURE 32-8.

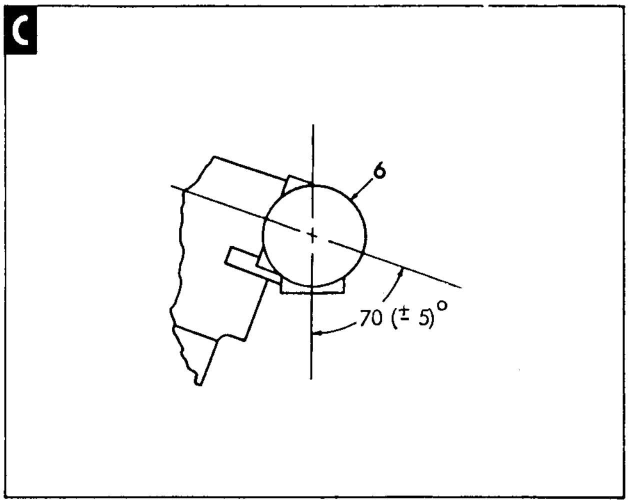


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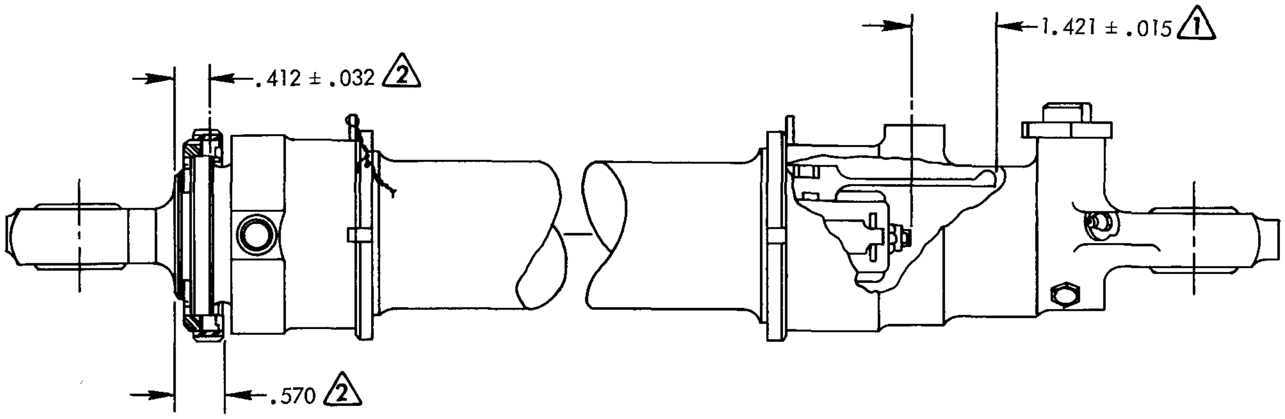
Figure 32-5. Main Gear Downlock Switch and Adapter Replacement



NOTE

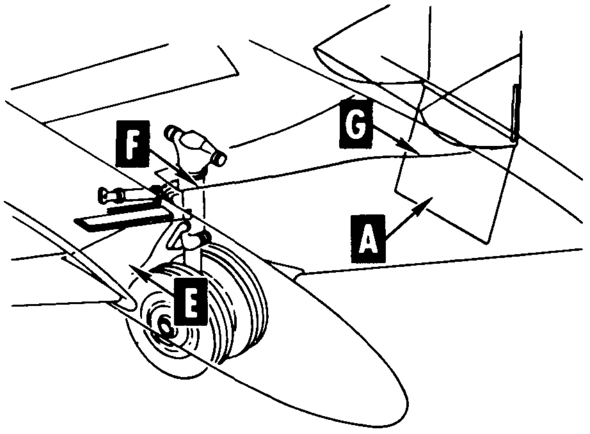
IF IT BECOMES NECESSARY TO DISASSEMBLE THE JR1400 DOWN LOCK ACTUATOR FOR ANY REASON, THE FOLLOWING DIMENSIONS MUST BE CHECKED AND SET AS REQUIRED.

1 THE 1.421 ± .015 DIMENSION CONTROLS THE OPERATION OF THE GEAR DOWN AND LOCKED INDICATOR SWITCH.

2 CLEARANCE ADJUSTED BY ROTATING CAP AND HEAD (THREAD ENGAGEMENT TO BE EQUAL ON EACH) ON CYLINDER TO OBTAIN .412 ± .032 DIMENSION, ADD SHIMS TO OBTAIN .570 DIMENSION WITH ACTUATOR FULLY COMPRESSED. TIGHTEN NUT TO RING TO A TORQUE VALUE OF 60 INCH-POUNDS. ☛ USE LOCTITE AND LOCK TOGETHER BY STAKING WITH CENTER PUNCH AT THE TWO SLOTS IN NUT.

Figure 32-5A. Main Gear Downlock Actuator Adjustments

JS-32-0-005A

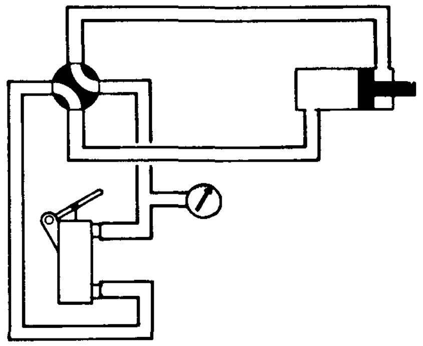


NOTE

THE AIRPLANE MUST BE JACKED WITH THE WHEELS CLEARING THE GROUND.

CAUTION

DO NOT WORK AROUND OPEN MAIN GEAR INBOARD DOORS WHEN THE DOOR ACTUATING CYLINDERS ARE CONNECTED TO THE AIRPLANE HYDRAULIC SYSTEM UNLESS THE INBOARD DOOR SAFETY LOCKS ARE INSTALLED. ACCIDENTAL ACTUATION CAN CRITICALLY INJURE PERSONNEL.



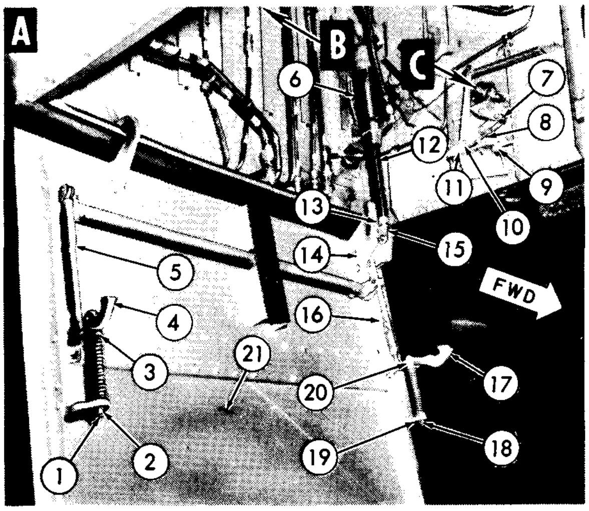
SCHEMATIC DIAGRAM OF
SUGGESTED HANDPUMP CONNECTION
USE MIL-H-5606 HYDRAULIC FLUID

STEP 1

DISCONNECT THE DOOR ACTUATING CYLINDER FROM THE INBOARD DOOR THEN DISCONNECT THE HYDRAULIC LINES FROM THE CYLINDER, AND CONNECT A HYDRAULIC HANDPUMP TO THE CYLINDER. (SEE SCHEMATIC DIAGRAM FOR SUGGESTED METHOD OF CONNECTING THE HANDPUMP).

STEP 2

ADJUST THE STOPS (8) AND (30) ON THE BRACKET ASSEMBLIES (11) AND (29) TO OBTAIN AN OVER-CENTER DIMENSION OF 0.060 TO 0.10 INCH AS MEASURED FROM A LINE BETWEEN THE CENTERS OF THE ROLLERS (7) AND (27) TO THEIR RESPECTIVE LEVER FULCRUMS (10) AND (31) AS SHOWN IN VIEWS C AND D.



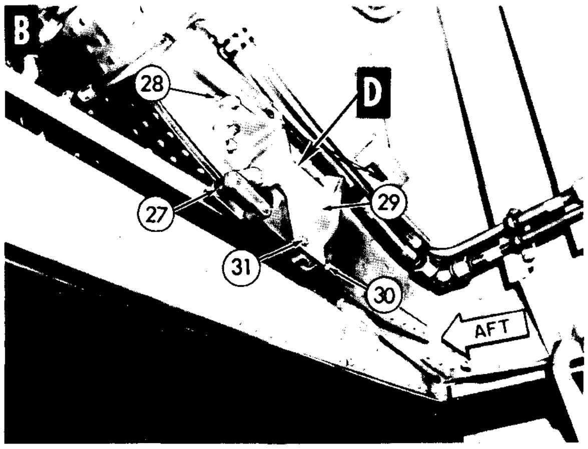


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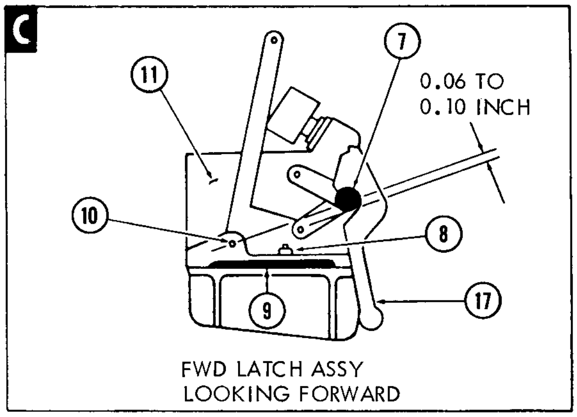
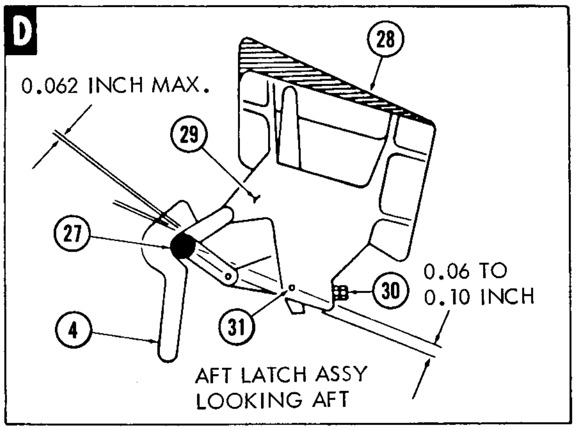
Figure 32-6. Main Gear Door Rigging Insructions (Sheet 1 of 5)





NOTE

THE LINKS MUST BE OVER CENTER THE NOTED AMOUNT AND AGAINST THE ADJUSTING BOLT TO ENSURE THAT THE DOOR REMAINS UP AFTER THE HYDRAULIC PRESSURE IS RELEASED ON THE DOOR ACTUATING CYLINDER. WHEN THE OVER CENTER MECHANISM IS RELEASED, THE LINKS WILL COLLAPSE AND ALLOW THE DOOR TO OPEN.

STEP 3

WITH THE INBOARD DOOR CLOSED AND LATCHED AND THE ACTUATING CYLINDER DEPRESSURIZED, MAKE SURE THAT THE AFT ROLLER (27) IS FREE TO ROTATE. IF THE ROLLER (27) IS NOT FREE TO ROTATE, LOWER THE BRACKET ASSEMBLY (29) AS REQUIRED. THE GAP BETWEEN THE TOP OF THE AFT ROLLER (27) AND THE AFT HOOK (4) SHOULD NOT EXCEED 0.062 INCH WITH THE DOOR CLOSED TIGHTLY.

NOTE

TO MOVE THE FORWARD BRACKET UP OR DOWN, ADD OR REMOVE SHIMS (9). TO MOVE THE AFT BRACKET UP OR DOWN, LOOSEN THE BRACKET AND MOVE IT UP OR DOWN ON THE SERRATED PLATE (28). THERE ARE NO PROVISIONS FOR INBOARD OR OUTBOARD ADJUSTMENTS.

STEP 4

WITH THE DOOR CLOSED AND THE ACTUATING CYLINDER PRESSURIZED AT 2,700 TO 3,000 PSI TO THE FULLY RETRACTED POSITION, ADJUST THE LENGTH OF THE PISTON SHAFT (12) TO POSITION THE ROLLER (ROLLER AND SLOT NOT VISIBLE IN VIEW A, BUT CALLOUT (15) SHOWS LOCATION) AT THE TOP OF THE SLOTTED HOLE IN THE HINGE BRACKET (14). CONNECT THE CYLINDER TO THE DOOR; THEN LOOSEN THE LOCKNUT (13) AND TURN THE PISTON SHAFT (12) ONE COMPLETE REVOLUTION TO SHORTEN CYLINDER LENGTH. RETIGHTEN THE LOCKNUT UPON COMPLETION OF THIS STEP

STEP 5

WITH THE DOOR STILL CLOSED, BUT WITH THE ACTUATING CYLINDER DEPRESSURIZED, MAKE SURE THE ROLLERS (7) AND (27) FIT SNUGLY AGAINST THE TOPS OF THE HOOKS (17) AND (4). (SEE VIEWS C AND D). ADJUST THE LENGTHS OF THE ROD ASSEMBLIES (5) AND (16) TO ALIGN THE CENTERS OF THE ROLLERS (7) AND (27) WHEN THE SLIDING PORTION OF THE ROD ASSEMBLIES (5) AND (16) ARE BOTTOMED AGAINST THE FIXED PORTIONS OF THE ROD ASSEMBLIES. LOCK THE AFT ROD (5) IN THIS POSITION. MEASURE THE LENGTH OF THE FORWARD ROD ASSEMBLY (16); THEN DECREASE ITS LENGTH 0.10 TO 0.14 INCH. LOCK THE FORWARD ROD ASSEMBLY AND CONNECT IT TO THE DOOR. MAKE SURE A GAP OF 0.002 TO 0.030 INCH EXISTS BETWEEN THE NUT (18) AND THE WASHER (19), AND A GAP OF 0.000 TO 0.030 INCH EXISTS BETWEEN THE NUT (1) AND THE WASHER (2), (SEE VIEW A).

STEP 6

USE THE HANDPUMP TO CYCLE THE DOOR, AND OBSERVE DOOR OPERATION CLOSELY. AS THE DOOR OPENS, THE AFT HOOK MUST DISENGAGE BEFORE THE FORWARD HOOK. IF THE FORWARD HOOK DISENGAGES FIRST, CHECK FOR WORN COMPONENTS AND ROD ADJUSTMENT AS GIVEN IN STEP (5). REPLACE ANY WORN COMPONENTS AND START THE RIGGING PROCEDURE AGAIN AT STEP (2).

NOTE

MAXIMUM PRESSURE REQUIRED TO UNLOCK DOORS SHALL BE 700 PSI. CHECK WITH A HYDRAULIC CART OR HAND PUMP.

STEP 7

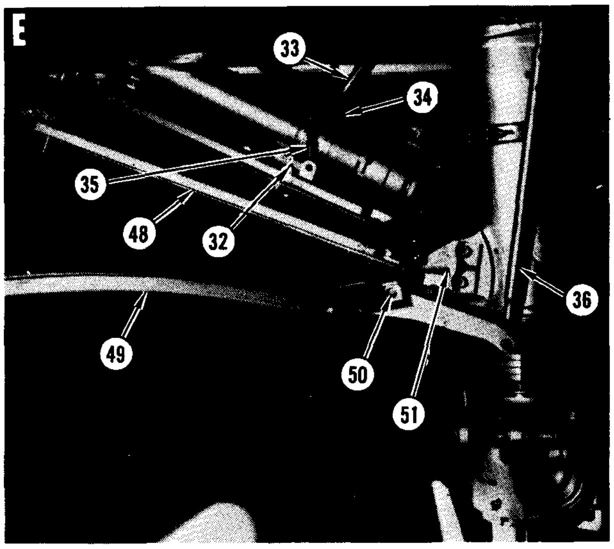
AFTER ALL NECESSARY CORRECTIONS ARE MADE TO THE INBOARD DOOR MECHANISM, CLOSE THE INBOARD DOOR, DISCONNECT THE HANDPUMP FROM THE DOOR ACTUATING CYLINDER, AND RECONNECT THE CYLINDER TO THE AIRPLANE HYDRAULIC SYSTEM.

JS-1-2-X0/2-3206-2

Figure 32-6. Main Gear Door Rigging Instructions (Sheet 2 of 5)

STEP 8

LOOSEN THE ATTACH BRACKETS (32) AND (34) AND POSITION THE BRACKET (32) 18.96 INCHES FROM THE OUTBOARD PIVOT POINT OF THE MAIN GEAR STRUT ACTUATING CYLINDER. IT MAY BE NECESSARY TO REPOSITION THESE BRACKETS TO RIG THE DOORS PROPERLY.



STEP 9

DISCONNECT THE MAIN GEAR STRUT ACTUATING CYLINDER FROM THE AIRPLANE HYDRAULIC SYSTEM AND CONNECT IT TO A HYDRAULIC HANDPUMP. (SEE SCHEMATIC DIAGRAM ON SHEET 1 FOR SUGGESTED HANDPUMP CONNECTION).

STEP 10

OPEN THE MAIN GEAR INBOARD DOOR, INSTALL AN INBOARD DOOR GROUND LOCK, REMOVE THE MAIN GEAR GROUND LOCK AND THEN USE THE HANDPUMP TO RETRACT THE MAIN GEAR. ADJUST THE ROD ASSEMBLIES (33), (35), AND (36) AS NECESSARY TO OBTAIN A CLOSED DOOR FIT WITH STEPS AS FOLLOWS: STEPS PARALLEL TO THE AIRSTREAM SHOULD NOT EXCEED 0.060 INCH EITHER POSITIVE OR NEGATIVE. STEPS PERPENDICULAR TO THE AIRSTREAM SHOULD NOT EXCEED 0.030 INCH EITHER POSITIVE OR NEGATIVE; HOWEVER, IF THE PROTRUDING EDGE IS ROUNDED OFF, THE STEP MAY BE AS MUCH AS 0.060 INCH.

STEP 11

EXTEND THE GEAR WITH THE HANDPUMP, AND MAKE SURE THE DOORS AND MECHANISM DO NOT BIND OR INTERFERE WITH AIRPLANE SKIN OR STRUCTURE. PAY PARTICULAR ATTENTION TO THE ROD ASSEMBLIES AND THE ENDS OF THE DOORS WHEN CHECKING OPERATION.

STEP 12

CORRECT DOOR OPERATION AS NECESSARY; THEN DISCONNECT THE HANDPUMP FROM THE MAIN GEAR STRUT ACTUATING CYLINDER AND RECONNECT THE CYLINDER TO THE AIRPLANE HYDRAULIC SYSTEM.

STEP 13

CONNECT AN EXTERNAL SOURCE OF HYDRAULIC POWER TO THE NORMAL (NO. 1) HYDRAULIC SYSTEM GROUND TEST CONNECTIONS, AND PRESSURIZE THE SYSTEM TO 2,700 TO 3,000 PSI ADJUST THE POWER SOURCE TO DELIVER NO FLOW AT 3000 PSI AND 3 GPM AT 2700 PSI.

STEP 14

REMOVE GROUND LOCKS AS NECESSARY, THEN RETRACT THE LANDING GEAR AND CHECK ALL MAIN GEAR DOOR GAPS AND STEPS WITH THE DOORS CLOSED. STEPS PARALLEL TO THE AIRSTREAM SHOULD NOT EXCEED 0.060 INCH EITHER POSITIVE OR NEGATIVE. STEPS PERPENDICULAR TO THE AIRSTREAM SHOULD NOT EXCEED 0.030 INCH EITHER POSITIVE OR NEGATIVE; HOWEVER, IF THE PROTRUDING EDGE IS ROUNDED OFF, THE STEP MAY BE AS MUCH AS 0.060 INCH. MAIN GEAR DOOR GAPS SHOULD BE BETWEEN 0.060 AND 0.120 INCH.

STEP 15

EXTEND THE LANDING GEAR AND REDUCE NORMAL HYDRAULIC SYSTEM PRESSURE TO ZERO.

STEP 16

REINSTALL THE MAIN GEAR GROUND LOCK AND MAKE SURE ALL PULLEYS AND LEVERS OF THE STRUT-COVER-DOOR LATCHING MECHANISM MOVE FREELY WITH NO BINDING AND/OR INTERFERENCE.

STEP 17

DISCONNECT THE MAIN GEAR STRUT ACTUATING CYLINDER FROM THE AIRPLANE HYDRAULIC SYSTEM, AND CONNECT THE HYDRAULIC HANDPUMP TO THE CYLINDER, (SEE SCHEMATIC DIAGRAM FOR SUGGESTED HANDPUMP CONNECTION). REMOVE THE MAIN GEAR GROUND LOCK.

STEP 18

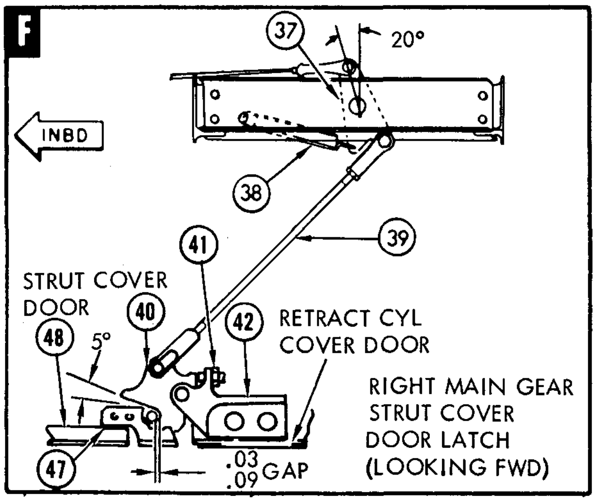
RETRACT THE MAIN GEAR TO THE UP-AND-LOCKED POSITION.

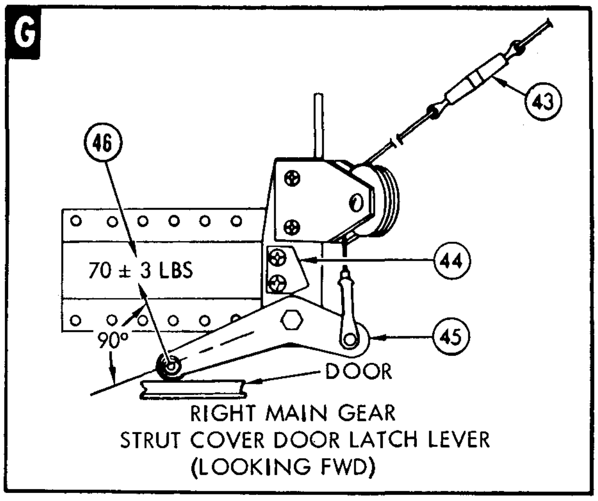
JS-1-2-X0/2-3206-3

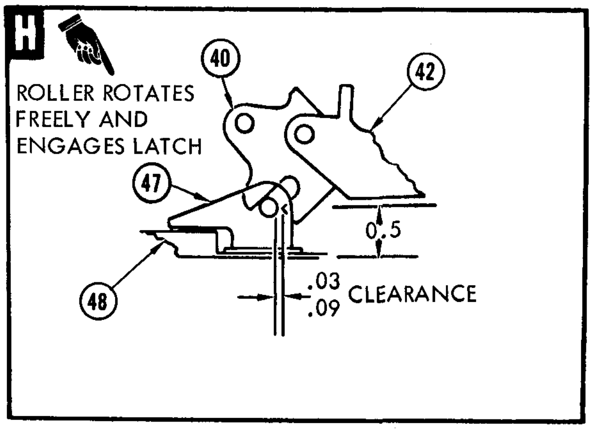
Figure 32-6. Main Gear Door Rigging Instructions (Sheet 3 of 5)

STEP 19

WITH THE MAIN GEAR UP AND LOCKED, REMOVE THE GEAR ACCESS PANEL ON TOP OF THE WING, AND ADJUST THE STOP (41) TO BOTTOM AGAINST THE LATCH (40) WITH THE LATCH ENGAGING THE STRUT-COVER DOOR.







STEP 20

WITH THE LATCH (40) AGAINST THE ADJUSTED STOP (41), ADJUST THE ROD(39)TO POSITION THE LEVER (37) AS SHOWN IN VIEW F. OPEN THE MAIN GEAR INBOARD DOORS, INSTALL THE INBOARD DOOR GROUND LOCKS THEN ADJUST THE ROD ASSEMBLY (35) ON THE STRUT-COVER DOOR TO PROVIDE 0.5 ± 0.1 DIMENSION IN VIEW H. ADJUST LATCH BRACKET (47) INBOARD OR OUTBOARD TO GIVE 0.03 TO 0.09 CLEARANCE. THE LATCH (40) CLOSES THE STRUT-COVER DOOR WHEN THE INBOARD MAIN GEAR DOOR CLOSES.

STEP 21

CLOSE THE MAIN GEAR INBOARD DOORS AFTER MAKING SURE THE LATCHING MECHANISM CABLES DO NOT RUB AGAINST THE AIRPLANE STRUCTURE.

STEP 22

WITH THE DOORS STILL CLOSED, ADJUST THE TURNBUCKLE (43) TO OBTAIN FROM 80 - TO 100 - POUNDS TENSION IN THE LATCHING MECHANISM CABLES.

STEP 23

OPEN THE MAIN GEAR INBOARD DOOR, INSTALL THE INBOARD DOOR GROUND LOCK; THEN EXTEND THE MAIN GEAR TO RELIEVE CABLE TENSION. MAKE SURE THE LATCH (40) IS FULLY RELEASED AND BOTTOMED AGAINST THE LATCH BRACKET (42). IF THE LATCH IS NOT FULLY RELEASED, CHECK THE LEVER (45) TO SEE IF THE FIXED STOP (44) IS PREVENTING THE LATCH FROM ATTAINING THIS POSITION. CHECK ALSO FOR BINDING IN LEVER (37). IF THE FIXED STOP (44) IS RESTRICTING MECHANISM TRAVEL TOO MUCH, READJUST TURNBUCKLE (43) AND ROD (39) TO CORRECT THIS CONDITION; THEN REPEAT RIGGING FROM STEP (20). CORRECT ANY DEFICIENCIES IN THE SYSTEM AND RE-RIG THE LATCHING SYSTEM ACCORDINGLY. MAKE SURE THE SPRING (38) IS SERVICEABLE AND SECURELY ATTACHED.

STEP 24

SECURE ALL ADJUSTMENT PROVISIONS AFTER COMPLETING THE RIGGING, AND SERVICE THE HYDRAULIC RESERVOIRS.

STEP 25

AFTER THE MAIN GEAR DOORS AND LATCHING MECHANISMS ARE RIGGED PROPERLY CHECK THE ADJUSTMENT OF ALL MAIN-GEAR AND MAIN-GEAR-DOOR SWITCHES. READJUST SWITCHES AS NECESSARY PER FIGURE 32-8.

STEP 26

REINSTALL THE MAIN GEAR GROUND LOCK, DISCONNECT THE HYDRAULIC HANDPUMP FROM THE MAIN GEAR STRUT ACTUATING CYLINDER, AND RECONNECT THE ACTUATING CYLINDER TO THE AIRPLANE HYDRAULIC SYSTEM. REPLACE THE ACCESS PANEL ON TOP OF THE WING.

JS-1-2-X0/5-3206-4

Figure 32-6. Main Gear Door Rigging Instructions (Sheet 4 of 5)

STEP 27

IF THE NOSE GEAR DOORS ARE NOT TO BE RIGGED, REMOVE ALL GEAR AND DOOR LOCKS, CYCLE THE LANDING GEAR SEVERAL TIMES TO BLEED THE HYDRAULIC SYSTEM, THEN PERFORM AN OPERATIONAL CHECKOUT OF THE LANDING GEAR. IF THE NOSE GEAR DOORS ARE TO BE RIGGED, RIG THEM BEFORE PERFORMING AN OPERATIONAL CHECKOUT OF THELANDING GEAR.

STEP 28 (SEE VIEW **E**)

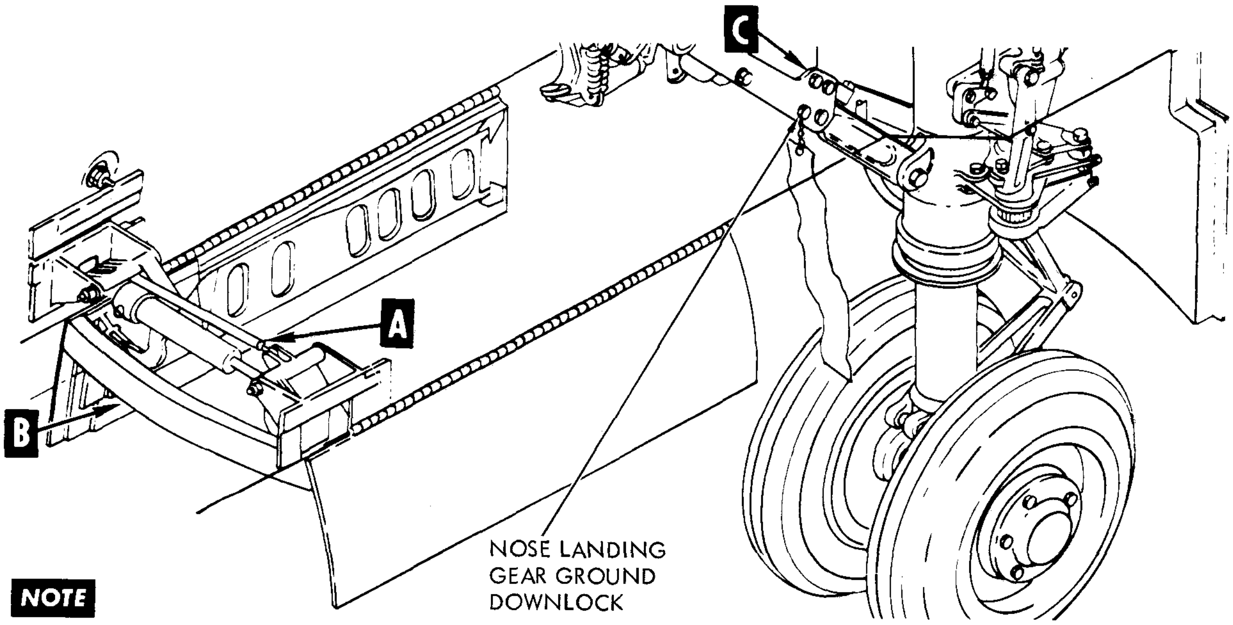
WITH THE GEAR UP AND LOCKED AND BEFORE THE INBOARD DOOR COMPLETELY CLOSES, CAREFULLY OBSERVE THE FIT OF THE STRUT COVER DOOR (48) AND WHEEL COVER DOOR (49) AT THEIR HINGE LINE (50) WITH THE WING SURFACE. WHEN THE INBOARD DOOR (21) (SEE VIEW A) CLOSES AND HYDRAULIC PRESSURE IS RELIEVED ON THE SIDE BRACE ACTUATOR, THE SPINNER (9) (SEE FIGURE 32-9A) ON THE GEAR AXLE SETTLES DOWN ON THE MAIN GEAR UPLOCK LATCHES (5) (SEE FIGURE 32-9A). THE HINGE LINE (50) OF THE DOORS SHOULD DROP DOWN APPROXIMATELY 0.06 INCHES. IF IT DOES NOT, THEN DROP THE DOOR HINGE FITTINGS ON THEIR SERRATED PLATES (51) UNTIL A DISCERNIBLE MOVEMENT IS OBTAINED AT THE END OF THE GEAR CYCLE.

NOTE

THE GEAR SPINNER MUST CLOSE THE GEAR UPLOCK LATCHES AS THE GEAR GOES UP. IF THE HINGE LINE AREA OF THE COVER DOORS INTERFERE WITH THE WING SURFACE BEFORE THE SPINNER TRIPS THE UPLOCK LATCHES COMPLETELY CLOSED, THE GEAR MAY LATER FALL OUT OF THE UPLOCK AND REST ON THE INBOARD DOOR. THIS WILL RESULT IN A RED WARNING IN THE LANDING GEAR CONTROL HANDLE AND AN ABORTED FLIGHT.

JS-1-2-X0/2-3206-5

Figure 32-6. Main Gear Door Rigging Instructions (Sheet 5 of 5)



NOTE

THE AIRPLANE MUST BE JACKED WITH THE WHEELS CLEARING THE GROUND.

STEP 1

REMOVE THE ACCESS PANEL FORWARD OF THE NOSE WHEEL WELL AND CONNECT HYDRAULIC HANDPUMPS TO THE NOSE GEAR STRUT ACTUATING CYLINDER AND TO THE ACTUATING CYLINDER FOR THE FORWARD NOSE GEAR DOORS. (SEE SCHEMATIC DIAGRAM FOR SUGGESTED HANDPUMP CONNECTING).

STEP 2

REMOVE THE LOCKING DEVICES FROM THE DOOR ACTUATING CYLINDER ROD-END CONNECTION (1), THE ADJUSTABLE END (8) OF THE ROD (7), AND THE ADJUSTABLE ENDS OF THE GOOSE-NECK ARMS (6). (SEE VIEWS "A" AND "B"). LOOSEN THE CONNECTIONS.

STEP 2A

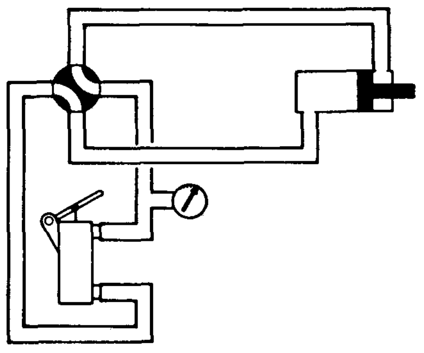
CHECK THAT ARM (5) IS MOUNTED IN THE CORRECT POSITION ON THE CONTROLLABLE CHECK VALVE SHAFT BY DISCONNECTING LINKAGE (4) AT OUTBOARD END AND THE SPRING AT THE INBOARD END. THE BOTTOM OF THE ROLLER ON THE ARM (5) SHOULD BE 0.2 TO 0.5 INCH ABOVE THE FUSELAGE OUTER SKIN LINE WHEN THE SHAFT IS ROTATED CLOCKWISE (LOOKING FORWARD) AGAINST THE VALVE MECHANICAL STOP. RECONNECT LINKAGE (4) AND SPRING AND ADJUST LINKAGE (4) LENGTH SO THAT BOTTOM OF ROLLER ON ARM EXTENDS 0.4 TO 0.6 INCH BELOW THE FUSELAGE OUTER SKIN LINE. ENSURE THAT NO LINKAGE RUBBING OR BINDING IS POSSIBLE FOR THE ENTIRE STROKE. ENSURE THAT ROLLER CONTACT WITH DOOR WILL ALSO PROVIDE PROPER ACTION. ADJUSTMENTS, IF REQUIRED, CAN BE MADE BY USING DIFFERENT LENGTH SCREWS AND SPACERS TO ATTACH ARM (5).

STEP 3

LOOSEN THE STRUT DOOR ADJUSTMENTS (9) AND THE ATTACH BRACKET (11). (SEE VIEW "C").

STEP 4

REMOVE THE NOSE GEAR GROUND LOCK (12), (SEE VIEW "C") AND USE THE HYDRAULIC HANDPUMPS TO RETRACT THE NOSE GEAR; THEN CLOSE THE NOSE GEAR FORWARD DOORS.

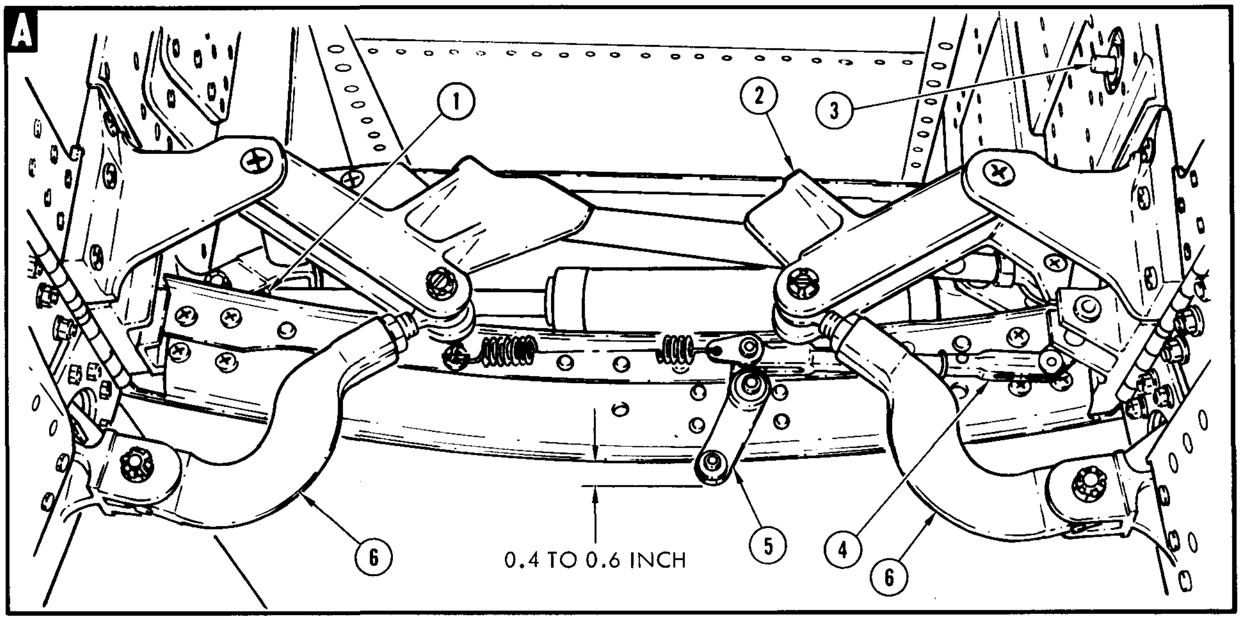


SCHEMATIC DIAGRAM OF SUGGESTED HANDPUMP CONNECTION

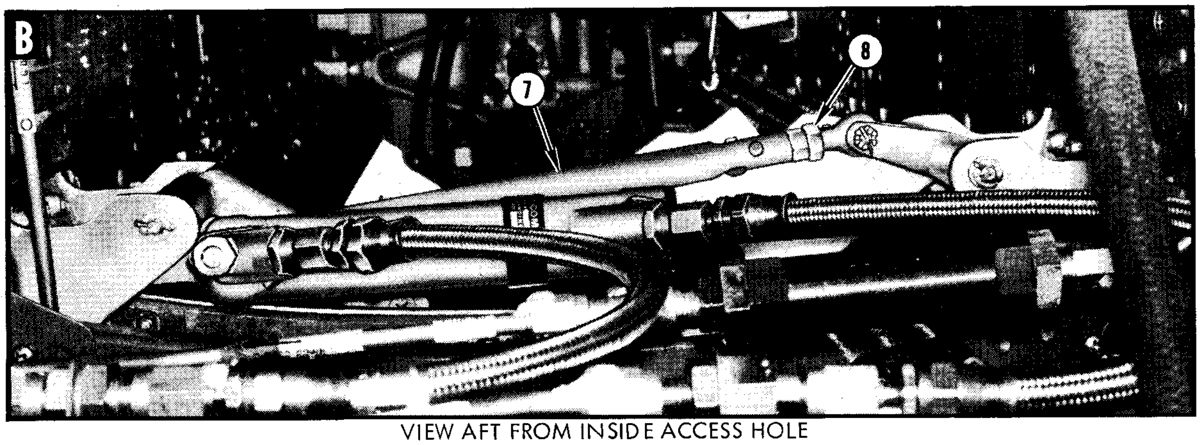
☞ USE MIL-H-5606 HYDRAULIC FLUID

SMR151-1-2-X0/11-3207-1

Figure 32-7. Nose Gear Door Rigging Instructions (Sheet 1 of 4)



VIEW FWD FROM INSIDE WHEEL WELL



VIEW AFT FROM INSIDE ACCESS HOLE

STEP 5

WITH THE NOSE GEAR DOORS CLOSED AND LOCKED, REACH THROUGH THE ACCESS OPENING FORWARD OF THE NOSE WHEEL WELL AND DISCONNECT THE DOOR LINKAGE ROD (7). (SEE VIEW B).

STEP 6

WITH THE NOSE GEAR DOORS STILL CLOSED AND LOCKED, ADJUST THE LENGTHS OF THE GOOSE-NECK ARMS (6) TO OBTAIN STEPS AT THE FRONTS OF THE NOSE GEAR DOORS NO GREATER THAN 0.020 INCH EITHER POSITIVE OR NEGATIVE. GAPS AROUND THE NOSE GEAR FORWARD DOORS SHOULD BE FROM 0.030 TO 0.060 INCH EXCEPT AT THE FRONT WHERE GAPS SHOULD BE 0.060 TO 0.120 INCH.

NOTE

IF THE SWITCH PLUNGER (3) HOLDS THE TORQUE TUBE (2) AWAY FROM THE WHEEL WELL STRUCTURE, ADJUST THE SWITCH TO RELIEVE THIS CONDITION. (SEE STEP 17).

NOTE

THE NOSE GEAR DOOR LATCH SHOULD HAVE A PRELOAD OF APPROXIMATELY 20 POUNDS. IF THE FORCE REQUIRED TO BREAK THE LINKS OVER-CENTER IS TOO LOW, ADJUST THE GOOSE-NECK ARMS (6) TO OBTAIN THE PROPER PRELOAD.

STEP 7

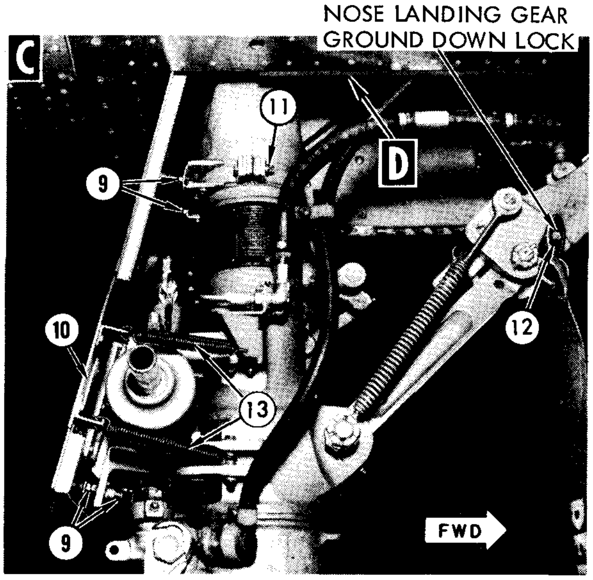
ADJUST THE STRUT DOOR ADJUSTMENTS (9) (SEE VIEW "C") AS REQUIRED TO OBTAIN STEPS AROUND THE STRUT DOOR OF NO MORE THAN 0.020 INCH EITHER POSITIVE OR NEGATIVE. GAPS AROUND THE FRONT AND SIDES OF THE NOSE GEAR STRUT DOOR SHOULD BE 0.030 TO 0.060 INCH. THE GAP AT THE REAR OF THE STRUT DOOR SHOULD BE APPROXIMATELY 1 INCH.

JS-1-2-XO/5-3207

Figure 32-7. Nose Gear Door Rigging Instructions (Sheet 2 of 4)

NOTE

IT MAY BE NECESSARY TO EXTEND THE NOSE GEAR AND REPOSITION THE BRACKET (11) TO OBTAIN THE PROPER GAPS.



STEP 8

WITH THE DOORS IN THE POSITIONS AS DESCRIBED IN STEPS (6) AND (7), CONNECT THE ROD (7) (SEE VIEW "B") WITHOUT DISTRUBING THE POSITIONS OF THE DOORS.

STEP 9

WITH THE DOORS STILL CLOSED AND LOCKED, MAKE SURE THE DOOR-OPERATED CONTROLLABLE CHECK VALVE LINKAGE (4) HAS SOME PLAY WHEN THE DOORS ARE FULLY CLOSED. IF THE LINKAGE BINDS, RECHECK THE POSITION OF THE ARM (5) AS INDICATED IN STEP 2A.

STEP 10

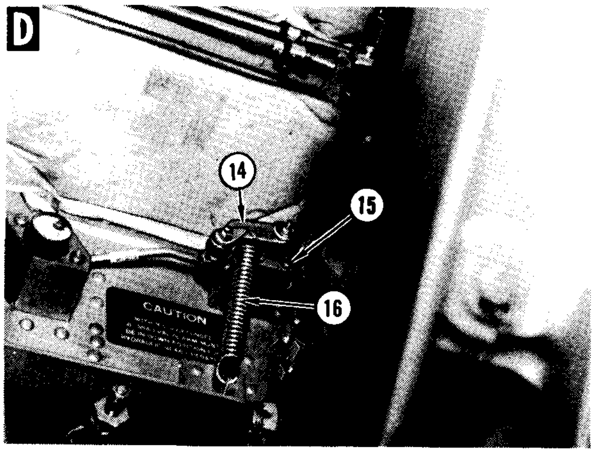
USE THE HANDPUMPS TO OPEN THE NOSE GEAR FORWARD DOORS, AND THEN TO EXTEND THE NOSE GEAR. MAKE SURE NEITHER THE DOORS NOR THE DOOR LINKAGES BIND OR RUB AGAINST THE AIRPLANE STRUCTURE OR SKIN.

STEP 11

MAKE SURE THE ARM (14) OF THE GEAR-OPERATED CONTROLLABLE CHECK VALVE (15) RETURNS TO THE DOWN POSITION WHEN THE NOSE GEAR EXTENDS. MAKE SURE THE SPRING (16) IS SERVICEABLE AND SECURELY ATTACHED.

STEP 12

SECURE ALL NOSE GEAR DOOR LINKAGE AND ADJUSTMENT POINTS AS REQUIRED, AND DISCONNECT THE HYDRAULIC HANDPUMPS FROM THE GEAR AND THE DOOR ACTUATING CYLINDERS.



STEP 13

CONNECT THE ACTUATING CYLINDERS TO THE AIRPLANE HYDRAULIC SYSTEM, AND CONNECT AN EXTERNAL SOURCE OF HYDRAULIC POWER TO THE NORMAL (NO. 1) HYDRAULIC SYSTEM GROUND TEST CONNECTIONS. ADJUST THE POWER SOURCE TO DELIVER NO FLOW AT 3000 PSI AND 3 GPM AT 2700 PSI.

STEP 14

REMOVE BOTH MAIN GEAR GROUND LOCKS, AND RETRACT THE LANDING GEAR WHILE OBSERVING CLOSELY THE OPERATION OF THE NOSE GEAR DOORS. THE NOSE GEAR FORWARD DOORS SHOULD START TO CLOSE WHEN THE STRUT DOOR IS 1/10 TO 1/4 INCH FROM THE FULLY CLOSED POSITION. THIS DISTANCE IS FROM EITHER FORWARD CORNER OF THE STRUT DOOR TO THE NEAREST FUSELAGE SKIN.

STEP 15

IF THE NOSE GEAR FORWARD DOORS DO NOT START TO CLOSE AS SPECIFIED IN STEP (4). ADJUST THE ARM (14) TO OBTAIN THE PROPER SEQUENCING. (SEE VIEW D). INSTALL THE NOSE GEAR GROUND LOCK UPON COMPLETING THIS STEP.

STEP 16

WITH THE LANDING GEAR EXTENDED, ADJUST THE STEERING CYLINDER DOOR (10) (SEE VIEW C) TO FIT FLUSH AGAINST THE STRUT DOOR WITH STEPS NOT EXCEEDING 0.020 INCH EITHER POSITIVE OR NEGATIVE. GAPS AROUND THE STEERING CYLINDER DOOR SHOULD BE BETWEEN 0.030 AND 0.060 INCH.

JS-1-2-X0/5-3207-3

Figure 32-7. Nose Gear Door Rigging Instructions (Sheet 3 of 4)

NOTE

MAKE SURE THE TWO SPRINGS (13) (SEE VIEW "C") ARE SERVICEABLE AND SECURELY ATTACHED.

STEP 17

CHECK THE ADJUSTMENT OF ALL NOSE-GEAR AND NOSE-GEAR DOOR SWITCHES, AND ADJUST AS NECESSARY PER FIGURE 32-8.

STEP 18

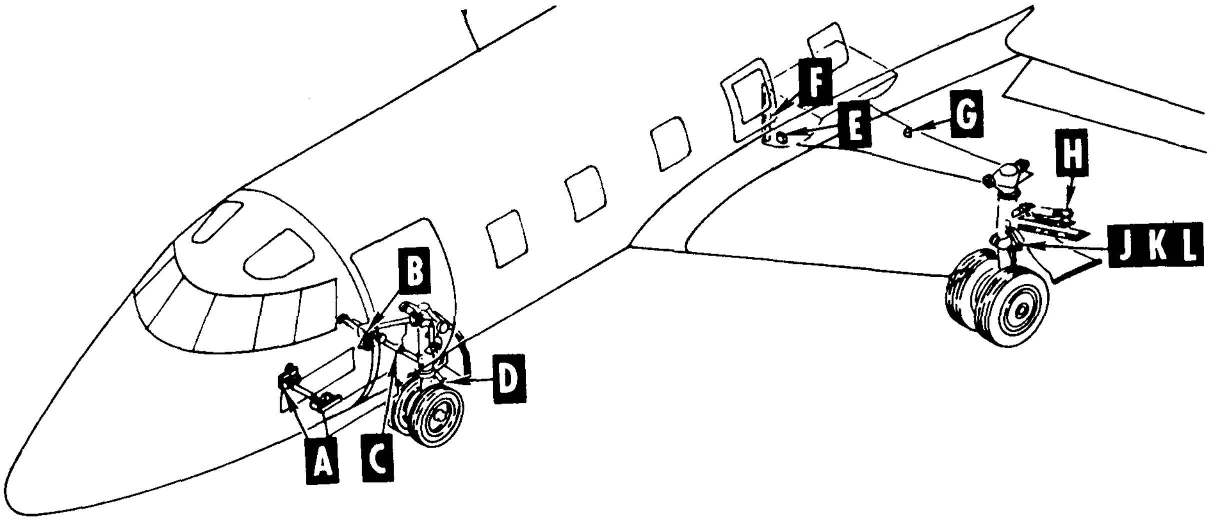
REPLACE THE ACCESS PANEL FORWARD OF THE NOSE WHEEL WELL.

STEP 19

IF THE MAIN GEAR DOORS ARE NOT TO BE RIGGED, REMOVE ALL GEAR AND DOOR LOCKS, CYCLE THE LANDING GEAR SEVERAL TIMES TO BLEED THE SYSTEM, THEN PERFORM AN OPERATIONAL CHECKOUT OF THE LANDING GEAR. IF THE MAIN GEAR DOORS ARE TO BE RIGGED, RIG THEM BEFORE PERFORMING AN OPERATIONAL CHECK-OUT OF THE LANDING GEAR.

JS-1-2-X0/2-3207-4

Figure 32-7. Nose Gear Door Rigging Instructions (Sheet 4 of 4)



NOTE

1. LEFT MAIN GEAR AND DOOR SWITCHES ARE SHOWN; OTHER MAIN GEAR SWITCHES ARE SIMILARLY LOCATED.
2. WHERE SLOTTED SWITCH-MOUNTING HOLES ARE PROVIDED, MAKE SURE THE SWITCH TAB-WASHERS ARE TURNED SO THE TABS FIT INTO THE SLOTS TO PREVENT SWITCH ROTATION.

NOSE GEAR DOOR UPLOCK SWITCH ADJUSTMENT (SEE VIEW A)

STEP 1

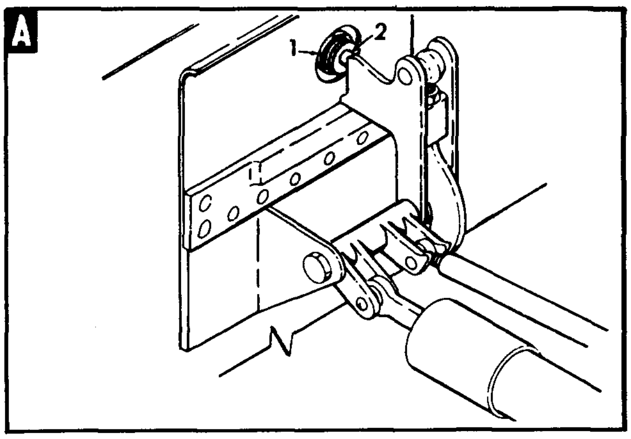
ADJUST THE SWITCH (1) SO THAT THE PLUNGER (2) TRAVELS 0.10 TO 0.15 WHEN THE FORWARD DOORS (SEE LOCATOR) CLOSE AND LOCK. DETERMINE THE MAKE/BREAK POSITION OF THE SWITCH PLUNGER (2) AS FOLLOWS:

> DISCONNECT WIRE G18H20 FROM TERMINAL 1 OF TERMINAL STRIP C5 (RH CONSOLE). WIRE G18H20 AND TERMINAL 2 WILL INDICATE A CLOSED CIRCUIT FOR DOOR NOT LOCKED AND AN OPEN CIRCUIT FOR DOORS CLOSED AND LOCKED. MAKE A REMOVABLE MARK ON THE PLUNGER (2) OF THE SWITCH (1) AT THE MAKE/BREAK POSITION ADJACENT TO BARREL END AND THEN A REMOVABLE MARK 0.10 TO 0.15 TOWARDS THE END OF THE PLUNGER. ADJUST THE NUTS ON THE BARREL OF THE SWITCH (1) TO HOLD THE PLUNGER AT THE LAST MARK WITH THE DOORS CLOSED AND LOCKED.

TORQUE AND SAFETY-WIRE NUTS AS REQUIRED.

NOTE

THIS SWITCH (1) MUST BE OPEN (CONTACTS 1 AND 3, AND 4 AND 6) TO CORRECTLY SEQUENCE HYDRAULIC SELECTOR VALVES IN THE MAIN WHEEL WELL AND TO ENSURE NO RED LIGHT IN THE LANDING GEAR CONTROL HANDLE IN THE CREW COMPARTMENT WITH THE NOSE GEAR AND DOORS UP AND LOCKED. ACCURATE CONTROL OF THE NOSE GEAR DOOR UPLOCK SWITCH PLUNGER IS REQUIRED BECAUSE THE OVER CENTER TRAVEL OF THE GOOSE-NECK ARM (SEE FIGURE 32-7, SHEET 2) IS APPROXIMATELY 0.25 FROM NO LOCK TO FULL OVER CENTER. SAFETY-WIRE THE SWITCH (1) AS REQUIRED.



JS-1-2-X0/2-3208-1

Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 1 of 6)

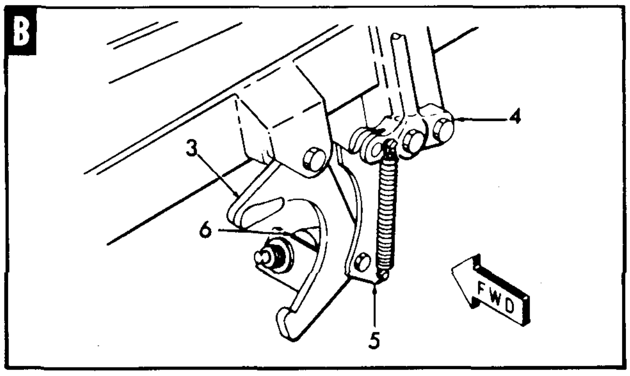
NOSE GEAR UPLOCK SWITCH ADJUSTMENT (SEE VIEW B)

STEP 2

WITH THE NOSE GEAR IN THE UPLOCK LATCH (3) AND WITH THE GAP BETWEEN THE STOP FACES OF LINK (5) AND LINK (4) NO MORE THAN 0.30 INCH, SWITCH LEADS NO. 1 AND 3 SHOULD INDICATE ELECTRICAL CONTINUITY.

STEP 3

WITH THE NOSE GEAR OUT OF THE UPLOCK LATCH (3) AND/OR WITH THE GAP BETWEEN THE STOP FACES OF LINK (4) AND LINK (5) OVER 0.060 INCH, SWITCH LEADS NO. 1 AND 2 SHOULD INDICATE ELECTRICAL CONTINUITY.



STEP 4

SAFETY THE SWITCH (6) AS REQUIRED.

NOSE GEAR DOWNLOCK SWITCH ADJUSTMENT (SEE VIEW C)

STEP 5

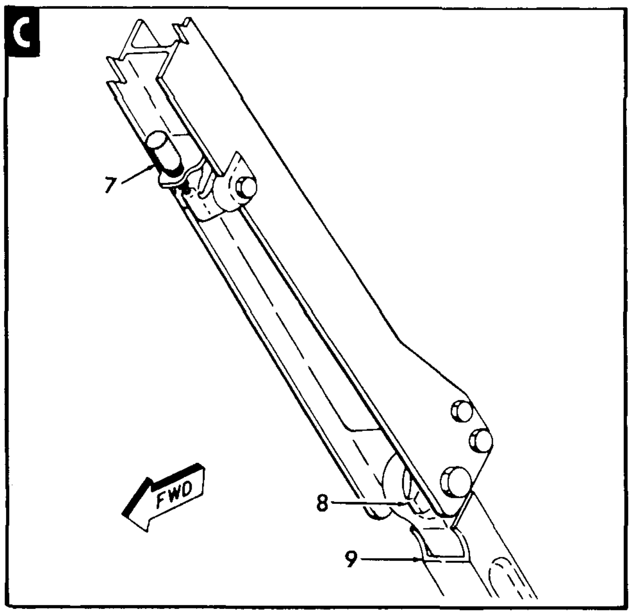
WITH THE LATCH (8) ENGAGED WITH THE DOWNLOCK SURFACE OF LINK (9) WITH AT LEAST 0.25 INCH OVERLAP, SWITCH LEADS NO. 1 AND 3 SHOULD INDICATE ELECTRICAL CONTINUITY.

STEP 6

SWITCH LEADS NO. 1 AND 2 SHOULD INDICATE ELECTRICAL CONTINUITY WHEN SUCH OVERLAP IS LESS THAN 0.12 INCH AND/OR WHEN THE GEAR IS NOT LOCKED IN THE DOWN POSITION.

STEP 7

SAFETY THE SWITCH (7) AS REQUIRED.



NOSE GEAR SCISSORS SWITCH ADJUSTMENT (SEE VIEW D)

STEP 8

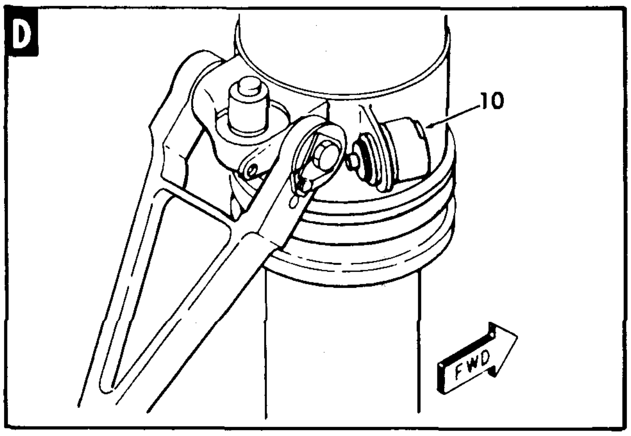
WITH THE STRUT FULLY EXTENDED, SWITCH LEADS NO. 1 AND 2 SHOULD INDICATE ELECTRICAL CONTINUITY.

STEP 9

WITH THE STRUT COMPRESSED MORE THAN 2 1/4 INCHES FROM THE FULLY EXTENDED POSITION, SWITCH LEADS NO. 1 AND 3 SHOULD INDICATE ELECTRICAL CONTINUITY.

STEP 10

SAFETY THE SWITCH (10) AS REQUIRED.



JS-1-2-X0/2-3208-2

Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 2 of 6)

MAIN GEAR DOOR UPLOCK SWITCH ADJUSTMENT (SEE VIEW E)

STEP 11

WITH THE INBOARD DOOR CLOSED AND LOCKED, THE SWITCH PLUNGER (11) SHOULD BE CAPABLE OF 0.12 (± 0.03) INCH ADDITIONAL TRAVEL BEFORE BOTTOMING. DETERMINE THE MAKE/BREAK POSITION OF THE DOOR UPLOCK SWITCH (12) AS FOLLOWS:

> FOR THE L.H. SIDE DISCONNECT WIRE G21B20 FROM TERMINAL 14 ON TERMINAL BLOCK F-3. WIRE G21B20 AND TERMINAL 2 ON TERMINAL BLOCK F-3 WILL INDICATE AN OPEN CIRCUIT FOR DOOR UP AND CLOSED CIRCUIT FOR DOOR DOWN. THIS IS THE MAKE-BREAK POSITION. FOR THE R.H. SIDE DISCONNECT WIRE G12B20 FROM TERMINAL 15 ON TERMINAL BLOCK G-3. WIRE G12B20 AND TERMINAL 4 WILL INDICATE AN OPEN CIRCUIT FOR DOOR UP AND CLOSED CIRCUIT FOR DOOR DOWN. THIS IS THE MAKE-BREAK POSITION. MAKE A REMOVABLE MARK ON THE SWITCH PLUNGER (11) AT THE MAKE/BREAK POSITION ADJACENT TO THE BARREL END AND THEN A REMOVABLE MARK 0.12 ± 0.03 TOWARDS THE END OF THE PLUNGER. ADJUST THE NUTS ON THE BARREL OF THE SWITCH (12) TO HOLD THE PLUNGER AT THE LAST MARK WITH THE DOOR UP AND HOOKED.

NOTE

TO GAIN ACCESS TO TERMINAL STRIPS F-3 AND G-3 REMOVE THE SECOND ACCESS DOOR FORWARD OF THE MAIN GEAR WHEEL WELLS ON THE BELLY FAIRING OF THE AIRPLANE (L.H. AND R.H. SIDE). THE TERMINAL STRIPS ARE ON THE OUTBOARD SIDE OF B.L.20 WEB AND AFT OF F.S.470 MAIN FRAME.

STEP 12

SAFETY THE SWITCH (12) AS REQUIRED.

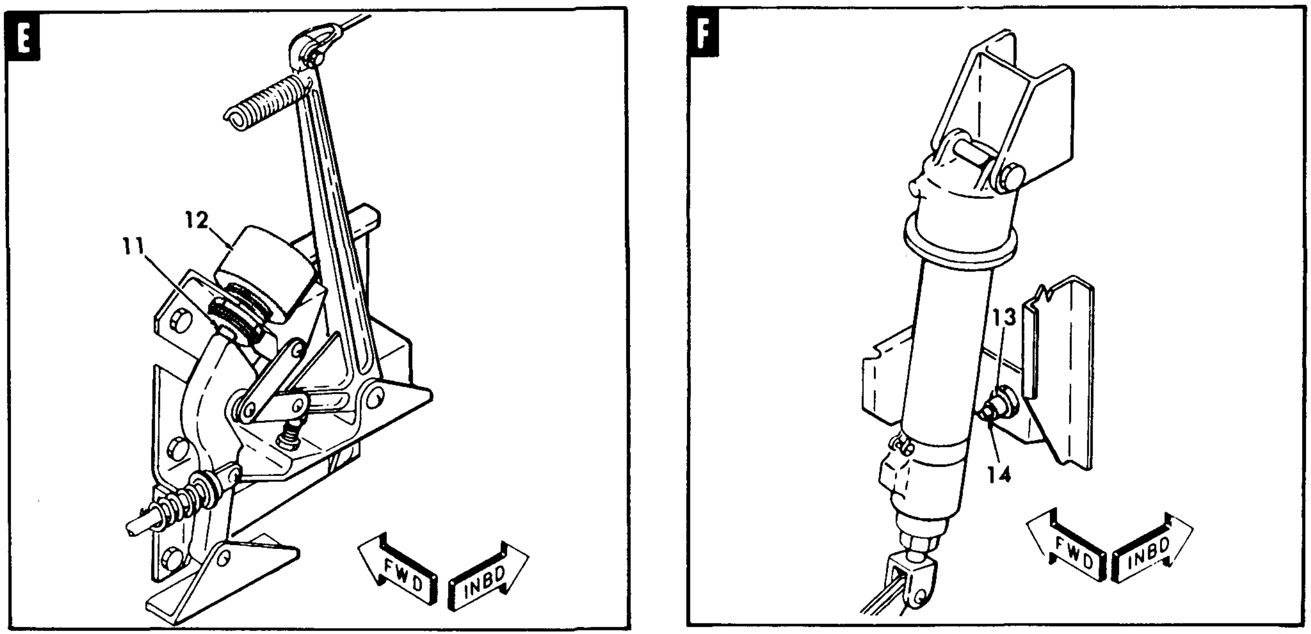
MAIN GEAR DOOR DOWN SWITCH ADJUSTMENT (SEE VIEW F)

STEP 13

WITH THE INBOARD DOOR PRESSURIZED TO THE FULLY OPEN POSITION, THE SWITCH PLUNGER (14) SHOULD BE CAPABLE OF 0.20 (±0.03) INCH ADDITIONAL TRAVEL BEFORE BOTTOMING. DETERMINE THE MAKE/BREAK POSITION OF THE DOOR DOWN SWITCH (13) AS FOLLOWS:

> FOR THE L.H. SIDE DISCONNECT WIRE G60C20 FROM TERMINAL 22 ON TERMINAL BLOCK F-3. WIRE G6020 AND TERMINAL 19 ON TERMINAL BLOCK F-3 WILL INDICATE AN OPEN CIRCUIT FOR DOOR UP AND CLOSED CIRCUIT FOR DOOR DOWN. THIS IS THE MAKE-BREAK POSITION. FOR THE R.H. SIDE DISCONNECT WIRE G60A20 FROM TERMINAL 26 ON TERMINAL BLOCK G-3. WIRE G60A20 AND TERMINAL 27 WILL INDICATE AN OPEN CIRCUIT FOR DOOR UP AND CLOSED CIRCUIT FOR DOOR DOWN. THIS IS THE MAKE/BREAK POSITION. MAKE A REMOVABLE MARK ON THE SWITCH PLUNGER (14) AT THE MAKE/BREAK POSITION ADJACENT TO THE BARREL END AND THEN A REMOVABLE MARK 0.20 ± 0.03 TOWARDS THE END OF THE PLUNGER. ADJUST THE NUTS ON THE BARREL OF THE SWITCH (13) TO HOLD THE PLUNGER AT THE LAST MARK WITH THE DOOR ALL THE WAY DOWN.

SAFETY THE SWITCH (13) AS REQUIRED.



JS-I-2-X0/2-3208-3

Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 3 of 6)

NOTE

THE MAIN GEAR DOOR UPLOCK SWITCH (12) AND MAIN GEAR DOOR DOWN SWITCH (13) CONTROL THE RED LIGHT IN THE LANDING GEAR CONTROL HANDLE IN THE CREW COMPARTMENT AND THE DOOR AND GEAR HYDRAULIC SEQUENCING VALVES. THE OVER-TRAVEL OF THE SWITCH PLUNGERS AS GIVEN ABOVE IN STEP (11) AND STEP (13) IS REQUIRED FOR THE CORRECT POSITIONING OF THE DOOR HOOK OVER ITS ROLLER AND THE LOCATION OF THE DOOR IN THE DOWN POSITION TO ASSURE CORRECT HYDRAULIC SEQUENCING.

MAIN GEAR UPLOCK SWITCH ADJUSTMENT

CAUTION

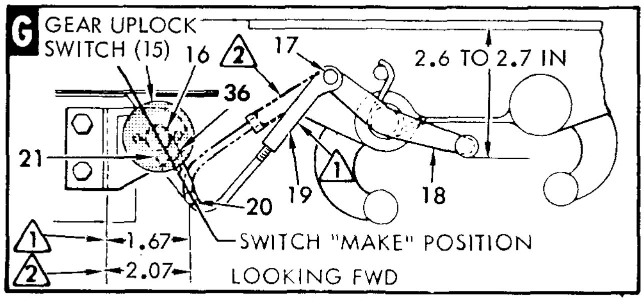
PRIOR TO SWITCH ADJUSTMENT, REMOVE THE ELECTRICAL CONNECTORS FROM THE SOLENOIDS OF THE GEAR AND DOOR SELECTOR CONTROL VALVES IN THE LEFT HAND WHEEL WELL.

STEP 14

POSITION THE BOTTOM OF LEVER (18) AT 2.6 TO 2.7 INCHES FROM THE BOTTOM OF THE TOP FLANGE OF THE UPLOCK ASSEMBLY AS SHOWN IN VIEW "G". HOLD IN THIS POSITION UNTIL THE SWITCH (15) AND PUSHROD (19) FINAL ADJUSTMENTS ARE MADE. REMOVE THE PIN (17) AND DISCONNECT THE PUSHROD (19) FROM THE LEVER (18). POSITION SWITCH LEVER ARM (20) TO THE 1.67 △1 OR 2.07 △2 APPROXIMATE DIMENSION FROM THE INBOARD FACE OF THE SWITCH BRACKET BY ADJUSTING THE ROD END OF PUSHROD (19). RECONNECT PUSHROD (19) TO LEVER (18) AND REINSTALL PIN (17).

STEP 15

POSITION THE INITIAL MAKE POSITION OF THE UPLOCK SWITCH (15) AS FOLLOWS:
LOOSEN THE LOCKNUT (16) AND THE WORM LOCK SCREW (36). TURN THE WORM ADJUSTING SCREW (21) TO POSITION THE MAKE POSITION OF THE SWITCH CAM WITH THE SWITCH LEVER ARM (20) HELD AT 1.67 INCHES △1 OR 2.07 INCHES △2 (APPROXIMATELY) AND LEVER (18) AT THE 2.6 TO 2.7 INCH POSITION.



NOTE

TO DETERMINE THE CORRECT CAM POSITION ON THE LEFT HAND UPLOCK SWITCH (15), DISCONNECT WIRE G106B20 FROM TERMINAL 16 ON TERMINAL STRIP F-3 AND CONNECT A METER TO WIRE G106B20 AND TERMINAL 2 ON TERMINAL STRIP F-3. THE CIRCUIT SHALL BE CLOSED WHEN THE SWITCH LEVER ARM (20) IS SLIGHTLY MORE THAN 1.67 INCHES △1 OR 2.07 INCHES △2. THIS IS THE INITIAL "MAKE" POSITION. TO DETERMINE THE CORRECT CAM POSITION ON THE RIGHT HAND UPLOCK SWITCH (15), DISCONNECT WIRE G10R20 FROM TERMINAL 4 ON TERMINAL STRIP G-3 AND CONNECT A METER TO WIRE G10R20 AND TERMINAL 15 ON TERMINAL STRIP G-3. THE CIRCUIT SHALL BE CLOSED WHEN THE LEVER ARM (20) IS SLIGHTLY MORE THAN 1.67 INCHES △1 OR 2.07 INCHES △2. THIS IS THE INITIAL MAKE POSITION. FOR TERMINAL STRIP LOCATIONS SEE NOTE FOR STEP 11.

STEP 16

TIGHTEN LOCKNUT (16) AND THE WORM LOCK SCREW (36) TO HOLD THE SWITCH CAM AT THE CORRECT INITIAL MAKE POSITION. SAFETY WORM LOCK SCREW WITH MS20995C32 LOCKWIRE.

STEP 17

RECHECK SWITCH ADJUSTMENT, AND MAKE SURE THAT ALL CONNECTIONS AND ADJUSTMENT POINTS ARE TIGHT AND SAFETY AS REQUIRED. RECONNECT THE ELECTRICAL CONNECTORS TO THE GEAR AND DOOR SELECTOR CONTROL VALVES IN THE LEFT HAND MAIN GEAR WHEEL WELL

MAIN GEAR DOWNLOCK SWITCH ADJUSTMENT

NOTE

THE ACTUATOR MUST BE REMOVED FROM THE AIRPLANE FOR SWITCH ADJUSTMENT. (SEE VIEW H).

STEP 18

WITH THE SWITCH (24) AND ADAPTER (23) INSTALLED PER FIGURE 32-5 AND WITH THE ACTUATOR IN THE RETRACTED AND LOCKED POSITION, ADJUST SWITCH BY ROTATING THE ADAPTER CLOCKWISE UNTIL SWITCH LEADS # 1 AND # 3 ARE ELECTRICALLY CONTINUOUS. NOTE ADAPTER POSITION.

△1 AIRPLANES 5001 THRU 5092, 5094 THRU 5096 NOT MODIFIED BY S.B. 329-230 OR S.B. 329-251.

△2 AIRPLANES 5001 THRU 5092, 5094 THRU 5096 MODIFIED BY S.B. 329-230 OR S.B. 329-251 AND AIRPLANES 5093, 5097 AND UP.

JS-1-2-X0/3-3208-4

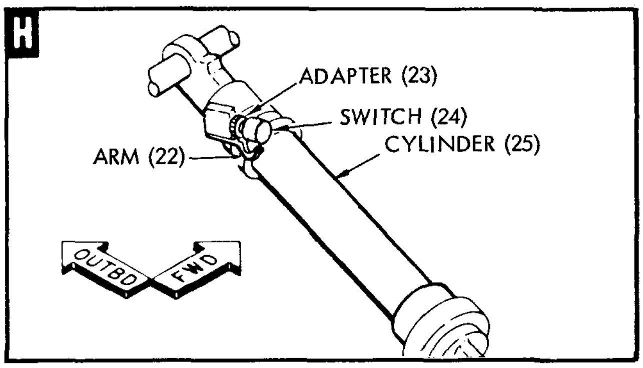
Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 4 of 6)

STEP 19

EXTEND (UNLOCK) ACTUATOR AND READJUST ADAPTER (23) COUNTER CLOCKWISE UNTIL LEADS # 1 AND # 3 ARE AGAIN ELECTRICALLY CONTINUOUS. THEN BACK OFF 1/2 THE NUMBER OF TEETH REQUIRED. IN THE EXTENDED (UNLOCKED) POSITION CHECK LEADS # 1 AND # 2 FOR ELECTRICAL CONTINUITY AND LEADS # 1 AND # 3 ARE ELECTRICALLY DISCONTINUOUS. OPERATE ACTUATOR WITH 1500 PSI MINIMUM. ACTUATE CYLINDER APPROXIMATELY 20 TIMES TO CHECK OPERATION OF SWITCH.

STEP 20

WHILE HOLDING THE ADAPTER (23) IN THE POSITION OBTAINED IN STEP 19, ADJUST THE ADAPTER LOCK-NUT TO A TORQUE OF FROM 100 TO 125 INCH-POUNDS AND LOCKWIRE THE LOCKNUT TO THE ARM (22).



STEP 21

REDUCE HYDRAULIC PRESSURE TO ZERO AND RETURN THE CYLINDER (25) TO NORMAL OPERATING CONFIGURATION.

STEP 22

POSITION SWITCH LEVER (29) TO BE IN FREE AREA OF SWITCH (31) (AREA WHERE MOVEMENT IS EASY AND UNRESTRICTED). WHILE HOLDING SWITCH IN MOUNTING POSITION ROTATE SWITCH LEVER (29) CLOCKWISE ON LEFT SIDE, COUNTERCLOCKWISE ON RIGHT SIDE UNTIL FREE MOTION IS RESTRICTED. FROM THIS POSITION ROTATE SWITCH LEVER AN ADDITIONAL 8° TO 10°. REMOVE SWITCH LEVER FROM SWITCH, TAKING CARE NOT TO CHANGE SWITCH SHAFT (34) POSITION. (SEE VIEW K).

STEP 23

ATTACH SWITCH LEVER (29) TO ROD END (30) AND ADJUST TO OBTAIN A DIMENSION OF 3.40 INCHES BETWEEN POINT "A" AND POINT "B". INSTALL ON ARM ASSY AT POINT "B" WITH PIN (32), WASHER, AND COTTER PIN.

NOTE

ACTUATOR MUST BE IN THE FREE POSITION WHEN ADJUSTING. OPERATING PRESSURE ON THE ACTUATOR WHEN ADJUSTING MAY PERMANENTLY DAMAGE THE ADJUSTING MECHANISM.

STEP 24

INSTALL SWITCH (31) IN MOUNTING HOLES ON BRACKET (35), ALIGNING SWITCH SHAFT (34) AND SWITCH LEVER (29). INSTALL MOUNTING NUT. NOTE: CARE MUST BE TAKEN TO PREVENT SWITCH FROM ROTATING BEFORE SWITCH LEVER IS ATTACHED.

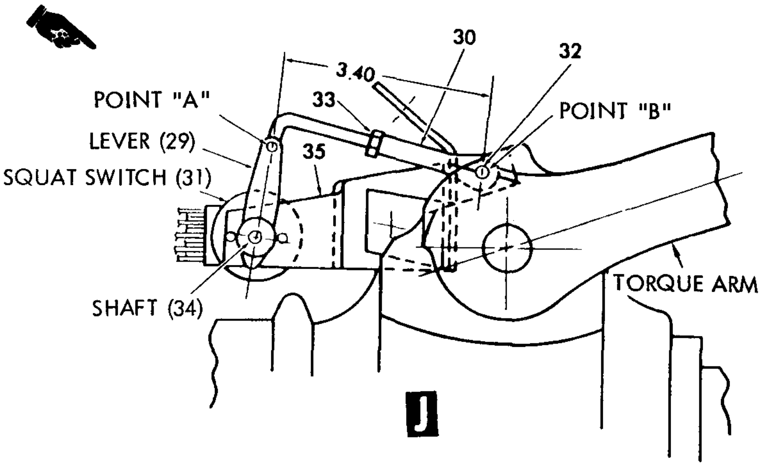
STEP 25

PERFORM CONTINUITY CHECK.

MAIN GEAR SQUAT SWITCH ADJUSTMENT (AIRPLANES 5046 AND UP AND AIRPLANES MODIFIED BY SERVICE BULLETIN 329-72) (SEE VIEW J)

NOTE

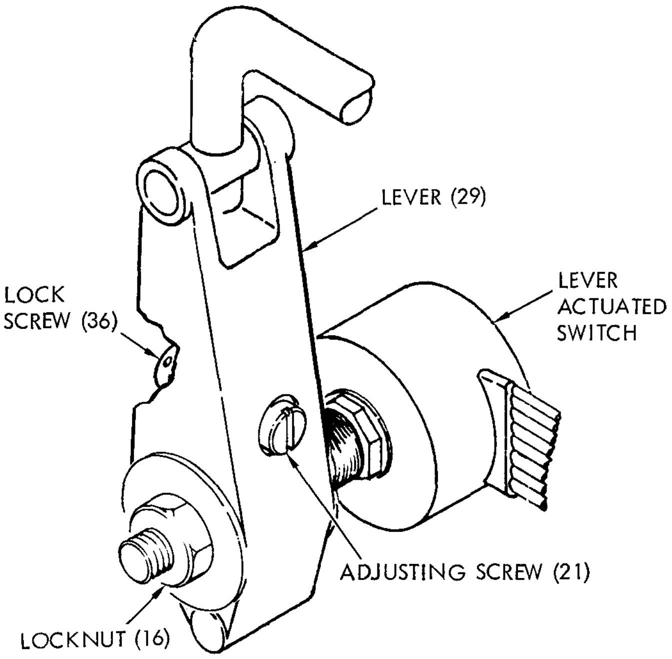
WITH THE MAIN GEAR STRUT FULLY EXTENDED (AIRPLANE ON JACKS) LEADS NO. 1 AND NO. 2 SHOULD BE ELECTRICALLY CONTINUOUS. WITH THE STRUT COMPRESSED 2.0 TO 2.5 INCHES FROM THE FULLY EXTENDED POSITION, LEADS NO. 1 AND NO. 3 SHOULD BE ELECTRICALLY CONTINUOUS. IF THIS CONDITION DOES NOT EXIST ADJUST SWITCH PER STEPS 22 THROUGH 25.



STRUT SHOWN IN EXTENDED POSITION

JS-1-2-X0/3-3208-5

Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 5 of 6)



K

ADJUSTMENT INSTRUCTIONS FOR LEVER ACTUATOR (SEE VIEW K)

ADJUST LEVER SERRATIONS TO APPROXIMATE OPERATING POSITION BEFORE ASSEMBLING. DO NOT FORCE LEVER ALL THE WAY ON TO SWITCH DURING THIS PRE-ASSEMBLING OPERATION. TAPERED SERRATIONS ON LEVER AND SWITCH SHAFT MAKE REMOVAL EXTREMELY DIFFICULT.

TO ASSEMBLE: WHEN APPROXIMATE OPERATING POSITION HAS BEEN DETERMINED, PLACE LEVER ON SHAFT, ENGAGING SERRATIONS. TIGHTEN HEXAGON LOCKING NUT MAKING SURE WASHER HAS BEEN INCLUDED.

TO ADJUST: LEVER MAY BE ADJUSTED TO AN INFINITE NUMBER OF POSITIONS THROUGH 360°. ADJUSTMENT IS MADE BY LOOSENING LOCKSCREW (36) ON LEVER APPROXIMATELY ONE TURN AND TURNING ADJUSTING SCREW (21) UNTIL LEVER IS IN DESIRED POSITION. LOCKSCREW IS THEN TIGHTENED TO SECURE LEVER IN POSITION. A 0.037 INCH DIAMETER HOLE IS LOCATED IN LEVER LOCKSCREW TO PROVIDE A POSITIVE WIRE LOCKING DEVICE.

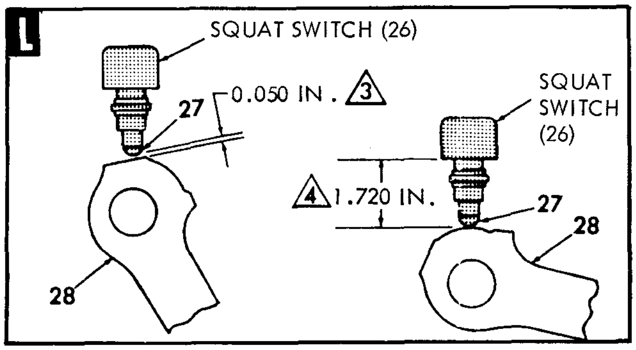
MAIN GEAR SQUAT SWITCH ADJUSTMENT (AIRPLANES 5001 THRU 5045 NOT MODIFIED BY SERVICE BULLETIN 329-72) (SEE VIEW L)

STEP 26

WITH THE MAIN GEAR STRUT FULLY EXTENDED (AIRPLANE ON JACKS), THE SWITCH (26) AND ROLLER (27) SHOULD BE 0.050 INCH FROM THE CONTACT POINT ON THE SCISSORS (28) AS SHOWN IN VIEW "L". WITH THE WHEELS ON THE GROUND AND THE STRUT COMPRESSED NORMALLY, THIS ADJUSTMENT WILL RESULT IN A DIMENSION OF 1.720 INCH FROM THE BOTTOM OF THE SWITCH HOUSING TO THE BOTTOM OF THE SWITCH ROLLER AS SHOWN IN VIEW "L".

STEP 27

SAFETY THE SWITCH (26) AS REQUIRED.



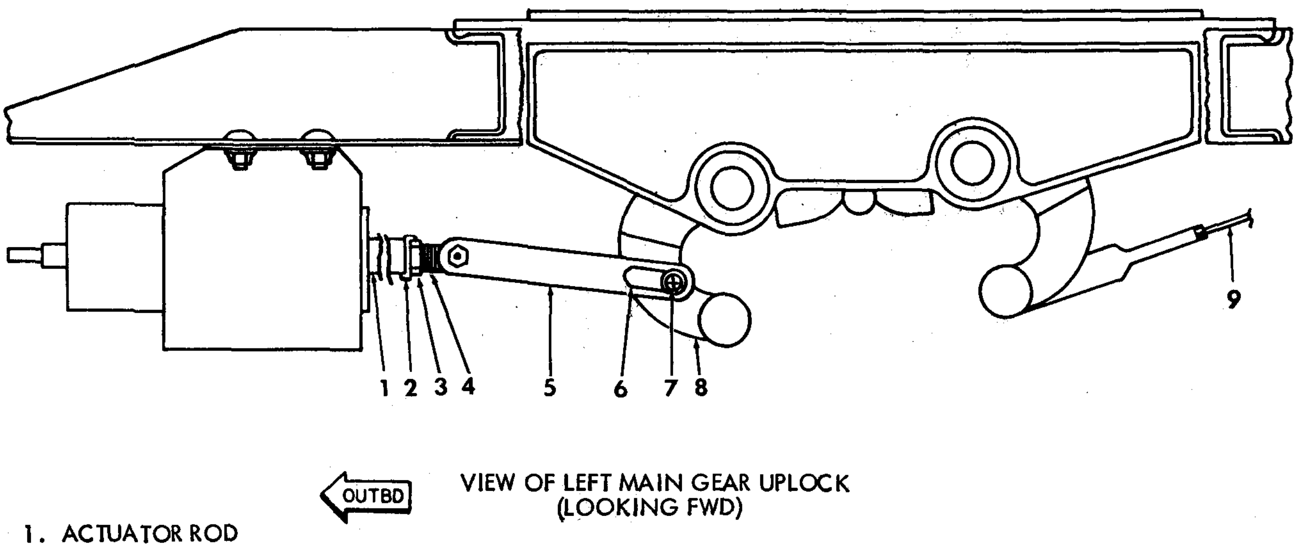
NOTE

△3 SWITCH DIMENSION WITH STRUT FULLY EXTENDED.

△4 SWITCH DIMENSION WITH STRUT COMPRESSED NORMALLY.

JS-1-2-X0/3-3208-6

Figure 32-8. Landing Gear and Door Switch Adjustment (Sheet 6 of 6)



1. ACTUATOR ROD
2. TAB WASHER
3. JAM NUT
4. ADAPTER
5. LINK △1
6. LINK SLOT △1
7. SCREW
8. UPLOCK CLAW
9. EMERGENCY GEAR RELEASE CABLE

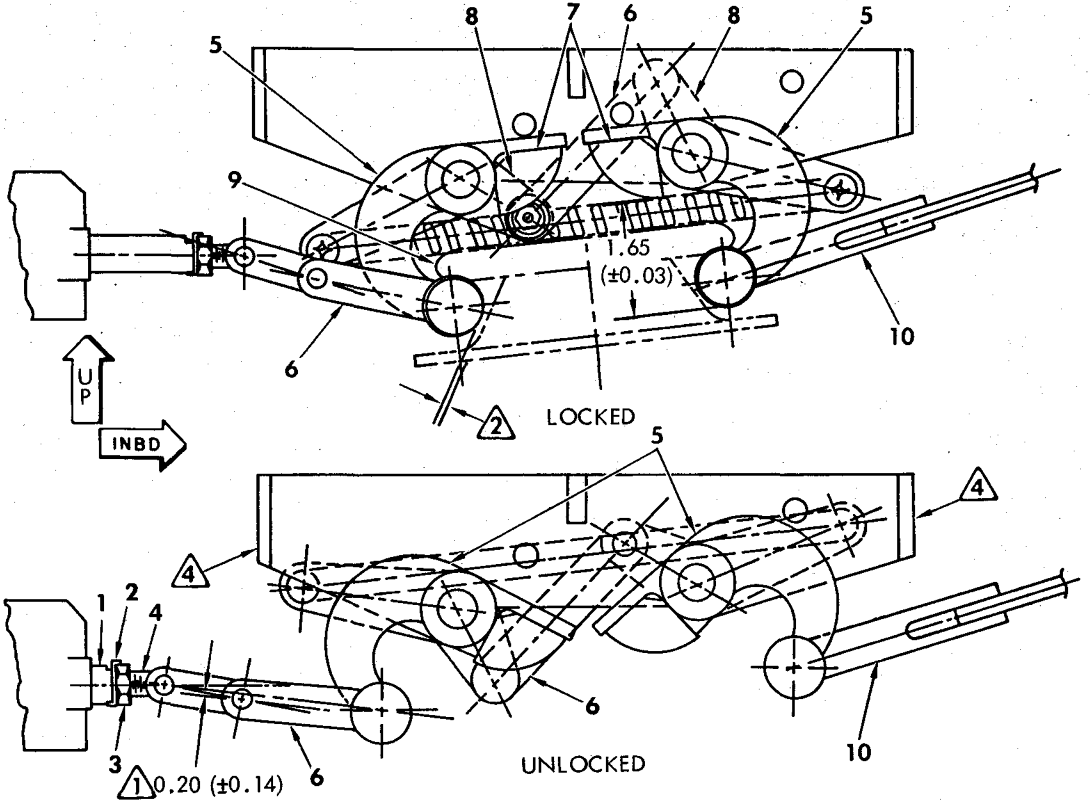
NOTE

△1 ONE LINK SHOWN; OTHER LINK DIRECTLY BEHIND IT.

2. SWITCH AND LINKAGE OMITTED FOR CLARITY.

JS-1-2-X0/2-3209

Figure 32-9. Main Gear Uplock Cylinder Linkage Adjustment (Airplanes 5001 thru 5055)



1. ACTUATOR ROD
2. TAB WASHER
3. JAM NUT
4. ADAPTER
5. UPLOCK CLAW
6. LINK ASSEMBLY
7. WASHERS (AS REQ'D)
8. LEVER
9. MAIN LANDING GEAR SPINNER (GEAR UP POSITION)
10. EMERGENCY RELEASE

NOTE

△1 DIMENSION TO BE MEASURED WITH UPLOCK CLAWS OPEN AND ACTUATOR ROD FULLY RETRACTED.

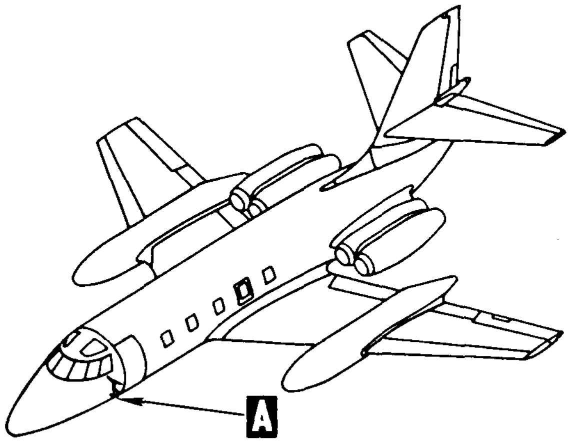
△2 THIS DIMENSION, ON BOTH SIDES OF THE SPINNER, SHALL BE EQUAL (WITHIN 0.06 INCHES) TO ENSURE EQUAL ENGAGEMENT OF ROLLERS WITH SPINNER.

3. TO PROVIDE CORRECT MLG UPLOCK ENGAGEMENT, MAINTAIN THE 1.65 (±0.03) DIMENSION. SHIM AS REQUIRED WITH AN960-8 WASHERS. SHIM EACH FASTENER EQUALLY. 0.08 MAXIMUM MAY BE TRIMMED FROM WASHERS TO CLEAR MATING RADII.

△4 CHECK ADJUSTER PLATES AND ATTACHMENTS FOR WEAR AND TIGHTNESS.

JS-1-2-X0/6-3209A

Figure 32-9A. Main Gear Uplock Cylinder Linkage Adjustment (Airplanes 5056 and Up)



NOTE

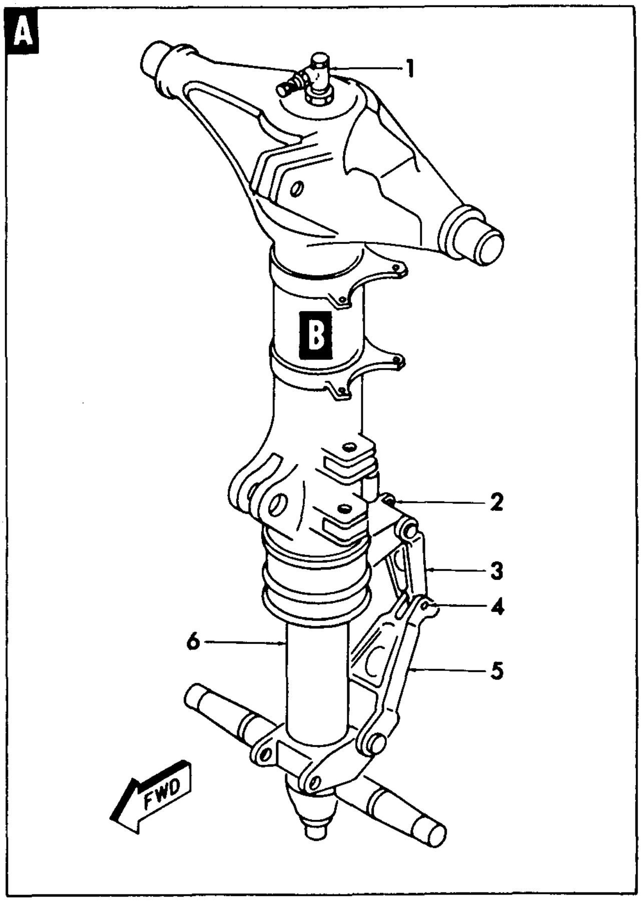
REFER TO PARAGRAPH 29.0.1.9 FOR INFORMATION ON HYDRAULIC CYLINDER SEALS.

STEP 1

JACK THE AIRPLANE WITH FUSELAGE AND WING JACKS UNTIL THE NOSE STRUT STARTS TO EXTEND.

STEP 2

RELIEVE ALL OF THE AIR PRESSURE IN THE STRUT THROUGH THE AIR FILLER VALVE (1).



STEP 3

DISCONNECT THE NOSE STEERING TORQUE ARMS (3) AND (5) BY REMOVING THE DISCONNECT PIN (4). INSTALL THE PIN THROUGH THE HOLE (2) AND THE TORQUE ARM (3) TO HOLD THE UPPER TORQUE ARM IN THE SERVICING POSITION.

STEP 4

CUT THE SAFETY-WIRE AND REMOVE THE GLAND NUT LOCK (12) BY REMOVING THE SCREW (13).

STEP 5

LOOSEN THE GLAND NUT (27) WITH A SPANNER WRENCH AND SLIDE THE NUT DOWN THE STRUT.

NOTE

IT IS NOT NECESSARY TO LOOSEN THE STEERING COLLAR NUT (53) TO REPACK THE NOSE GEAR STRUT

STEP 6

JACK THE AIRPLANE UNTIL THE NOSE WHEELS CLEAR THE GROUND, REMOVE THE GROUND SAFETY PIN, THEN MANUALLY UNLOCK THE NOSE GEAR DOWN LOCK AND PARTIALLY RETRACT THE NOSE GEAR.

STEP 7

PLACE A CONTAINER UNDER THE STRUT TO CATCH SPILLED FLUID, AND THEN PULL THE STRUT PISTON STRAIGHT OUT OF THE CYLINDER. THE PISTON SHOULD CLEAR THE CYLINDER AFTER APPROXIMATELY 13 INCHES OF TRAVEL.

NOTE

IT MAY REQUIRE CONSIDERABLE FORCE TO OVERCOME THE FRICTION OF THE SEALS AND RINGS WHEN PULLING THE PISTON (6) OUT.

STEP 8

REMOVE THE SPACER (22), SLIDE THE BEARING (21) DOWN, AND REMOVE THE UPPER RING (20); THEN REMOVE THE KEYS (24), UPPER CAM (40) AND LOWER RING (23).

STEP 9

REMOVE THE KEYS (25) AND THE LOWER CAM (39).

STEP 10

REMOVE THE UPPER RING (38), GASKET (37), AND LOWER RING (36) FROM THE BEARING (35), THEN REMOVE THE BEARING (35), RING (34), PACKING (33), AND LOWER RING (32).

STEP 11

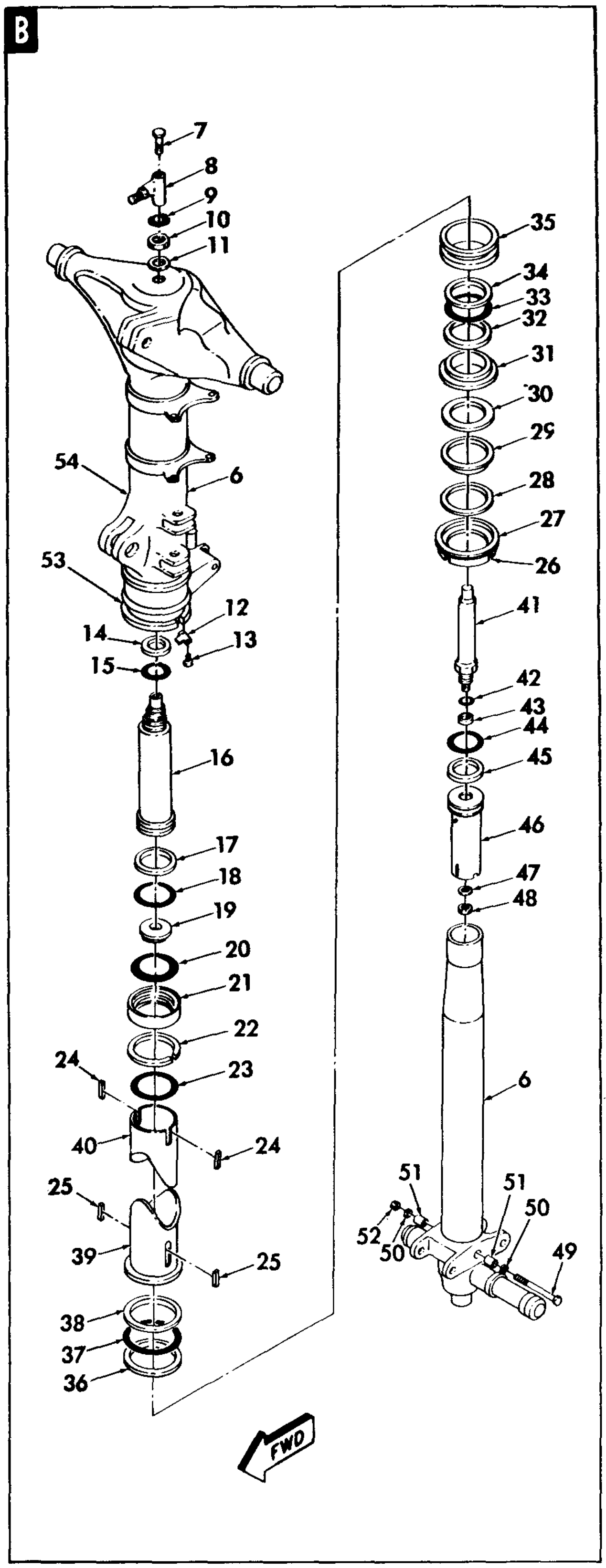
REMOVE THE ADAPTER (31), FELT (30), AND ADAPTER (29), AND THEN REMOVE THE SCRAPER (28).

STEP 12

REMOVE BOLT (7), VALVE (8), O-RING (9), NUT (10), WASHER (11) AND REMOVE ITEMS (14) THROUGH (19) AS AN ASSEMBLY FROM CYLINDER ASSEMBLY (54).

JS-1-2-X0/3-3210-1

Figure 32-10. Nose Gear Strut Packing Instructions (Sheet 1 of 3)



STEP 13

REMOVE RING (14) AND PACKING (15) FROM TUBE (16).

STEP 14

REMOVE ORIFICE (19), PACKING (18) AND RING (17) FROM TUBE (16).

STEP 15

REMOVE NUT (52), BOLT (49), WASHER (50) AND SPACERS (51) FROM STRUT PISTON (6).

STEP 16

REMOVE NUT (48), WASHER (47) AND DISASSEMBLE ITEMS (41) THROUGH (45) FROM TUBE (46).

STEP 17

CLEAN AND INSPECT ALL PARTS FOR WEAR AND DAMAGE. VISUALLY INSPECT THE EXTERNAL AND INTERNAL SURFACES OF THE SHOCK STRUT. PENETRANT INSPECT (ZYGLO) ANY SUSPECTED AREAS. REPLACE PARTS WHERE NECESSARY.

STEP 18

REPLACE ALL SEALS; REPLACE OTHER COMPONENTS IF WORN.

STEP 19

LUBRICATE THE PISTON ASSEMBLY WITH MIL-H-5606 HYDRAULIC FLUID, AND INSTALL THE COMPONENTS BY REVERSING STEPS 16 THROUGH 8.

STEP 20

INSTALL THE PISTON ASSEMBLY KEEPING IT CENTERED SO THAT THE RING (18) WILL COMPRESS NORMALLY.

STEP 21

SLIDE THE NUT (27) UP AND SCREW IT PARTIALLY INTO THE CYLINDER.

STEP 22

MANUALLY POSITION THE NOSE GEAR IN THE DOWN-AND-LOCKED POSITION, AND REINSTALL THE GROUND LOCK SAFETY PIN.

STEP 23

LOWER THE JACKS UNTIL THE NOSE STRUT STARTS TO COMPRESS, THEN TIGHTEN THE GLAND NUT (27) UNTIL IT BOTTOMS.

STEP 24

BACK THE GLAND NUT (27) OFF UNTIL THE FIRST LOCKING CUTOUT (26) LINES WITH THE STEERING COLLAR NUT CUTOUT (53).

STEP 25

INSTALL THE GLAND NUT LOCK (12) BY INSTALLING AND SAFETYING THE SCREW (13).

JS-1-2-X0/2-3210-2

Figure 32-10. Nose Gear Strut Packing Instructions (Sheet 2 of 3)

STEP 26

SLOWLY LOWER THE AIRPLANE FROM THE JACKS UNTIL THE NOSE STRUT BOTTOMS.

CAUTION

DO NOT ALLOW THE STRUT PISTON (6) TO BIND IN THE CYLINDER. BINDING MAY CAUSE THE PISTON TO BECOME SCORED.

STEP 27

SERVICE THE SHOCK STRUT ACCORDING TO THE INSTRUCTIONS IN CHAPTER 12.

STEP 28

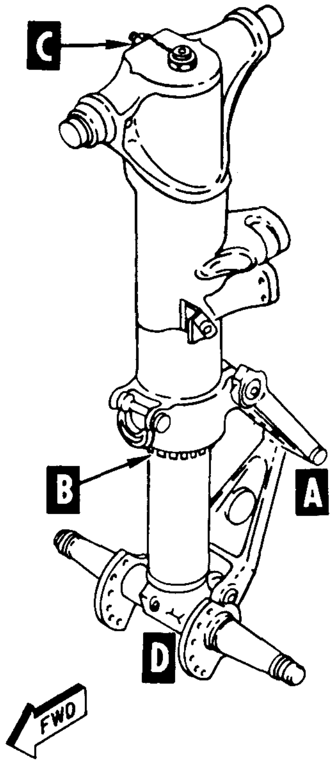
CLEAN THE NOSE GEAR AREA, AND RETURN THE NOSE GEAR TO NORMAL OPERATING CONFIGURATION.

NOTE

SCISSORS DISCONNECT PIN (4) SHOULD BE INSTALLED ON RH SIDE AS SHOWN IN VIEW B FIGURE 9-1.

JS-1-2-X0/8-3210-3

Figure 32-10. Nose Gear Strut Packing Instructions (Sheet 3 of 3)



CAUTION

STRUT GLAND NUTS ARE NOT ADJUSTABLE. USE THE FOLLOWING PROCEDURE TO STOP FLUID LEAKAGE AROUND THE STRUT PISTON.

CAUTION

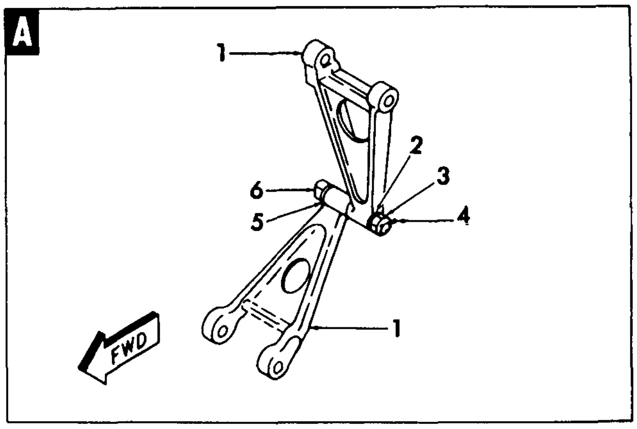
ASSEMBLIES MUST BE KEPT CLEAN AND FREE OF ALL FOREIGN MATTER DURING THIS PROCEDURE.

NOTE

REFER TO PARAGRAPH 29.0.1.9 FOR INFORMATION ON HYDRAULIC CYLINDER SEALS.

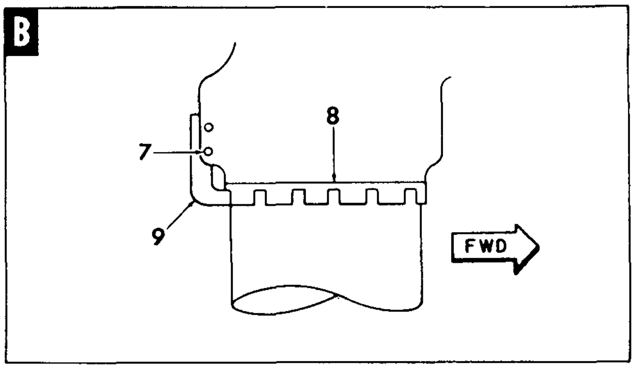
STEP 1

REMOVE THE COTTER PIN (4), NUT (3), AND WASHER (2); THEN REMOVE THE BOLT (6), AND WASHER (5), AND DISCONNECT THE SCISSORS ARMS (1).



STEP 2

REMOVE THE COTTER PIN (7) FROM THE GLAND NUT LOCKING LATCH (9) AND SWING THE LATCH FREE OF THE GLAND NUT (8).

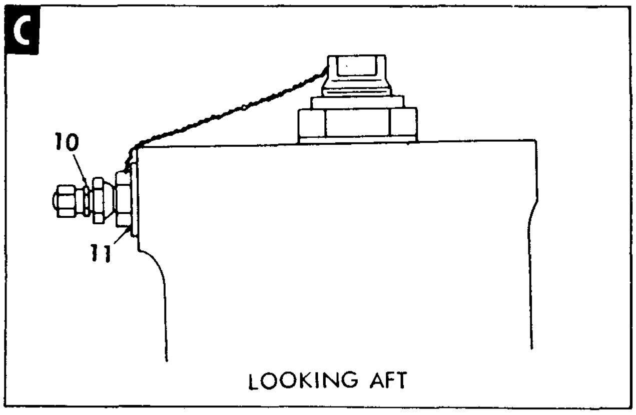


STEP 3

RELIEVE THE AIR PRESSURE IN THE STRUT ASSEMBLY THROUGH THE AIR SERVICING VALVE (10). (SEE CHAPTER 12 FOR STRUT DEPRESSURIZATION PROCEDURES).

NOTE

STAND THE STRUT ASSEMBLY IN THE GEAR DOWN POSITION TO RELIEVE THE AIR PRESSURE.



JS-1-2-X0/3-3211-1

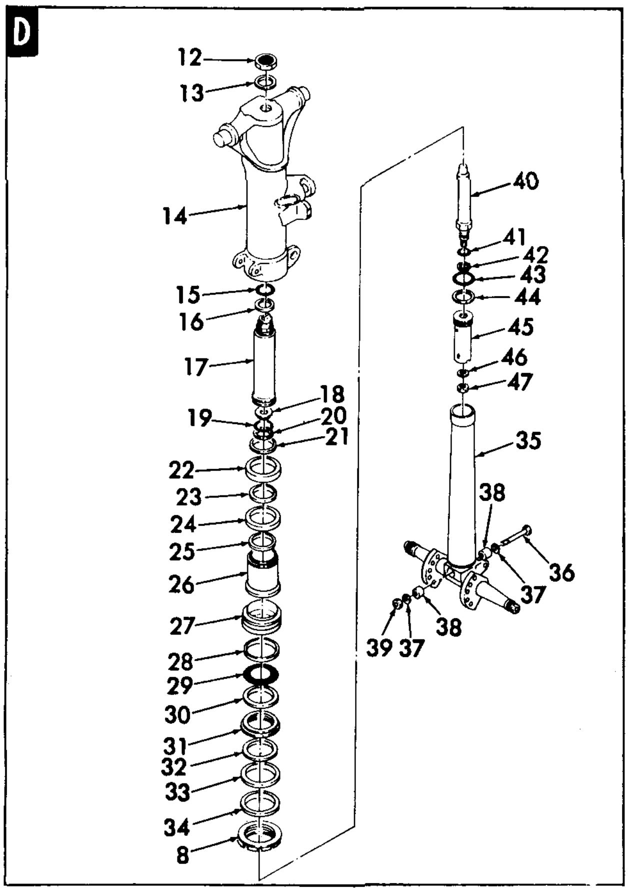
Figure 32-11. Main Gear Strut Packing Instructions (Sheet 1 of 3)

STEP 4

REMOVE THE SERVICING VALVE (10) AND DRAIN THE HYDRAULIC FLUID FROM THE STRUT. LEAVE THE SERVICING VALVE PORT (11) OPEN.

STEP 5

LOOSEN THE GLAND NUT (8) AND SLIDE IT DOWN THE STRUT PISTON (35).



STEP 6

PULL THE PISTON (35) OUT OF THE CYLINDER ASSEMBLY (14). ITEMS (8), AND (21) THROUGH (34) WILL SLIDE OUT WITH THE PISTON.

NOTE

BEFORE REMOVING PISTON, DISCONNECT THE ANTI-SKID WIRING. ON AIRPLANES 5001 THRU 5065, CUT THE WIRES AT THE EXISTING WIRE SPLICE. ON AIRPLANES 5066 AND UP, DISCONNECT THE WIRING FROM THE CANNON PLUG AT THE TOP OF THE STRUT BY PUSHING THE PINS OUT OF THE RUBBER INSERT. REINSTALL AFTER THE STRUT HAS BEEN REPACKED. AFTER REASSEMBLY OF STRUT, RUN AN OPERATIONAL CHECK ON THE ANTI-SKID SYSTEM ON AIRPLANES 5001 AND UP.

STEP 7

REMOVE THE RING (23), SLIDE THE BEARING (22) DOWN AND REMOVE THE UPPER RING (21).

STEP 8

SLIDE THE BEARINGS (22) AND (24) OFF THE PISTON AND REMOVE THE LOWER RING (25).

STEP 9

SLIDE THE SPACER (26), BEARING (27), RING (28), AND PACKING (29) OFF THE PISTON.

NOTE

IT IS NOT NECESSARY TO REMOVE THE RING (30), ADAPTER (31), FELT (32), ADAPTER (33), SCRAPER (34), OR NUT (8) FROM THE STRUT PISTON TO REPACK THE MAIN GEAR STRUT.

STEP 10

REMOVE NUT (12), WASHER (13) AND REMOVE TUBE (17) FROM CYLINDER ASSEMBLY (14).

STEP 11

REMOVE PACKING (15) AND RING (16) FROM TUBE (17).

STEP 12

REMOVE RING (21), RING (20), PACKING (19) AND ORIFICE (18) FROM TUBE (17).

STEP 13

REMOVE NUT (39), BOLT (36), WASHERS (37) AND SPACERS (38) FROM STRUT PISTON ASSEMBLY (35).

STEP 14

REMOVE ITEMS (40) THROUGH (47) FROM STRUT PISTON (35) AS ASSEMBLY.

STEP 15

REMOVE NUT (47) AND WASHER (46) FROM PIN (40) AND REMOVE PIN (40) FROM TUBE (45).

STEP 16

REMOVE RING (44), GASKET (43), RING (42) AND PACKING (41) FROM PIN (40).

STEP 17

CLEAN AND INSPECT ALL PARTS FOR WEAR AND DAMAGE. VISUALLY INSPECT THE EXTERNAL AND INTERNAL SURFACES OF THE SHOCK STRUT. PENETRANT INSPECT (ZYGLO) ANY SUSPECTED AREAS. REPLACE PARTS WHERE NECESSARY.

STEP 18

DIP NEW PACKINGS (41), (29), (19) AND (15) INTO CLEAN MIL-H-5606 HYDRAULIC FLUID AND REASSEMBLE THE REVERSE OF STEPS 11, 12 AND 16.

STEP 19

REVERSE STEPS 16 THROUGH 7 BEING SURE ALL PARTS ARE CLEAN AND LUBRICATED LIGHTLY WITH MIL-H-5606 HYDRAULIC FLUID.

JS-1-2-X0/2-3211-2

Figure 32-11. Main Gear Strut Packing Instructions (Sheet 2 of 3)

STEP 20

SLIDE THE PISTON ASSEMBLY (35) INTO THE CYLINDER ASSEMBLY (14) BEING CAREFUL NOT TO DAMAGE THE COMPONENTS. KEEP THE PISTON ASSEMBLY CENTERED SO AS NOT TO DAMAGE THE RING (20) INSIDE THE CYLINDER ASSEMBLY.

STEP 21

INSTALL AND TIGHTEN THE GLAND NUT (8) UNTIL IT BOTTOMS, THEN BACK IT OFF UNTIL THE FIRST CASTELLATION OF THE NUT LINES UP WITH THE LOCKING LATCH (9).

STEP 22

SWING THE LATCH (9) INTO PLACE AND SAFETY IT WITH A COTTER PIN.

STEP 23

INSTALL THE SERVICING VALVE (10).

STEP 24

REVERSE STEP (1)

STEP 25

SERVICE THE STRUT ACCORDING TO THE INSTRUCTIONS IN CHAPTER 12.

STEP 26

LUBRICATE THE STRUT ACCORDING TO THE INSTRUCTIONS IN CHAPTER 12.

JS-1-2-XO/2-3211-3

Figure 32-11. Main Gear Strut Packing Instructions (Sheet 3 of 3)

STRAIGHT BUSHINGS JL1398	FLANGED BUSHINGS JL1399	FLANGED BUSHINGS JL1624	INSIDE DIAMETER AFTER INSTALLATION AND △1 OR △2 INCHES	MAXIMUM ALLOWABLE INSIDE DIAMETER INCHES	ASSEMBLY USED ON (PART NO.)
-1			.3125/.3151	.3157	JL1814-1
-3			.3125/.3151	.3157	JL1813-1
-7			.3753/.3778	.3788	JL1801-1
-11			.3742/.3767	.3777	JL1801-1
-13			.6246/.6273	.6286	JL1911-1;JL1902-1
-15			.6253/.6280	.6294	JL1801-1
-17			.7500/.7527	.7541	JL1802-1
-25			.4378/.4405	.4415	JL1801-1
-31			.3753/.3778	.3788	JL1801-1
-33			.7500/.7527	.7541	JL1959-1
-35			.3750/.3775	.3785	JL1836-1
-41			.6250/.6277	.6290	JL1901-1
-43			.2496/.2504	.2515	JL1945-1
-45			.3750/.3758	.3764	JL1945-1
-47 △1			.5017/.5025	.5031	JL1860-1
-49			.3750/.3758	.3764	JL1945-1
	-5		1.2503/1.2532	1.2555	JL1158-1
	-9		.6253/.6280	.6294	JL1811-1
	-11		.6249/.6276	.6289	JL1101-1L; JL1101-1R
	-13		.6249/.6276	.6290	JL1110-1;JL1167-1
	-15		.7503/.7530	.7544	JL1969-1
	-19		.7493/.7520	.7534	JL1601-1
	-25		.7503/.7530	.7544	JL1801-1
	-27		1.2503/1.2532	1.2555	JL1102-1
	-29		1.1250/1.1280	1.1303	JL1954-1L; JL1954-1R JL1916-1' JL1917-1
	-31		1.2503/1.2532	1.2555	JL1110-1; JL1167-1
	-33		1.2503/1.2532	1.2555	JL1101-1L; JL1101-1R
	-35		1.2503/1.2532	1.2555	JL1101-1L; JL1101-1R
	-37		1.8753/1.8783	1.8811	JL1954-1L; JL1954-1R JL1916-1; JL1917-1
	-39		.6249/.6276	.6289	JL1601-1
	-43		.4378/.4405	.4415	JL1801-1
	-45		.4378/.4405	.4415	JL1801-1
	-47		.4395/.4422	.4432	JL1839-1
	-55		.6249/.6276	.6290	JL1110-1; JL1167-1
	-57		.6253/.6280	.6293	JL1803-1
	-59		.6253/.6280	.6293	JL1803-1
	-61		.6253/.6280	.6293	JL1803-1
	-63		.3114/.3139	.3149	JL1610-1
	-65		.3114/.3139	.3149	JL1610-1
	-67		.7492/.7497	.7504	JL1605-1; JL1610-1
	-69 △1		.7492/.7497	.7504	JL1602-1
		-1 △2	.7492/.7497	.7504	JL1626-1
		-2	.7492/.7497	.7504	JL1605-5; JL1628-1

NOTE

△1 REAM AFTER INSTALLATION TO OBTAIN INSIDE DIA.

△2 HANDLAP TO OBTAIN INSIDE DIA. AFTER INSTALLATION.

Figure 32-12. Landing Gear Bushing Wear

STEP 1

INSTALL THE TIRE ON THE NOSE WHEEL IN THE NORMAL MANNER, MAKING SURE THAT THE CHINE (IF CHINE TIRE IS USED) IS ON THE OUTBOARD SIDE OF THE WHEEL.

STEP 2

INFLATE THE TIRE TO 25 TO 50 PSI.

STEP 3

LOCATE A DYNAMIC WHEEL-BALANCING MACHINE, AND MODIFY THE MACHINE TO ACCEPT A PULLY APPROXIMATELY 3 INCHES IN DIAMETER ON THE WHEEL DRIVE SHAFT.

NOTE

A BEAR AUTOMOBILE DYNAMIC TIRE-BALANCING MACHINE IS RECOMMENDED AS A CONVENIENT METHOD OF NOSE WHEEL TIRE BALANCING.

STEP 4

PLACE A PULLEY BELT AROUND THE DRIVE WHEEL AND THE 3 - INCH PULLEY AS SHOWN BELOW. THIS MODIFICATION IS NECESSARY TO PROVIDE A WHEEL SPEED OF 3,000 RPM, WHICH IS ROUGHLY EQUAL TO 140 KNOTS.

NOTE

SINCE THE ALIGNMENT MACHINE IS CAPABLE OF ONLY ABOUT 100 MPH BY USE OF THE FRICTION DRIVE, THE USE OF THE BELT DRIVE WILL GIVE THE NECESSARY WHEEL SPEED FOR PROPER BALANCING.

STEP 5

MANUFACTURE ENOUGH OF THE WHEEL-BALANCING WEIGHTS AS SHOWN IN DETAIL.

STEP 6

BALANCE THE WHEEL STATICALLY AND DYNAMICALLY BY ADDING THE WEIGHTS AS NECESSARY TO THE WHEEL BOLTS.

NOTE

DO NOT REMOVE MORE THAN ONE BOLT AT A TIME. RETORQUE EACH BOLT TO 100 TO 120 INCH-POUNDS AFTER INSTALLING THE WEIGHTS.

STEP 7

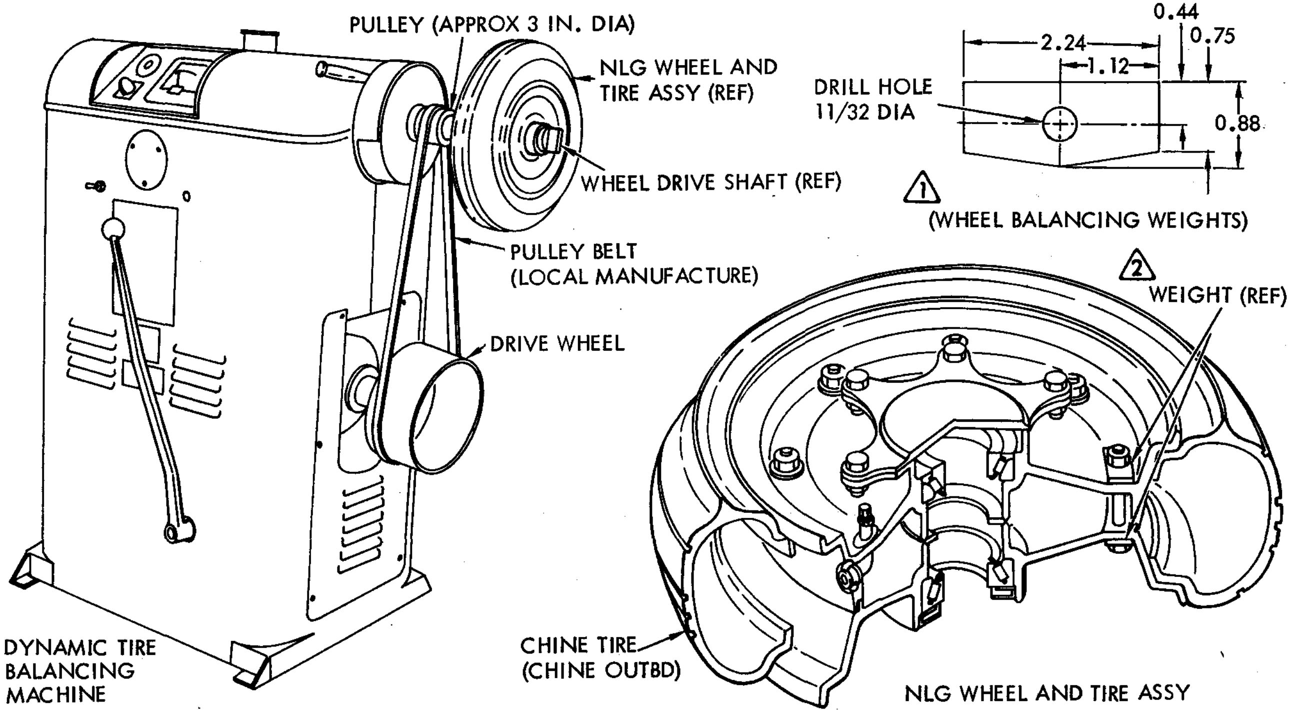
IT IS PERMISSIBLE TO REMOVE THE WASHER AND SUBSTITUTE A WEIGHT EITHER UNDER THE HEAD OR UNDER THE NUT. HOWEVER, IF MORE WEIGHTS ARE NEEDED, USE A LONGER BOLT (PART NO. MS20005-34).

STEP 8

BALANCE THE WHEEL AND TIRE ASSEMBLY AT SPEEDS UP TO 3000 RPM.

STEP 9

REMOVE THE WHEEL AND INFLATE THE CHINE TIRE TO 180 ± 5 PSI (5001 THRU 5092, 5094 THRU 5096) 220 ± 5 PSI (5093, 5097 AND UP) IN ACCORDANCE WITH THE INSTRUCTIONS IN CHAPTER 12.


PULLEY (APPROX 3 IN. DIA)
NLG WHEEL AND TIRE ASSY (REF)
DRILL HOLE 11/32 DIA
0.44
2.24
1.12
0.75
0.88
WHEEL DRIVE SHAFT (REF)
1
(WHEEL BALANCING WEIGHTS)
2
WEIGHT (REF)
PULLEY BELT (LOCAL MANUFACTURE)
DRIVE WHEEL
DYNAMIC TIRE BALANCING MACHINE
CHINE TIRE (CHINE OUTBD)
NLG WHEEL AND TIRE ASSY

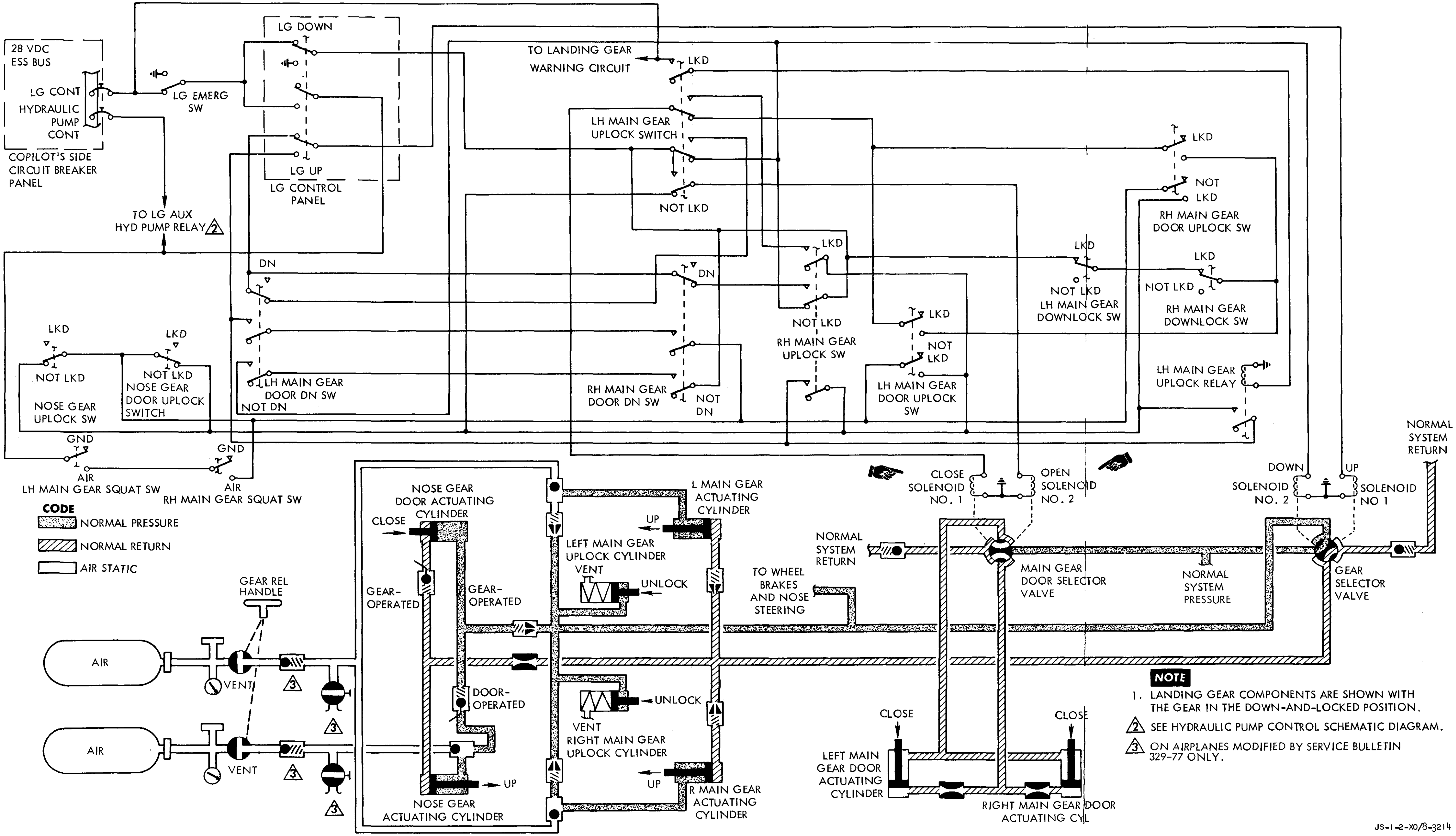
NOTE

△1 MAKE FROM ANNEALED OR HARD STAINLESS STEEL 0.063 IN. THICK. BREAK ALL CORNERS 1/32 TO 1/16.

△2 REMOVE WASHERS AND INSTALL WEIGHTS (ONE BOLT AT A TIME.) IF MORE WEIGHTS ARE REQUIRED USE LONGER BOLT (MS20005-34).

JS-32-0-013

Figure 32-13. Nose Landing Gear Wheel and Tire Balancing Instructions



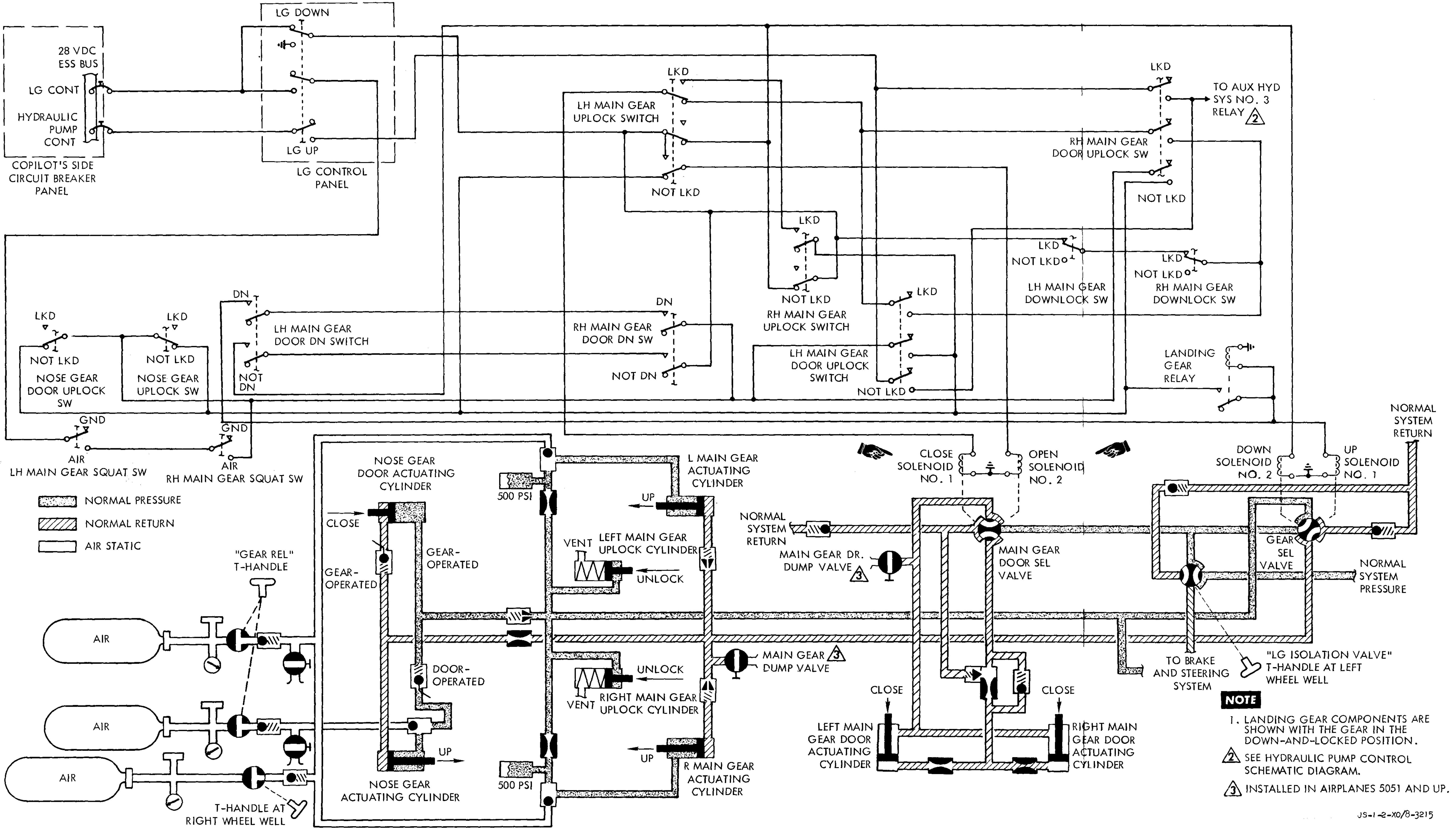
NOTE

1. LANDING GEAR COMPONENTS ARE SHOWN WITH THE GEAR IN THE DOWN-AND-LOCKED POSITION.

2 SEE HYDRAULIC PUMP CONTROL SCHEMATIC DIAGRAM.

3 ON AIRPLANES MODIFIED BY SERVICE BULLETIN 329-77 ONLY.

Figure 32-14. Landing Gear Control Schematic Diagram (Airplanes 5001 thru 5040)



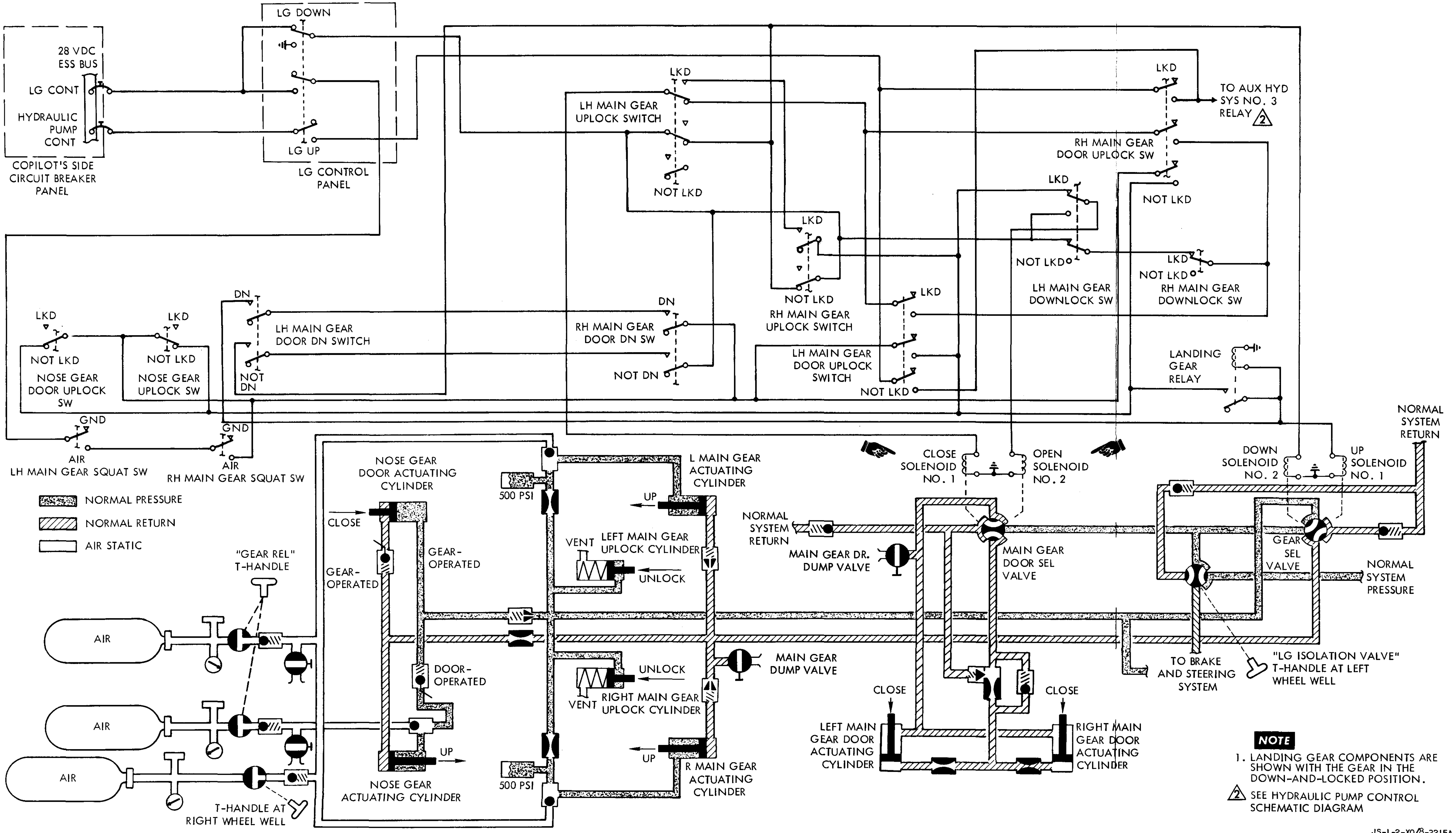
NOTE

1. LANDING GEAR COMPONENTS ARE SHOWN WITH THE GEAR IN THE DOWN-AND-LOCKED POSITION.

△2 SEE HYDRAULIC PUMP CONTROL SCHEMATIC DIAGRAM.

△3 INSTALLED IN AIRPLANES 5051 AND UP.

Figure 32-15. Landing Gear Control Schematic Diagram (Airplanes 5041 thru 5055)

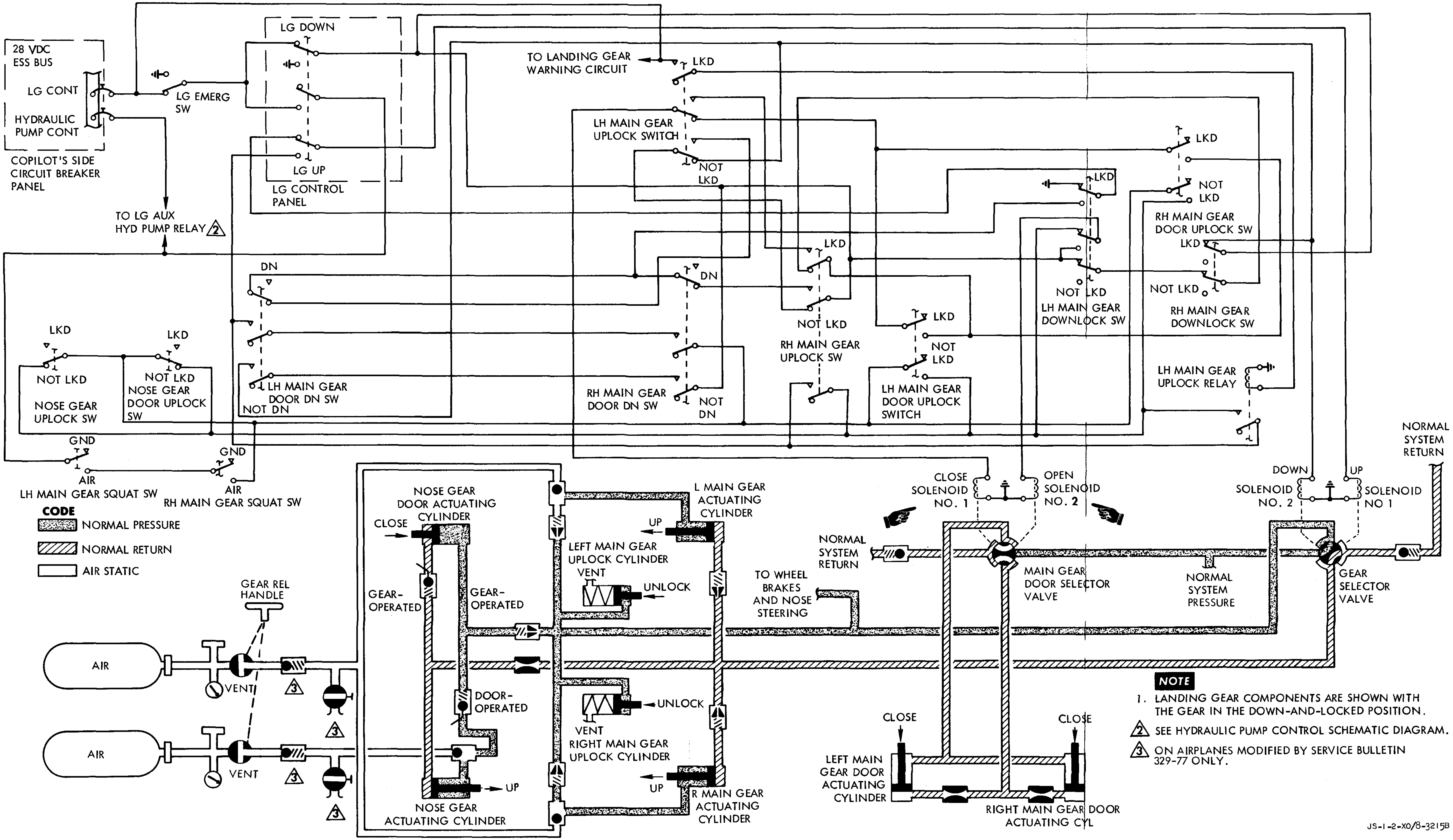


NOTE

1. LANDING GEAR COMPONENTS ARE SHOWN WITH THE GEAR IN THE DOWN-AND-LOCKED POSITION.

2 SEE HYDRAULIC PUMP CONTROL SCHEMATIC DIAGRAM

Figure 32-15A. Landing Gear Control Schematic Diagram (Airplanes 5056 and Up and 5041 thru 5055 Modified by Service Bulletin 329-184)



CODE

▨ NORMAL PRESSURE

▨ NORMAL RETURN

☐ AIR STATIC

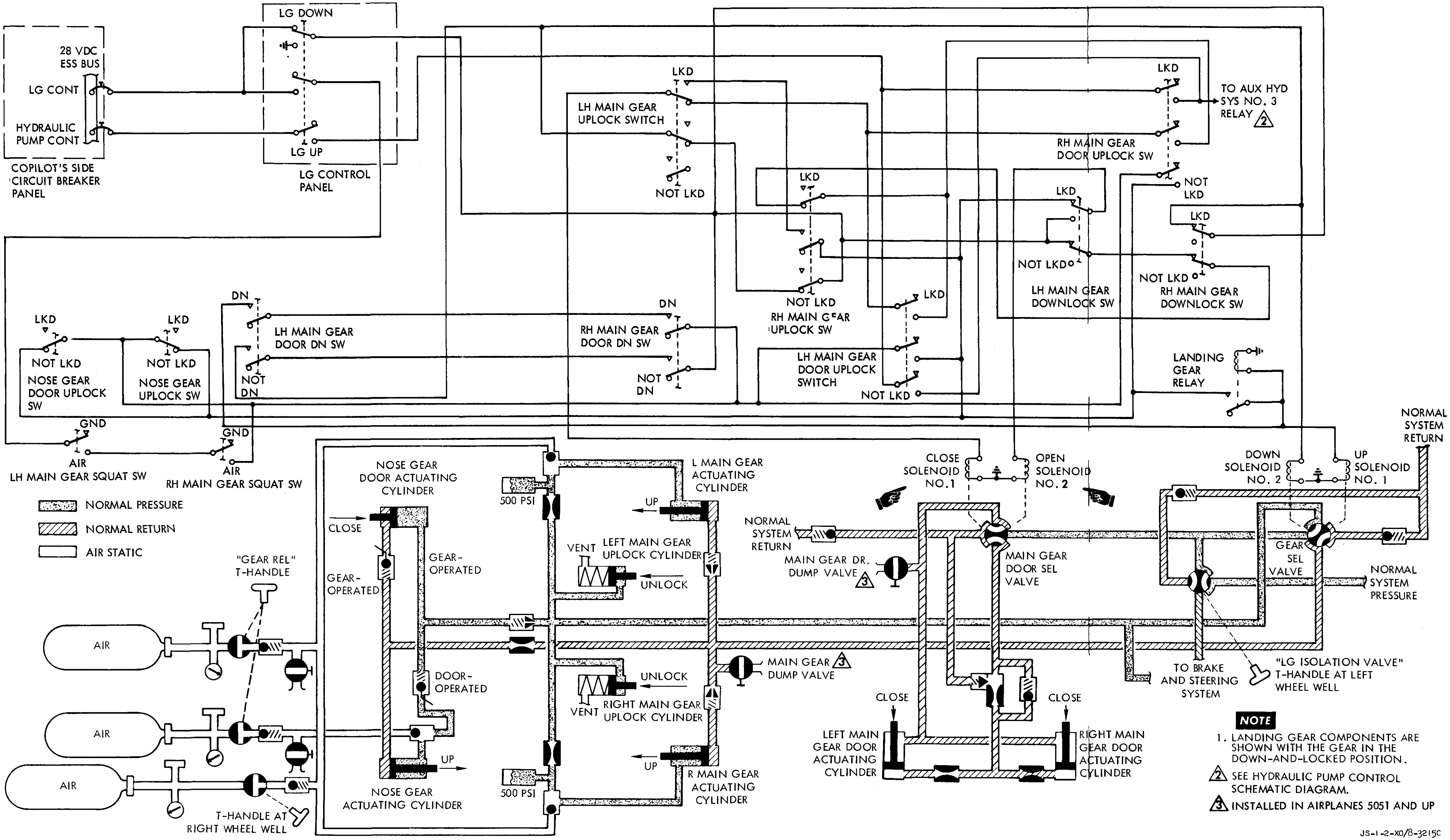
NOTE

1. LANDING GEAR COMPONENTS ARE SHOWN WITH THE GEAR IN THE DOWN-AND-LOCKED POSITION.

2 SEE HYDRAULIC PUMP CONTROL SCHEMATIC DIAGRAM.

3 ON AIRPLANES MODIFIED BY SERVICE BULLETIN 329-77 ONLY.

Figure 32-15B. Landing Gear Control Schematic Diagram (Airplanes 5001 thru 5040 Modified by Service Bulletin 329-291)



NOTE

1. LANDING GEAR COMPONENTS ARE SHOWN WITH THE GEAR IN THE DOWN-AND-LOCKED POSITION.

2 SEE HYDRAULIC PUMP CONTROL SCHEMATIC DIAGRAM.

3 INSTALLED IN AIRPLANES 5051 AND UP

Figure 32-15C. Landing Gear Control Schematic Diagram (Airplanes 5041 thru 5055 Modified by Service Bulletin 329-291)

Changed 16 October 1989

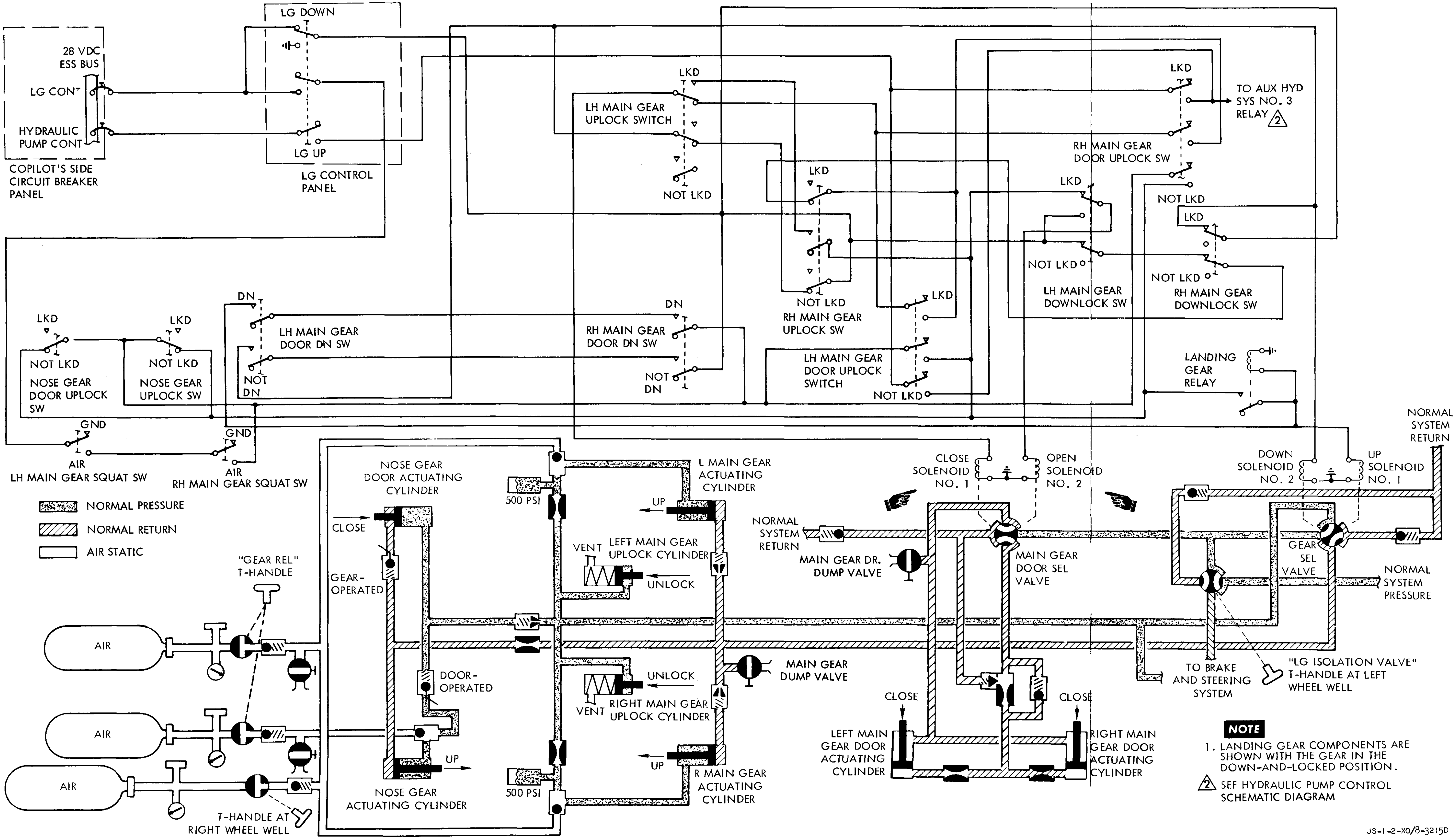


Figure 32-15D. Landing Gear Control Schematic Diagram (Airplanes 5056 thru 5162 Modified by Service Bulletin 329-291)

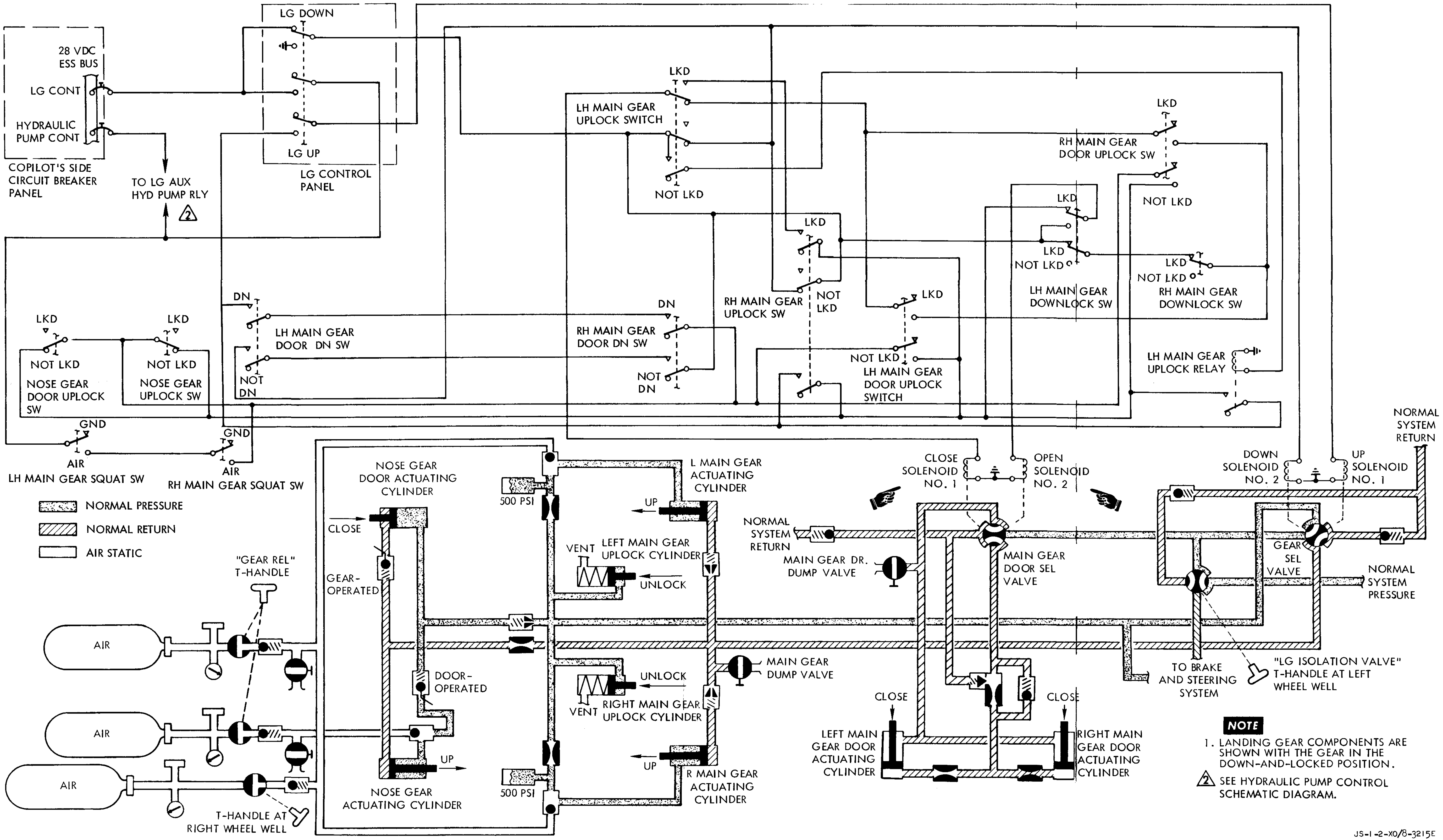


Figure 32-15E. Landing Gear Control Schematic Diagram
(Airplanes 5001 thru 5040 Modified by Service Bulletin 329-25B)

Changed 16 October 1989

SECTION 32.1
EMERGENCY GEAR RELEASE SYSTEM (AIRPLANES 5001 THRU 5085)

Contents Page

DESCRIPTION AND OPERATION 32-51
  General 32-51
  Gear Release Tee Handle 32-51
  LG EMERGENCY Selector Switch 32-52
  Standby Air Bottle, MLG Extension Tee Handle (Airplanes 5041 thru 5085) 32-52
  Standby Emergency Gear Release Tee Handles (Airplanes 5041 thru 5085) 32-52
  LG Isolation Valve Tee Handle (Airplanes 5041 thru 5085) 32-52
  Standby Nose Gear Door Release Pull Ring (Airplanes 5041 thru 5085) 32-52
  Standby Nose Gear Up-Latch Release Access Plate (Airplanes 5041 thru 5085) 32-52
  Inspection Windows (Airplanes 5041 thru 5085) 32-52
  Emergency Gear Extension Air Bottles 32-52
  Springs 32-52A
  Dump Valves 32-52A
  Vent Valves 32-52A
  Check Valves 32-52A

OPERATIONAL CHECKOUT 32-53
  Emergency Gear Release System (Airplanes 5001 thru 5040) 32-53
  Emergency Gear Release System (Airplanes 5041 thru 5085) 32-54

TROUBLESHOOTING 32-55
  Emergency Gear Release System 32-55

REMOVAL AND REPLACEMENT 32-56
  Removal of the Nose Gear Air Bottle 32-56
  Replacement of the Nose Gear Air Bottle 32-56
  Removal of the Main Gear Air Bottle 32-56
  Removal of the Standby Emergency Gear Release Air Bottle 32-57
  Replacement of the Standby Emergency Gear Release Air Bottle 32-57

ADJUSTMENT 32-57
  Emergency Gear Release System Rigging Instructions 32-57

FIGURES
  32-16. Emergency Gear Release System Components and Locations (Airplanes 5001 thru 5085). 32-59
  32-17. Emergency Gear Release System Rigging Instructions 32-60

32.1.1 DESCRIPTION AND OPERATION.

32.1.1.1 GENERAL. (See Figures 32-14, 32-15, and 32-16.) The emergency gear release system is actuated by a GEAR REL tee handle on the pilot's side inclined panel. Pulling this handle straight out to the first stop releases the gear and door uplocks. Rotating the handle 90 degrees clockwise and pulling it to the second stop discharges one air bottle into the down sides of the main gear actuating cylinders and discharges another air bottle into the down side of the nose gear actuating cylinder. On airplanes not modified by Service Bulletin 329-24, the GEAR REL tee handle also actuates a microswitch to isolate electrical power from the gear control circuits when the handle is pulled. On airplanes modified by Service Bulletin 329-24, this microswitch is replaced by an LG EMERGENCY switch on the pilot's side of the main instrument panel.

32.1.1.2 The gear release system on airplanes 5041 thru 5085 is similar to that of airplanes 5001-5040 except for

32.1.1.2 (Cont).

the omission of the microswitch (or LG EMERGENCY switch), and the addition of a standby emergency gear release system containing additional tee handles, a pull ring, main gear inspection windows, an isolation valve, a nose gear uplock access plate, and an additional main gear air bottle.

32.1.1.3 "GEAR REL" TEE HANDLE. The GEAR REL tee handle on the pilot's side inclined panel releases the gear and door uplocks when pulled straight out to the first stop. Rotating this handle 90 degrees clockwise and pulling it to the second stop discharges air bottles into the down sides of the gear actuating cylinders. This tee handle positions the uplocks and air bottle valves by means of cable systems.

32.1.1.4 LG EMERGENCY SELECTOR SWITCH. A two-position ("NORM," "EMERG") LG EMERGENCY switch on the pilot's side of the main instrument panel of airplanes modified by Service Bulletin 329-24 and airplanes 5036 through 5040 provides manual control capabilities for isolating electrical power from the gear control circuits. This switch remains in the "NORM" (guarded) position during normal gear actuation. Placing this switch in the "EMERG" (unguarded) position isolates electrical power from the gear control valves to prevent inadvertent energization of the valves during emergency gear extension.

32.1.1.5 The following components are on airplanes 5041 and up only: a STANDBY AIR BOTTLE, MLG EXTENSION tee handle, two STANDBY EMERGENCY GEAR RELEASE tee handles, an LG ISOLATION VALVE tee handle, a STANDBY EMERGENCY NOSE GEAR UP-LATCH RELEASE access plate, a STANDBY NOSE GEAR DOOR RELEASE pull ring, one inspection window in each main gear wheel well, a landing gear isolation valve, and a standby emergency gear release system air bottle for the main gear.

32.1.1.6 The sequence of operation for the standby emergency gear release system on airplanes 5041 and up depends upon the conditions existing when the gear fails to extend normally.

32.1.1.7 STANDBY AIR BOTTLE, MLG EXTENSION TEE HANDLE (AIRPLANES 5041 THRU 5085). This tee handle, on the right wheel well, is attached to a second main gear air bottle in the right wheel well. Pulling this tee handle positions a control valve on the air bottle to discharge its air through two shuttle valves and into the down sides of the main gear actuating cylinders.

32.1.1.8 STANDBY EMERGENCY GEAR RELEASE TEE HANDLES (AIRPLANES 5041 THRU 5085). One tee handle on each main gear wheel well is connected to the main gear inboard door uplocks and the main gear uplocks by means of cables. The handle on the left wheel well controls the left gear and door uplocks, and the handle on the right wheel well controls the right gear and door uplocks.

32.1.1.9 LG ISOLATION VALVE TEE HANDLE (AIRPLANES 5041 THRU 5085). An LG ISOLATION VALVE tee handle forward of the left wheel well is linked to a four-way, two-position hydraulic selector valve forward of the left wheel well. Pulling the handle positions the valve to bypass the landing gear and main landing gear door selector valves to supply hydraulic power to the wheel brake and nose steering systems, and also to open the gear and gear door hydraulic lines to the system return line to prevent hydraulic locks in the gear actuating system during emergency gear extension. This tee handle mechanically performs the same function that the micro or LG EMERGENCY switch electrically performs on airplanes 5001 thru 5040.

32.1.1.10 STANDBY NOSE GEAR DOOR RELEASE PULL RING (AIRPLANES 5041 THRU 5085). A pull ring in the floor next to the pilot's seat can be pulled to release the nose gear forward door uplock. This pull ring is connected to the nose gear door uplock by means of a cable system.

32.1.1.11 STANDBY NOSE GEAR UP-LATCH RELEASE ACCESS PLATE (AIRPLANES 5041 THRU 5085). A round plate in the floor of the pilots' compartment can be removed to permit insertion of a rod to press down on and physically release the nose gear uplock.

32.1.1.12 INSPECTION WINDOWS (AIRPLANES 5041 THRU 5085). One inspection window in each main wheel well provides a means of visually inspecting the main gear and door uplocks during flight. These windows are in the inboard sides of the wheel well boxes.

32.1.1.13 EMERGENCY GEAR EXTENSION AIR BOTTLES. Airplanes 5001 through 5040 have two air bottles to assist in emergency gear extension. One discharges into the down sides of the main gear actuating cylinders and the other discharges into the down side of the nose gear actuating cylinder. The bottle for the main gear is in the left wheel well, and the bottle for the nose gear is in the nose wheel well. Both are controlled by valves on the air bottle assemblies themselves. The valves are positioned by cables running to the GEAR REL handle. Airplanes 5041 and up have both of these air bottles, and also have an additional main gear air bottle in the right wheel well. This standby air bottle for the main gear is controlled by a tee handle on the right wheel well. When air bottles are not discharged, their control valves vent the lines to prevent air lock interference with the hydraulic actuation of the gear. When an air bottle is discharged, the control valve closes the vent. A check valve in the air line of the standby air bottle prevents air from the normal air bottle from escaping through the vent of the standby air bottle. The normal air bottle is discharged first, so its vent is closed when the standby air bottle is discharged.

32.1.1.13 (Cont).

A pressure gauge and a filler valve are provided for each air bottle assembly to facilitate servicing. Airplanes 5044 through 5085, and airplanes incorporating Service Bulletin 329-96, are equipped with relief valves at each air bottle assembly. The relief valves protect bottles from overcharging and from the results of thermal expansion.

32.1.1.14 SPRINGS. Airplanes 5001 through 5040 have three springs - one at the nose gear uplock, one at the nose gear air bottle, and one at the main gear release air bottle - which maintain cable tension in the emergency gear release system. The spring at the nose gear uplock is attached at one end to a bracket on the structure, and at the other end to a cable clamping device on the uplock cable. The spring at the nose gear air bottle is attached at one end to the air bottle discharge line, and at the other end to a cable clamping device on the air bottle control cable. The spring at the main gear air bottle is attached at one end to a bracket on the structure, and at the other end to a cable clamping device on the air bottle control cable.

32.1.1.15 DUMP VALVES. Airplanes 5051 and up, and airplanes having Service Bulletin 329-177 or Service Bulletin 329-186 incorporated, have hydraulic fluid dump valves provided for the landing gear up and the main landing gear door close pressure lines. Opening the valve will relieve the pressure or pressure blockage in the associated line. The dump valves are located on the inboard wall of the right hand wheel well and are accessible through the removable panel that supports the inspection window.

CAUTION

The dump valves are to be used only in accordance with instructions in the JetStar Airplane Flight Manual (SMR 150).

Note

For airplanes modified by Service Bulletin 329-186, the dump valves are located inside the cabin on the upper inboard wall of the left hand main wheel well.

32.1.1.16 VENT VALVES. Airplanes 5044 and up, and airplanes having Service Bulletin 329-77 or Service Bulletin 329-162 incorporated, have vent valve assemblies installed in the plumbing lines for the nose landing gear and main landing gear emergency extension air bottles. The vent valves are provided to facilitate maintenance following discharge of an air bottle for any reason.

32.1.1.17 CHECK VALVES. Several check valve assemblies are installed in the emergency landing gear release system hydraulic plumbing, positioned so as to prevent undesirable hydraulic fluid flow.

a. Airplanes 5041 and up, and airplanes having Service Bulletin 329-25 incorporated, have a check valve in the discharge line from the right hand main gear emergency extension air bottle assembly. Airplanes 5044 and up and airplanes having Service Bulletin 329-25 incorporated have check valves in the discharge line from the nose gear and the left hand main gear emergency extension air bottle assemblies. These check valves protect the bottle assemblies from reverse flow into the bottle, whether air or fluid, and prevent loss of air from one bottle through the vent of another bottle.

b. Airplanes 5041 and up, and airplanes having Service Bulletin 329-25 incorporated, have a check valve assembly installed in the return line downstream of the isolation valve which protects against transient surges in the normal (No. 1) return system.

c. Airplanes 5041 and up, and airplanes having Service Bulletin 329-25 incorporated, have a check valve assembly installed in the pressure line downstream of the isolation valve, and downstream of the tap off for steering control. This valve prevents loss of normal (No. 1) system brake pressure through the isolation valve plumbing.

d. A check valve is installed in the normal (No. 1) system pressure line to the brake system. This check valve prevents loss of brake accumulator pressure when the normal system is shut down.

32.1.1.17 (Cont).

e. Airplanes 5041 and up, and airplanes having Service Bulletin 329-25 incorporated, have a check valve installed in the normal (No. 1) system pressure line to the steering system. This check valve prevents loss of isolation valve pressure through normal system plumbing.

32.1.2 OPERATIONAL CHECKOUT.

32.1.2.1 EMERGENCY GEAR RELEASE SYSTEM (AIRPLANES 5001 THRU 5040). Before starting this checkout, make sure that the main gear air bottle in the left wheel well and the nose gear air bottle in the nose wheel well are both charged to 1,800 (+200, −0) PSI as shown by the air pressure gages on the bottle assemblies.

a. Jack the airplane in accordance with Chapter 7, using wing and fuselage jacks. The wheels must clear the ground.

b. Connect external hydraulic power to the normal hydraulic system and connect external 28-volt, DC power to the airplane. Adjust the output to no flow at 3,000 PSI and 8 gallons per minute at 2,700 PSI.

c. Attach a force measuring tool to the GEAR REL handle. The tool should be capable of measuring up to 100 pounds of pull.

d. Remove the landing gear ground safety locks and retract the landing gear.

e. Shut down the hydraulic unit.

f. Deplete the normal hydraulic pressure to zero by cycling the ailerons and elevators with the SYS NO. 1 BOOST HYD PRESS switches on the pilot's control column cover in the "ON" position.

g. Drain or draw all of the hydraulic fluid out of the normal system reservoir, to prevent overflow during discharging of the air bottles.

h. With no pressure in the normal hydraulic system, place the gear control handle in the "LG DOWN" position.
RESULT: The landing gear should remain retracted.

i. Using the force measuring device, pull the GEAR REL handle out to the first stop, noting the amount of force necessary. Observe the landing gear.
RESULT: The amount of force required to pull the GEAR REL handle should not exceed 90 pounds. As the handle is pulled, the gear should release from its uplocks, and both the main and the nose gear should free-fall.

j. If the nose gear falls into its downlock, mechanically release it from the downlock and leave the gear in the partially down position.

k. Rotate the GEAR REL handle 90 degrees clockwise and pull it to the second stop.
RESULT: As the GEAR REL handle is pulled, the nose and main gear air bottles should discharge and force the gear into the downlocks.

l. Service the normal system reservoir according to the instructions in Chapter 12, and return the GEAR REL handle to the normal (fully in) position.

CAUTION

Do not use the fluid drained in step (g) for servicing unless absolutely sure that it is clean.

CAUTION

After returning the discharged air bottle lever to the normal position, bleed the air trapped in the down side of the nose and main gear actuators through the bleed valves provided.

m. With all personnel clear of the gear doors, pressurize the normal hydraulic system to approximately 3,000 PSI.
RESULT: The main gear inboard doors should close and lock.

CAUTION

After an emergency gear extension, always pressurize the down side of the gear system before retracting the gear. This will reset the shuttle valves to the normal operating position.

n. Cycle the landing gear six complete cycles to bleed all air from the system.

CAUTION

Cycle the gear according to the instructions for bleeding the normal and auxiliary systems in Chapter 29. If excessive pressure is used during bleeding of the gear actuating cylinders, uplock damage could result.

o. On airplanes modified by Service Bulletin 329-24, disregard step (p) and its result. On airplanes not modified by Service Bulletin 329-24, disregard step (q) and its result.

p. With the gear in the down-and-locked position, pull the GEAR REL tee handle out approximately 4/10 inch, and then place the gear control handle in the "LG UP" position. Upon completing this

32.1.2.1 (Cont).

step, return the gear control handle to the "LG DOWN" position, and then return the GEAR REL tee handle to the fully-in position.
RESULT: The gear should not actuate.

q. With the gear in the down-and-locked position, place the LG CONT switch on the pilot's side of the main instrument panel in the "EMERG" position, and then place the gear control handle in the "LG UP" position. Upon completing this step, return the gear control handle to the "LG DOWN" position, and then place the LG CONT switch in the "NORM" position.
RESULT: The gear should not actuate.

r. Install the landing gear ground safety locks, disconnect all external power, and lower the airplane to the ground.

s. Service the air bottles according to the instructions in Chapter 12.

32.1.2.2 EMERGENCY GEAR RELEASE SYSTEM (AIRPLANES 5041 thru 5085 ONLY). Before starting this checkout, make sure that the two main and one nose gear air bottles are charged to 1,800 (+200, −0) PSI according to the instructions in Chapter 12.

a. Jack the airplane in accordance with Chapter 7, using wing and fuselage jacks. The wheels must clear the ground.

b. Connect external hydraulic power to the normal hydraulic system, and connect 28-volt, DC power to the airplane service connections. Adjust the output of the hydraulic power source to deliver no flow at 3,000 PSI and 8 gallons per minute at 2,700 PSI.

c. Attach a force measuring device to the GEAR REL handle. The tool should be capable of measuring up to 100 pounds of pull.

d. Remove the landing gear ground safety locks and retract the landing gear.

e. Shut down the hydraulic unit.

f. Deplete the normal hydraulic system pressure to zero by cycling the aileron and elevator control systems with the BOOST HYD PRESS switches in the "ON" position.

g. Drain or draw all of the hydraulic fluid out of the normal system reservoir, to prevent overflow during air bottle discharge.

h. With no pressure on the normal hydraulic system, place the gear control handle in the "LG DOWN" position.
RESULT: The landing gear actuating system should not be affected.

i. Using the force measuring device, pull the GEAR REL handle out to the first stop, noting the amount of force required. Observe the landing gear.
RESULT: The amount of force required to pull the GEAR REL handle should not exceed 90 pounds. As the handle is pulled, the gear should release from the uplocks and free fall.

j. If the nose gear falls into its downlock, mechanically release it from the downlock and leave the gear in the partially down position.

k. Rotate the GEAR REL handle 90 degrees clockwise and pull it to the second stop.
RESULT: As the GEAR REL handle is pulled, the nose and main gear air bottles should discharge and force the gear into the downlocks.

l. Service the normal system reservoir according to the instructions in Chapter 12, and return the GEAR REL handle to the normal (fully in) position.

CAUTION

Do not use fluid drained in step (g) for servicing unless absolutely sure that it is clean.

CAUTION

After returning the discharged air bottle lever to the normal position, bleed the air trapped in the down side of the nose and main gear actuators through the bleed valves provided.

m. With all personnel clear of the gear doors, pressurize the normal hydraulic system to approximately 3,000 PSI.
RESULT: The main gear inboard doors should close and lock.

CAUTION

After an emergency gear extension, always pressurize the down side of the gear system before retracting the gear. This will reset the shuttle valves to the normal operating position.

n. Cycle the landing gear six complete cycles to bleed all air from the system.

CAUTION

Cycle the gear according to the instructions for bleeding the normal and auxiliary hydraulic systems in Chapter 29. If excessive pressure is used during bleeding of the gear system, uplock damage could result.

32.1.2.2 (Cont).

o. Retract the landing gear.

p. Deplete hydraulic pressure in the brake system by operating the wheel brakes, and observe the wheel brakes, and observe the BRAKE PRESS LOW warning light on the annunciator warning panel.
RESULT: The BRAKE PRESSURE LOW warning light should go on.

q. Pull the LG ISOLATION VALVE tee handle on the left wheel well and observe the BRAKE PRESS LOW warning light.
RESULT: The warning light should go out indicating that brake pressure is available.

r. With the landing gear control handle in the "LG UP" position, pull the STANDBY EMERGENCY NOSE GEAR DOOR RELEASE pull ring in the pilots' floor area.
RESULT: The forward nose gear doors should unlock.

s. Remove the EMERGENCY NOSE GEAR UPLATCH RELEASE access plate and use a rod to release the nose gear uplock.
RESULT: The nose gear should free-fall.

t. If the nose gear locks down, mechanically unlock the downlock and leave the nose gear partially down.

u. Pull the right and the left STANDBY EMERGENCY GEAR RELEASE tee handle and observe the effect on the main gear and doors.
RESULT: As each tee handle is pulled, the

32.1.2.2 (Cont).

corresponding gear and gear doors should unlock and the main gear should free-fall.

v. After repeating step (f), pull the STANDBY AIR BOTTLE MLG EXTENSION tee handle on the right wheel well.
RESULT: The standby air bottle should discharge and force the main gear into its downlocks.

w. Return the pull ring and all tee handles to their normal (in) positions, and replace the nose gear uplock access plate.

x. Place the gear control handle in the "LG DOWN" position, reapply hydraulic pressure to the normal system, and then bleed the landing gear system according to the instructions in Chapter 29.

y. Install the landing gear ground safety locks, disconnect all external power, and lower the airplane to the ground.

z. Service the normal system reservoir and all three air bottles according to the instructions in Chapter 12.

32.1.3 TROUBLESHOOTING.

32.1.3.1 EMERGENCY GEAR RELEASE SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
EXCESSIVE FORCE REQUIRED TO PULL GEAR RELEASE HANDLE.		
Binding, interference, or damage to handle mechanism and cable system.	Check the entire release system for evidence of binding, interference, or damage.	Repair or replace components as needed, and then re-rig the system.
Gear or door uplock binding or improperly adjusted.	Check the operation of all gear and door uplocks.	Repair, replace, or adjust components as needed.
AIR BOTTLES FAIL TO DISCHARGE.		
Emergency gear release system improperly rigged.	Check rigging of entire system.	Rig system as needed.
Defective control valves on one or more air bottles.	Check the operation of the valves.	Repair or replace the valves as needed.
GEAR AND DOOR UPLOCKS FAIL TO RELEASE WHEN THE "GEAR REL" HANDLE IS PULLED TO THE FIRST STOP.		
Improperly rigged gear release system.	Check the rigging of the entire system.	Replace components as needed, and then re-rig the system.
Landing gear doors interfering with gear operation.	Check main and nose gear doors for signs of interference or improper adjustment.	Replace components as needed, and then re-rig the release system and doors.
GEAR FAILS TO LOCK FULLY DOWN WHEN AIR BOTTLES ARE DISCHARGED.		
Defective downlock mechanisms.	Check the operation of the gear downlock mechanisms.	Repair or adjust the downlocks as needed.
Insufficient air pressure in air bottles.	Recharge air bottles as needed, and then perform an operational checkout of the system.	Repair air bottles or valves as needed.

32.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
GEAR FAILS TO LOCK FULLY DOWN WHEN AIR BOTTLES ARE DISCHARGED (Cont).		
If trouble occurs during standby air bottle use (on airplanes 5041 thru 5085 only), the check valve in the normal air bottle line may be stuck in the open position.	Check the system for evidence of leakage from the normal air bottle control valve vent.	Replace the check valve as needed.
Defective landing gear selector valve and microswitch on the "GEAR REL" handle assembly (airplanes not modified by Service Bulletin 329-24 only).	Check the operation of the selector valve and the microswitch.	Replace the defective valve and adjust or replace the microswitch as needed.
Defective landing gear selector valve, and "LG EMERGENCY" selector switch improperly positioned or wired. (Airplanes modified by Service Bulletin 329-24).	Check the operation of the valve, and then check the position and wiring of the switch.	Replace or repair components as needed.

32.1.4 REMOVAL AND REPLACEMENT.

32.1.4.1 REMOVAL OF THE NOSE GEAR AIR BOTTLE.

a. Remove the nose strut door.

b. Remove the deflector plate from under the air bottle.

c. Disconnect the emergency gear extension airline in front of the bracket on the nose gear actuating cylinder, so that when the bottle is discharged, the bracket will hold the line securely.

d. Discharge the nose gear air bottle by slowly opening the control valve.

WARNING

Keep personnel clear of the air discharge line during bottle discharge.

e. Disconnect the emergency gear release cable from the air bottle control arm, and disconnect the servicing valve and gage assembly from the air bottle.

32.1.4.1 (Cont).

f. Remove the four bolts, nuts, and washers attaching the air bottle to its shelf, and lift the air bottle free of the shelf.

32.1.4.2 REPLACEMENT OF THE NOSE GEAR AIR BOTTLE.

a. Reverse the removal procedures, being sure to check system rigging before installing the deflector.

b. Service the air bottle according to the instructions in Chapter 12.

32.1.4.3 REMOVAL OF THE MAIN GEAR AIR BOTTLE.

a. Open the left main gear inboard door, and install a ground safety lock on the actuating cylinder to hold the door in the open position.

b. Disconnect the left gear emergency extension line below the bracket near the inboard door actuating cylinder so that the bracket will keep the line from whipping during air bottle discharge.

32.1.4.3 (Cont).

c. Disconnect the emergency gear extension line from the top of the three-tube bracket near the right main gear inboard door actuating cylinder. Install a lock on the door if it is opened.

d. Slowly discharge the left main gear air bottle.

WARNING

Keep personnel clear of the air discharge lines during air bottle discharge.

e. Disconnect the cable from the air bottle in the left wheel well.

f. Remove the four screws holding the air bottle to the attaching bracket, and then loosen the clamp from around the bottle and remove the bottle.

32.1.4.4 REPLACEMENT OF MAIN GEAR AIR BOTTLE.

a. Reverse the removal procedures.

b. Check the operation of the gear release system, and rig the sytsem as required.

c. Service the air bottle according to instructions in Chapter 12.

32.1.4.5 REMOVAL OF THE STANDBY EMERGENCY GEAR RELEASE AIR BOTTLE.

a. Open the main gear inboard doors and install ground safety locks on their actuating cylinders.

b. Disconnect the emergency gear release air line near the three-tube bracket in the right wheel well near the door actuating cylinder, so that the tube will not whip during air bottle discharge.

c. Disconnect the emergency gear release air line in the left wheel well at the tee near the normal system air bottle.

d. Slowly discharge the standby air bottle.

WARNING

Keep personnel clear of the air discharge lines during air bottle discharge.

e. Disconnect the air bottle cable and remove the four screws holding the air bottle to its mounting bracket.

32.1.4.6 REPLACEMENT OF THE STANDBY EMERGENCY GEAR RELEASE AIR BOTTLE (AIRPLANES 5041 THRU 5085).

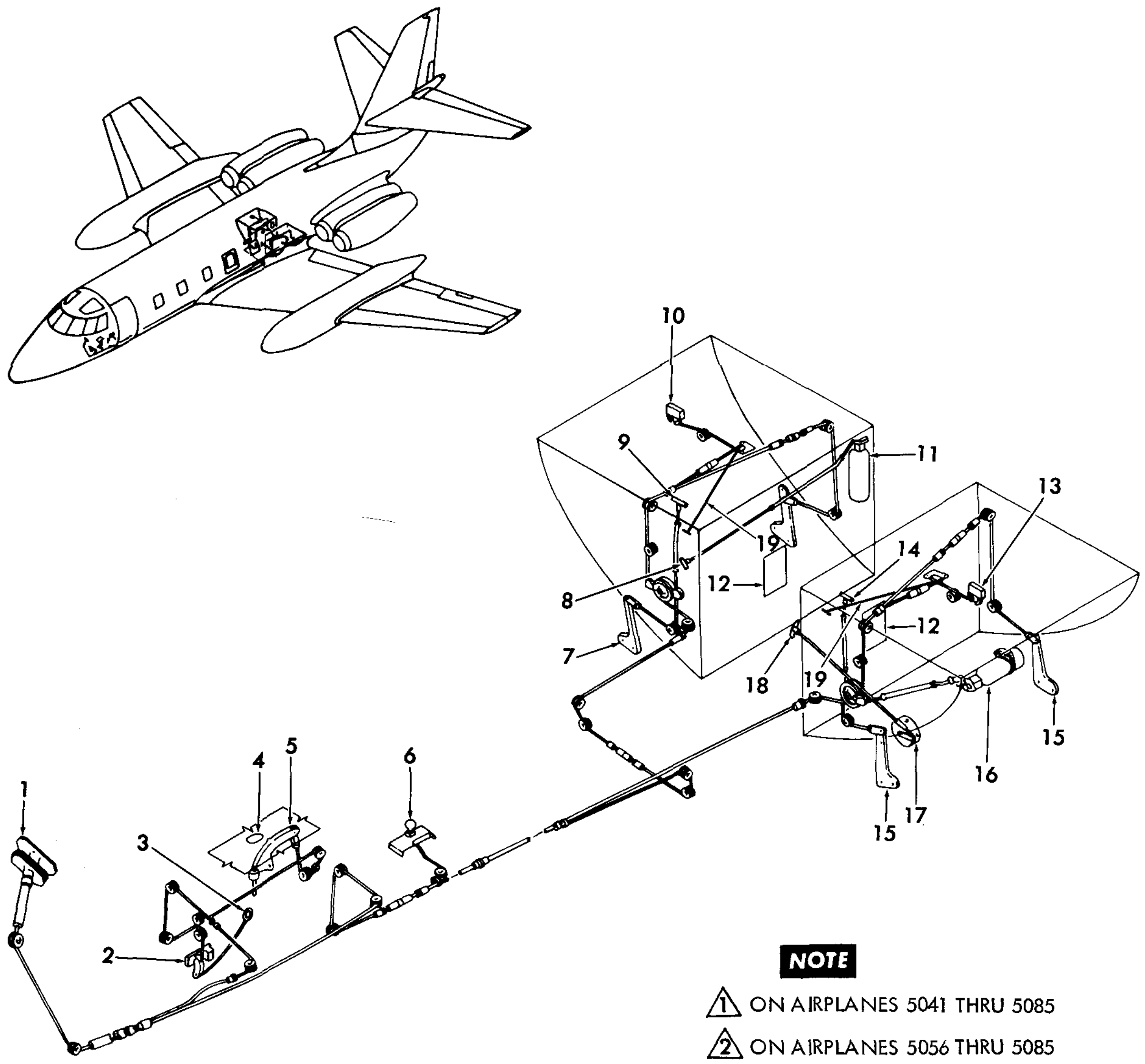
a. Reverse the removal procedures.

b. Check the operation of the gear release system, and rig the system as needed.

c. Service the air bottle according to the instructions in Chapter 12.

32.1.5 ADJUSTMENT.

32.1.5.1 EMERGENCY GEAR RELEASE SYSTEM RIGGING INSTRUCTIONS. (See figure 32-17.) Rig the emergency gear release system of all airplanes according to the instructions in figure 32-16. After rigging airplanes 5041 thru 5085, check that all standby tee handles and the pull ring are adjusted so that their cables or linkages are in the proper positions, with no slack.



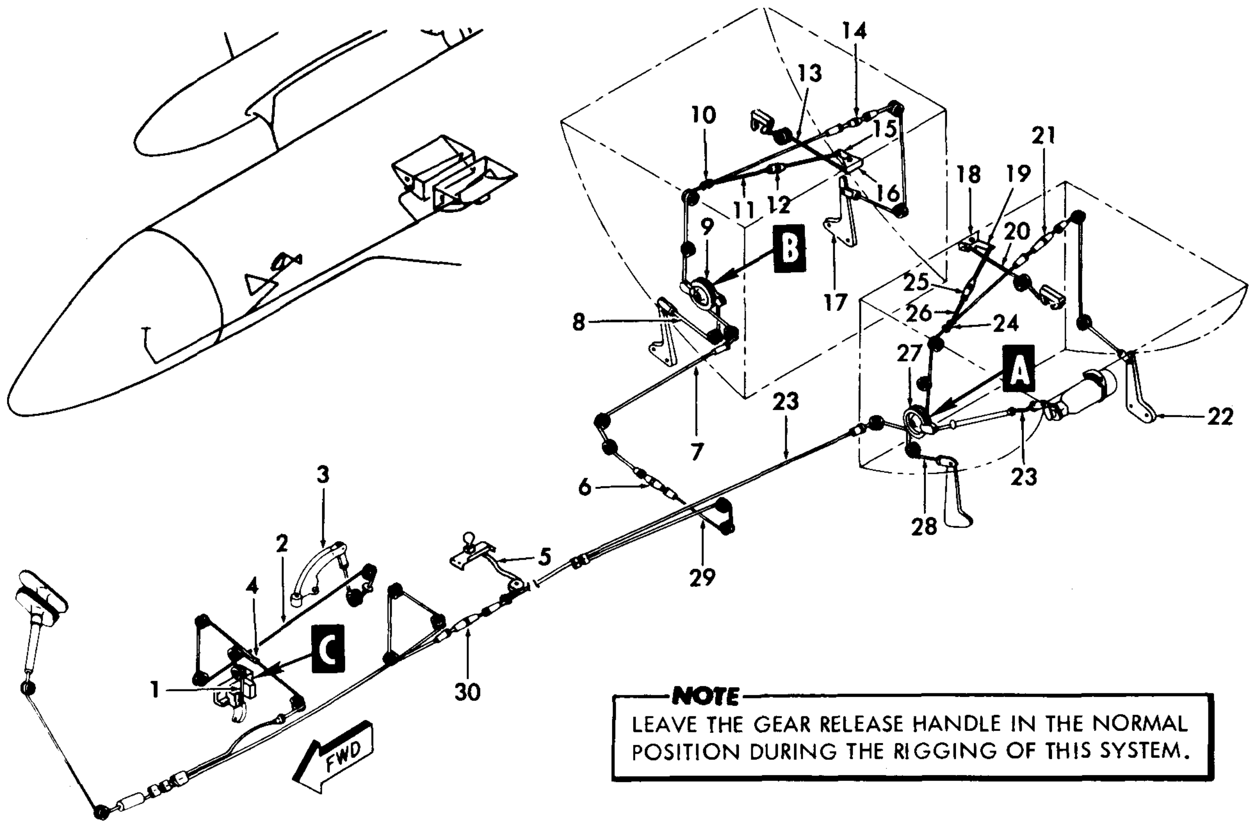
NOTE

△1 ON AIRPLANES 5041 THRU 5085

△2 ON AIRPLANES 5056 THRU 5085

1. "GEAR REL" TEE HANDLE
2. NOSE GEAR DOOR UPLOCK
3. "STANDBY NOSE GEAR DOOR RELEASE" PULL RING △1
4. "STANDBY EMERGENCY NOSE GEAR UP-LATCH △1 RELEASE" ACCESS HOLE
5. NOSE GEAR UPLOCK
6. NOSE GEAR AIR BOTTLE
7. RT MAIN GEAR DOOR UPLOCK
8. "STANDBY AIR BOTTLE M.L.G. EXTENSION" TEE HANDLE △1
9. "STANDBY EMERGENCY GEAR RELEASE" TEE HANDLE (RIGHT MAIN GEAR) △1
10. RT MAIN GEAR UPLOCK
11. "STANDBY AIR BOTTLE M.L.G. EXTENSION" △1
12. INSPECTION WINDOW △1
13. LEFT MAIN GEAR UPLOCK
14. "STANDBY EMERGENCY GEAR RELEASE" TEE HANDLE (LEFT MAIN GEAR) △1
15. LEFT MAIN GEAR DOOR UPLOCK
16. MAIN GEAR AIR BOTTLE
17. LG ISOLATION VALVE △1
18. "LG ISOLATION VALVE" TEE HANDLE △1
19. STANDBY UPLOCK RELEASE CABLE △2

Figure 32-16. Emergency Gear Release System Components and Locations (Airplanes 5001 thru 5085)



NOTE

LEAVE THE GEAR RELEASE HANDLE IN THE NORMAL POSITION DURING THE RIGGING OF THIS SYSTEM.

STEP 1

REMOVE THE FLOOR PANELS AS NECESSARY TO OBTAIN ACCESS TO THE GEAR RELEASE CABLE ADJUSTMENT POINTS.

WARNING

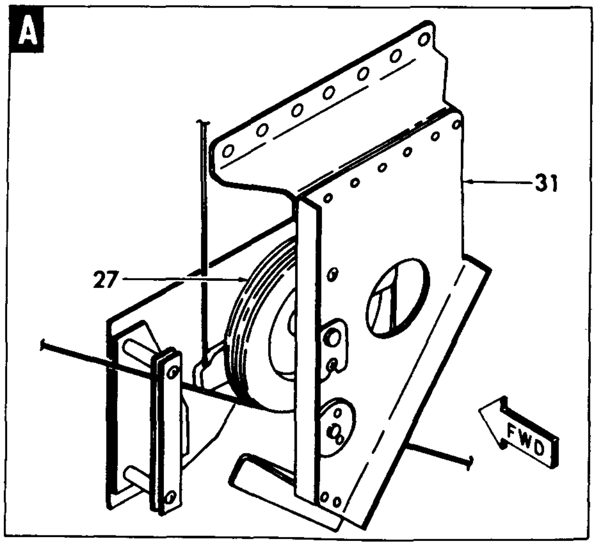
DO NOT WORK AROUND OPEN MAIN GEAR INBOARD DOORS UNLESS DOOR SAFETY LOCKS ARE INSTALLED. ACCIDENTAL ACTUATION CAN CRITICALLY INJURE PERSONNEL.

STEP 2

INSTALL A 3/16-INCH RIG PIN THROUGH THE BRACKET (18) AND THE LEVER (19) IN THE LEFT WHEEL WELL.

STEP 3

ADJUST THE TURNBUCKLE (25) UNTIL THE CABLE RUN (26) FROM THE LEVER (19) TO THE PULLEY (27) IS TAUT (NO SLACK) WHEN THE RIG PIN HOLES IN THE PULLEY (27) AND THE BRACKET (31) ARE ALIGNED.



STEP 4

ADJUST THE TURNBUCKLE (21) TO OBTAIN 1/4 INCH TRAVEL SLACK IN THE CABLE RUN BETWEEN THE SPLICE (24) AND THE AFT DOOR LATCH (22) WHEN THE PULLEY (27) IS IN THE POSITION OF STEP (3).

JS-2-2-3216-1

Figure 32-17. Emergency Gear Release System Rigging Instructions (Sheet 1 of 2)

STEP 5

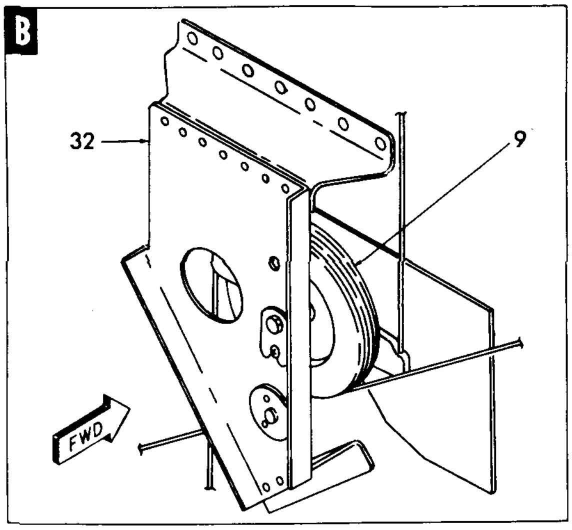
ADJUST THE LENGTH OF CABLE (28) TO OBTAIN 0.55 INCH TRAVEL SLACK WHEN THE PULLEY (27) IS IN THE POSITION OF STEP (3).

STEP 6

INSTALL A 3/16-INCH RIG PIN THROUGH THE BRACKET (16) AND THE LEVER (15) IN THE RIGHT WHEEL WELL .

STEP 7

ADJUST THE TURNBUCKLE (12) UNTIL THE CABLE RUN (11) FROM THE LEVER (15) TO THE PULLEY (9) IS TAUT (NO SLACK) WHEN THE RIG PIN HOLES IN THE BRACKET (32) AND THE PULLEY (9) ARE ALIGNED.



STEP 8

ADJUST THE TURNBUCKLE (14) TO OBTAIN 1/4 INCH TRAVEL SLACK IN THE CABLE RUN FROM THE SPLICE (10) TO THE AFT DOOR LATCH (17) WHEN THE PULLEY (9) IS IN THE POSITION OF STEP (7).

STEP 9

ADJUST THE LENGTH OF CABLE (8) TO OBTAIN 1.05 INCH TRAVEL SLACK.

STEP 10

REMOVE THE RIG PINS INSTALLED IN STEPS (2) AND (6).

STEP 11

ADJUST THE TURNBUCKLE (30) TO ALIGN THE RIG PIN HOLES IN THE LEVER (19) AND THE BRACKET (18).

STEP 12

INSERT A 3/16-INCH RIG PIN IN THE PULLEY (9), AND THEN ADJUST THE TURNBUCKLE (6) UNTIL THE CABLES (7) AND (29) ARE TAUT (NO SLACK).

STEP 13

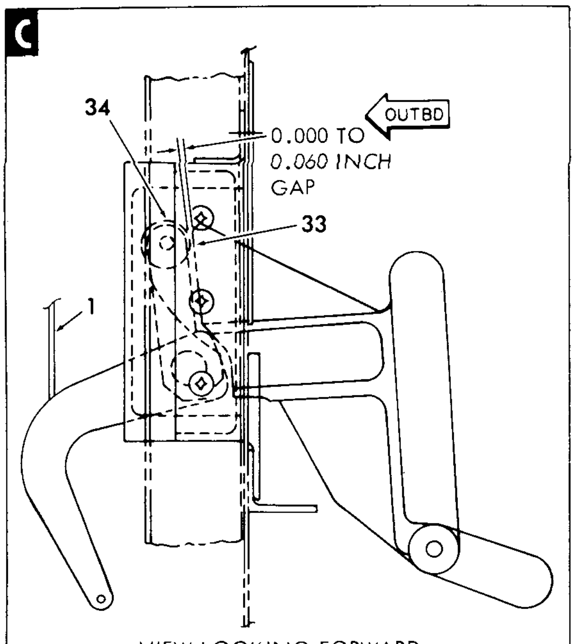
WITH THE MAIN AND NOSE GEAR AIR BOTTLE LEVERS HELD IN THE CLOSED POSITIONS, ADJUST THE LENGTHS OF CABLES (5) AND (23) TO OBTAIN 3.70 INCHES TRAVEL SLACK IN EACH OF THEM.

STEP 14

ADJUST THE LENGTH OF CABLE (2) TO OBTAIN 2-1/2 INCH TRAVEL SLACK IN THE CABLE RUN FROM THE SPLICE (4) TO THE ARM (3).

STEP 15

ADJUST THE LENGTH OF THE CABLE (1) TO OBTAIN CLEARANCE FROM ZERO TO 0.060 INCH BETWEEN THE ROLLER (34) AND THE ARM (33) WHEN THE FORWARD NOSE GEAR DOORS ARE CLOSED (SEE VIEW "C").



VIEW LOOKING FORWARD
AT LEFT SIDE OF NOSE WHEEL WELL

STEP 16

ADJUST THE LENGTHS OF CABLES (13) AND (20) UNTIL THEY ARE TAUT (BUT WITH NO TENSION) WHEN THE MAIN LANDING GEAR UPLOCKS ARE IN THE GEAR UP-AND-LOCKED POSITION.

STEP 17

REMOVE THE RIG PIN INSTALLED IN STEP (12), CHECK THAT ALL CONNECTIONS ARE SECURE AND SAFETIED AS REQUIRED; THEN PERFORM AN OPERATIONAL CHECKOUT OF THE SYSTEM.

Figure 32-17. Emergency Gear Release System Rigging Instructions (Sheet 2 of 2)

SECTION 32.2
LANDING GEAR INDICATING AND WARNING SYSTEM

Contents

Page

DESCRIPTION AND OPERATION 32-63
General 32-63
Landing Gear Indicator Lights 32-64
Landing Gear Warning Lights 32-64
Audible Gear Warning 32-64
Warning Silencer Circuit 32-64
Warning and Indicating Light Dimming Circuit 32-64
Gear Warning Relay 32-64A

OPERATIONAL CHECKOUT 32-65
Landing Gear Indicating and Warning System 32-65

TROUBLESHOOTING 32-65
Landing Gear Indicating and Warning System 32-65

ADJUSTMENT 32-67
Landing Gear and Door Switch Adjustment 32-67

FIGURES
32-18. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5001 thru 5040) 32-69
32-19. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5041 thru 5065) 32-71
32-19A. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5066 and Up) 32-72A

32.2.1 DESCRIPTION AND OPERATION.

32.2.1.1 GENERAL. (See figures 32-18, 32-19, and 32-19A.) The landing gear indicating and warning system attracts the pilots' attention through use of green indicator lights, red warning lights, and an audible warning signal.

32.2.1.2 Three green indicator lights on the landing gear control panel glow when the landing gear is down and locked. The light on the left is for the left main gear, the light in the center is for the nose gear, and the light on the right is for the right main gear. Each light operates independently of the others. As each gear reaches the down-and-locked position, the corresponding indicator light goes on. Electrical power to these lights is present when the 28-volt essential DC bus is energized. When the corresponding gear downlock switch is in the locked position, thus supplying a ground circuit to the light, the light goes on.

32.2.1.3 Two bulbs in the landing gear control handle go on when the landing gear is in an intermediate position. These bulbs illuminate the handle to give a warning when the throttles are retarded below the minimum cruise position with thelanding gear not down and locked, and also when the landing gear is not in the same position as shown by the position of the gear control handle. Electrical power to these bulbs is present when the 28-volt, essential DC bus is energized and the controlling ground circuit is completed by any one or all of the following: the right main gear downlock switch, the nose gear door uplock switch, the nose gear uplock switch, the left and right main gear door uplock switches, the left main gear uplock relay (5001 thru 5040), and/or the left main gear uplock switch (5041 and up), and the right main gear uplock switch.

32.2.1.4 The audible gear warning signal is a pulsating tone of approximately 13 to 18 decibels to warn of one or more throttles being retarded below a position corresponding to 1.4 to 1.5 engine pressure ratio (EPR), with the landing gear not down and locked.

32.2.1.5 The warning tone is silenced by pressing the WARNING SILENCER pushbutton on the landing gear control panel. This silence circuit silences the tone whether it is due to one or all of the throttles being retarded. Relays in the gear signal disconnect circuit break the circuit from the landing gear warning switch which is completing the warning circuit.

32.2.1.6 A warning and indicating light dimming circuit operates in conjunction with the exterior airplane position lights to dim the warning and indicating lights

32.2.1.6 (Cont).

at night, to prevent unnecessary glare in the pilots' compartment. This circuit is controlled by the NAV POS LIGHT switch. When the position lights are turned on, the switch energizes two relays in the dimming circuit to route indicating and warning light circuits through resistors to dim the lights.

32.2.1.7 LANDING GEAR INDICATOR LIGHTS. Three green indicator lights on the landing gear control panel go on when the landing gear is down and locked. Each light operates independently of the others, and all lights must show green for a safe landing indication. Electrical power to these lights is supplied by the 28-volt, essential DC bus through the landing gear down and locked indicator circuit breaker. The controlling ground circuits are completed through the gear downlock switches. Each indicator light incorporates a press-to-test feature to permit bulb testing. Pressing the housing of each light makes the bulb go on if the landing gear down and locked circuit breaker is engaged and if the bulb is good.

32.2.1.8 LANDING GEAR WARNING LIGHTS. Two bulbs in the landing gear control handle make the handle glow red when the gear is in an intermediate position; that is, when the gear is not down and locked with the handle in the "LG DOWN" position, or when the gear is not up and locked with the handle in the "LG UP" position. Electrical power is supplied to the bulbs by the 28-volt, essential DC bus through the landing gear down position indicator circuit breaker. The grounding circuit to the bulbs is the controlling circuit. During geardown actuation, the ground circuit can be completed through any of the following switches: the right main gear, downlock switch, the nose gear downlock switch, or the left main gear downlock switch. When these switches are not in the locked positions, a grounding circuit is completed to the warning light bulbs in the landing gear control handle. As the gear locks in the down position, these switches move to the locked position and break the grounding circuit. During gear-up actuation, the grounding circuit is completed through the nose gear door uplock switch, the nose gear uplock switch, the left and right main gear door uplock switches, the left main gear uplock relay (5001 thru 5040) and/or the left main gear uplock switch (5041 and up), and the right main gear uplock switch. To remove the ground signal from the left main gear uplock relay, it is necessary that the left main gear uplock switch be in the locked position to supply power to the solenoid of the relay. A HANDLE LIGHT TEST pushbutton on the gear control panel is used to test the bulbs in the gear control handle.

32.2.1.9 AUDIBLE GEAR WARNING. A pulsating tone is emitted to warn of one or more throttles being retarded below a position corresponding to 1.4 to 1.5 EPR with the landing gear not down and locked. The warning tone can be silenced by pressing the WARNING SILENCER pushbutton on the gear control panel.

a. On Airplane 5001 through 5065, the tone is heard on the cockpit speaker. The signal for the tone originates at an audible warning unit, and passes through a gear warning relay to the speaker. The audible warning unit and the gear warning relay both receive power from the landing gear warning circuit breaker. The relay is controlled through the grounding circuit to the solenoid by the same switches that control the warning lights in the gear control handle, but the circuit must go through one of the gear warning switches that are positioned by the throttles. The ground circuit to the solenoid of the gear warning relay controls the audible warning signal to the pilots' compartment speaker. At the same time the relay routes the warning signal to the speaker, it also isolates the speaker from other audio signals. The warning tone can be silenced by pressing the WARNING SILENCER pushbutton on the gear control panel.

b. On Airplanes 5066 and up, the tone is heard from the LANDING GEAR AND OXYGEN WARNING buzzer, located under the pilot's side console. The buzzer tone is steady for oxygen warning (refer to Section 35) and pulsed by a signal interrupter for landing gear warning. The interrupter, the landing gear warning relay, and the buzzer, through the interrupter and normally open contacts of the landing gear warning relay, receive 28-volt DC power from the landing gear warning circuit breaker. The relay and the interrupter are controlled by a grounding circuit through the same switches that control the warning lights in the gear control handle, but the circuit must go through one of the gear warning switches that are positioned by the throttles. The ground circuit to the solenoid of the gear warning relay controls the ground signal to the buzzer.

32.2.1.10 WARNING SILENCER CIRCUIT. The warning silencer circuit is made up of four gear signal disconnect (GSD) relays, a master disconnect relay, and a WARNING SILENCER pushbutton. Pressing the WARNING SILENCER pushbutton energizes the relay or relays, corresponding to the throttle switch or switches causing the warning tone, to break the grounding circuit to the solenoid of the gear warning relay. If the tone is due to throttle No. 1 being retarded, pressing the WARNING SILENCER pushbutton energizes the gear signal master disconnect relay to energize the No. 1 GSD relay and thus break the circuit to the gear warning relay providing a ground for the GSD relay when the pushbutton is released. If any of the other throttles are then retarded below the warning position, the warning tone will start again. Each time the pushbutton is pressed, the responsible relay or relays are energized, but the warning capability concerning the remaining throttles is not lost.

32.2.1.11 WARNING AND INDICATING LIGHT DIMMING CIRCUIT. The warning and indicating light

32.2.1.11 (Cont).

dimming relays No. 1 and No. 2 energize when the airplane position lights are turned on. Energizing the No. 1 relay routes the ground circuits for the indicating lights through resistors to dim the lights. Energizing the No. 2 relay routes the power circuit for the gear warning lights through a resistor to dim the warning lights in the control handle. Both of these relays are energized by power through the exterior position lights circuit breaker when the NAV POS LIGHT switch on the exterior lights control panel above the pilot is in the "ON" position.

32.2.1.12 GEAR WARNING RELAY. The solenoid of the gear warning relay receives its power from the landing gear warning circuit breaker. The grounding circuit to the solenoid controls relay operation. The same switches that control the gear warning lights in the landing gear control handle control the grounding circuit to the solenoid. The grounding circuit for

32.2.1.12 (Cont).

the gear warning relay must pass through at least one of the gear warning switches, and through its corresponding GSD relay. Energizing the gear warning relay causes the gear warning signal to sound.

32.2.2 OPERATIONAL CHECKOUT.

32.2.2.1 LANDING GEAR INDICATING AND WARNING SYSTEM. To check the landing gear indicating and warning system, perform an operational checkout of the landing gear actuating system according to the applicable instructions in Paragraph 32.0.2.

32.2.3 TROUBLESHOOTING

32.2.3.1 LANDING GEAR INDICATING AND WARNING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR LIGHT FAILS TO GO ON WHEN CORRESPONDING GEAR IS DOWN AND LOCKED.		
Defective or disengaged landing gear down and locked indicator lights circuit breaker.	Check for disengaged or defective circuit breaker.	Engage or replace circuit breaker as needed.
Defective bulb.	Press light housing to test bulb.	Replace defective bulb.
Open ground circuit from corresponding downlock switch through the warning and indicating light dimming relay No. 1.	Perform a continuity check of the system.	Repair the circuit as needed.
Defective or improperly adjusted downlock switch.	Check the operation and adjustment of the corresponding downlock switch.	Adjust the corresponding downlock switch as needed. Check all other possibilities before adjusting the main gear downlock switches, as the actuators must be removed to adjust them.
WARNING LIGHTS IN CONTROL HANDLE FAIL TO GO OUT AFTER GEAR ACTUATION CEASES AT THE GEAR DOWN-AND-LOCKED POSITION (GEAR CONTROL IN THE "LG DOWN" POSITION).		
Defective or improperly adjusted downlock switch for either the right main gear, the left main gear, or the nose gear.	Check the operation and the adjustment of the downlock switches.	Replace, repair, and/or adjust the switches as needed.
WARNING LIGHTS IN CONTROL HANDLE FAIL TO GO OUT AFTER GEAR ACTUATION CEASES AT THE GEAR-UP-AND-LOCKED POSITION (LANDING GEAR CONTROL IN THE "LG UP" POSITION).		
Defective left main gear uplock relay (airplanes 5001 thru 5040).	Check the operation of the relay.	Replace the relay as needed.
Defective or improperly adjusted gear or door uplock switch.	Check the operation and the adjustment of all gear and door uplock switches.	Repair, replace, and adjust switches as needed.

32.2.3.1 (Cont).

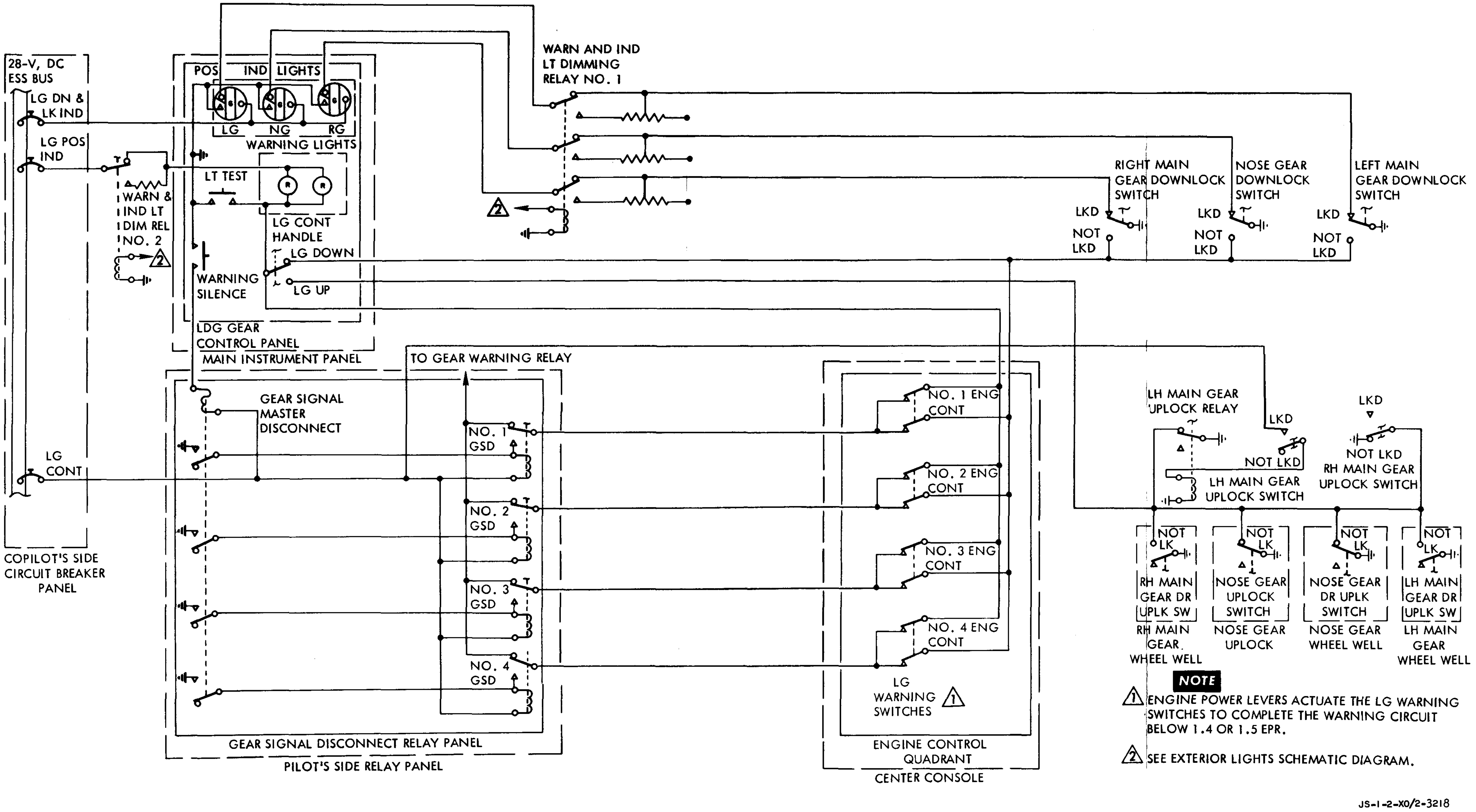
PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO WARNING TONE AUDIBLE WHEN THROTTLES ARE RETARDED BELOW A POSITION CORRESPONDING TO 1.4 TO 1.5 EPR WITH THE LANDING GEAR NOT DOWN AND LOCKED.		
Corresponding landing gear warning switch defective or improperly adjusted.	Check the adjustment and the operation of the switches in the throttle.	Adjust, repair, or replace the switches as needed.
Defective gear warning relay.	Check the operation of the relay.	Replace the relay as needed.
Defective audio warning unit. (5001 thru 5065).	Check operation of the unit.	Replace the defective unit.
Disengaged or defective landing gear warning circuit breaker.	Check for disengaged or defective circuit breaker.	Engage or replace circuit breaker as needed.
Defective interrupter (5066 and up).	Check operation of unit.	Replace interrupter.
Defective buzzer (5066 and up).	Check operation of buzzer.	Replace buzzer.
INDICATOR LIGHTS TOO BRIGHT AT NIGHT OR TOO DIM IN DAYLIGHT.		
Defective warning and indicating light dimming relay No. 1 or relay control circuit.	Check the brightness of the warning lights in the control handle. If both the indicators and the warning lights have the same trouble, check the airplane position lights circuit breaker and the relay control circuit. If only the indicator lights have the trouble, check the operation of the warning and indicating light dimming relay No. 1.	Replace defective relay or repair circuit as needed.
WARNING LIGHTS IN CONTROL HANDLE TOO BRIGHT AT NIGHT OR TOO DIM IN DAYLIGHT.		
Defective warning and indicating light dimming relay No. 2 or relay control circuit.	Check the brightness of the indicator lights on the control panel. If both the indicator and the warning lights have the same trouble, check the airplane position lights circuit breaker and the control circuit. If only the warning lights have the trouble, check the operation of the warning and indicating light dimming relay No. 2.	Replace defective relay or repair circuit as needed.

32.2.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
WARNING LIGHTS IN GEAR CONTROL HANDLE FAIL TO GO ON WHEN GEAR IS MOVING TO A DIFFERENT POSITION.		
Disengaged or defective landing gear position indicator circuit breaker.	Check for disengaged or defective circuit breaker.	Replace or engage circuit breaker as needed.
Defective bulbs.	Test the bulbs, using the button on the control panel.	Replace defective bulbs.
WARNING TONE DOES NOT CEASE WHEN THE WARNING SILENCE BUTTON IS PRESSED.		
Defective gear signal disconnect relay.	Check the operation of the relays.	Replace relays as needed.

32.2.4 ADJUSTMENT.

32.2.4.1 LANDING GEAR AND DOOR SWITCH ADJUSTMENT. For these instructions, refer to figure 32-8.

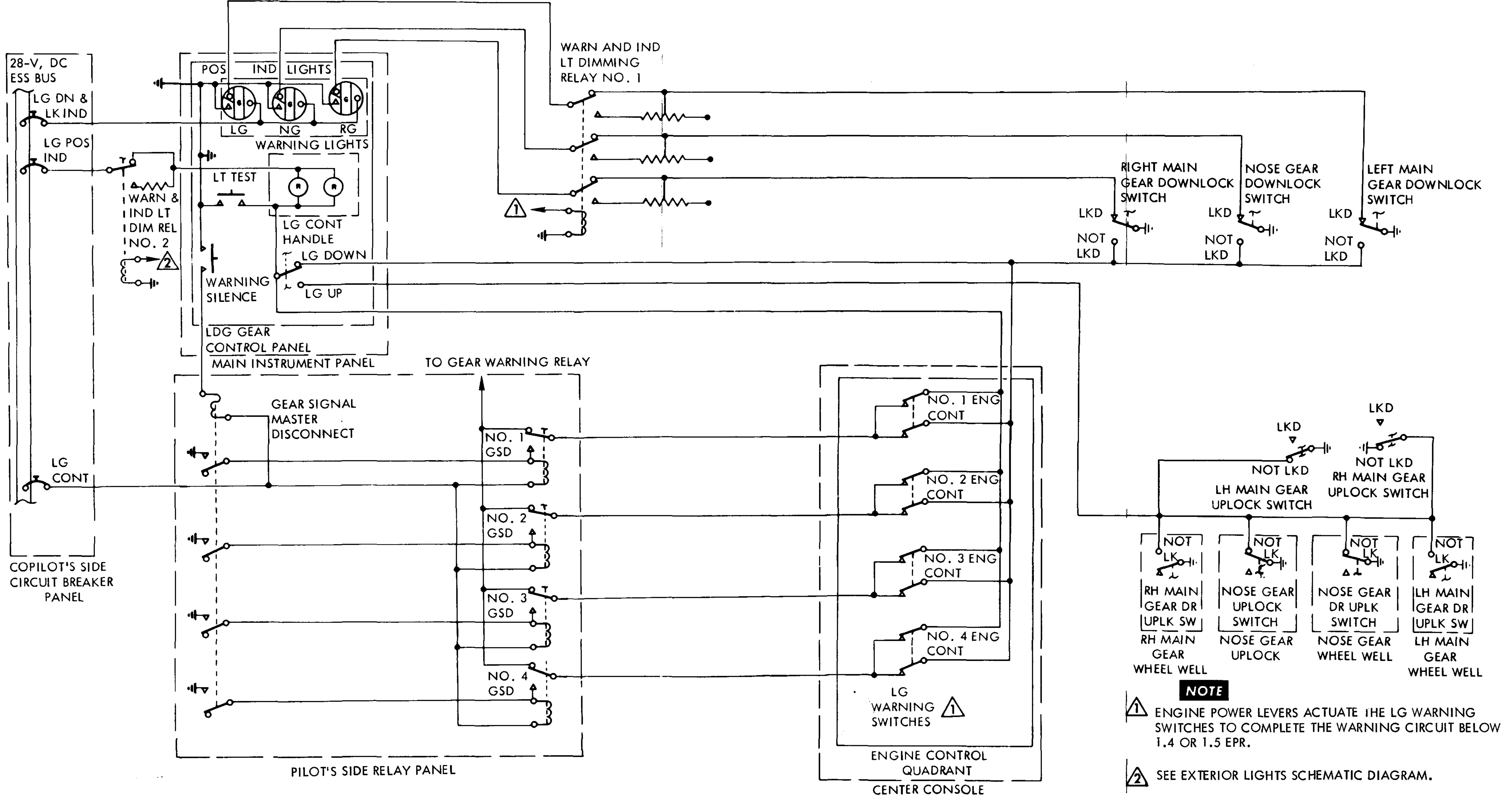


NOTE

1 ENGINE POWER LEVERS ACTUATE THE LG WARNING SWITCHES TO COMPLETE THE WARNING CIRCUIT BELOW 1.4 OR 1.5 EPR.

2 SEE EXTERIOR LIGHTS SCHEMATIC DIAGRAM.

Figure 32-18. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5001 thru 5040)



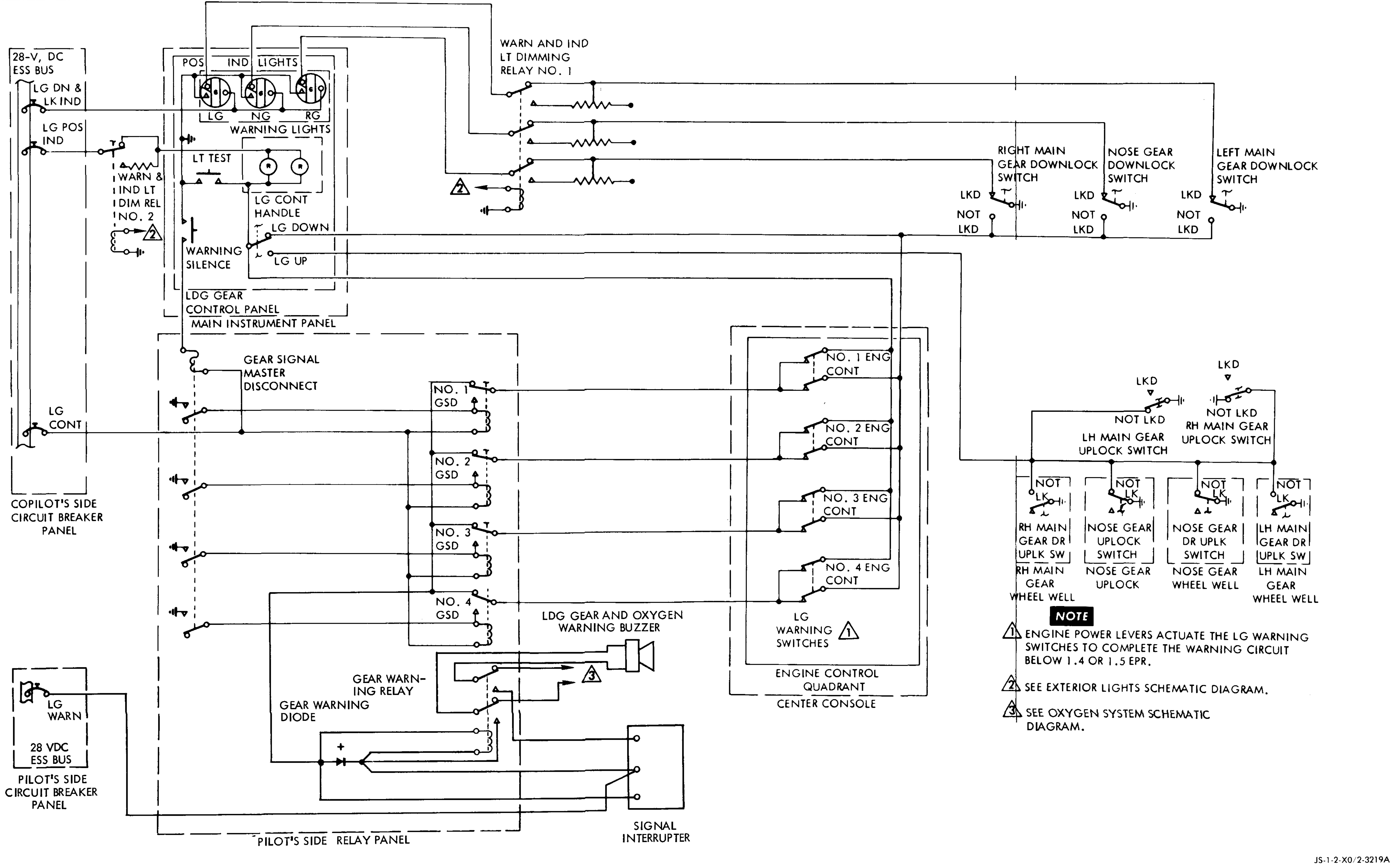
NOTE

⚠1 ENGINE POWER LEVERS ACTUATE THE LG WARNING SWITCHES TO COMPLETE THE WARNING CIRCUIT BELOW 1.4 OR 1.5 EPR.

⚠2 SEE EXTERIOR LIGHTS SCHEMATIC DIAGRAM.

Figure 32-19. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5041 thru 5065)

Changed 31 August 1983



NOTE

1. ENGINE POWER LEVERS ACTUATE THE LG WARNING SWITCHES TO COMPLETE THE WARNING CIRCUIT BELOW 1.4 OR 1.5 EPR.
2. SEE EXTERIOR LIGHTS SCHEMATIC DIAGRAM.
3. SEE OXYGEN SYSTEM SCHEMATIC DIAGRAM.

Figure 32-19A. Landing Gear Indicating and Warning System Schematic Diagram (Airplanes 5066 and Up)

Changed 31 August 1983

SECTION 32.3
BRAKE AND ANTI-SKID SYSTEMS

Contents Page

DESCRIPTION AND OPERATION 32-74
  General 32-74
  Brake System 32-74
  Parking Brakes 32-74
  Anti-Skid System 32-74
  Brake System Operation 32-74
  Anti-Skid System Operation 32-74
  Control Switches 32-75
  Brake Pedals 32-75
  Parking Brake Tee Handle 32-75
  Dual-Brake Control Valves 32-76
  Brake Assemblies 32-76
  Brake Accumulator 32-76
  Shuttle Valves 32-76
  Hydraulic Fuses 32-76
  Brake Selector Valves 32-76
  Low-Pressure Warning Switches 32-76
  Anti-Skid Control Boxes 32-76
  Anti-Skid Valves 32-76A
  Skid Detectors 32-76A
  Anti-Skid Out Warning Light 32-76A
  Landing Gear Isolation Valve (5041 and Up) 32-76A
  Check Valve 32-76A
  Snubber Assemblies 33-76A

OPERATIONAL CHECKOUT 33-76A
  Wheel Brake System 32-76A
  Anti-Skid System (Airplanes 5001 through 5092 and 5094 through 5096) 32-78
  Skid Detector Polarity Check 32-79
  Quick Check of Anti-Skid System (Airplanes 5001 through 5092 and 5094 through 5096) 32-80
  Anti-Skid System (Airplanes 5093, 5097, and Up) 32-80

TROUBLESHOOTING 32-81
  Wheel Brake System 32-81
  Anti-Skid System 32-81

REMOVAL AND REPLACEMENT 32-82
  Removal of Wheel Brake Assembly 32-82
  Replacement of Wheel Brake Assembly 32-82
  Removal of Dual-Brake Control Valves 32-82A
  Replacement of Dual-Brake Control Valves 32-82A

ADJUSTMENT 32-82A
  Dual-Brake Control Valve 32-82A

INSPECTION 32-82A
  Inspecting Brake Lining Wear 32-82A

FIGURES

  32-20. Wheel Brake and Anti-Skid System Schematic Diagram (Airplanes 5001 through 5092 and 5094 through 5096) 32-82C
  32-20A. Wheel Brake and Anti-Skid System Schematic Diagram (Airplanes 5093, 5097 and UP) 32-83
  32-21. Wheel Brake Assembly 5093, 5097 and Up 32-84A
  32-21A. Wheel Brake Assembly 5001 thru 5092, 5094 thru 5096 32-85
  32-22. Dual-Brake Control Valve Adjustment 32-86

32.3.1 DESCRIPTION AND OPERATION.

32.3.1.1 GENERAL. (See figures 32-20, 32-20A, and 32-21.) Airplanes have four multiple disk brakes, one on each main wheel. During normal brake operation, the normal hydraulic system powers the brake system; but the auxiliary hydraulic pump can be used independently of the normal system pump to power the brake system. If neither the normal, nor the auxiliary hydraulic pumps are operable, the standby hydraulic system can be selected to power the brake system; but when the standby system is used, the anti-skid system is inoperative. Parking brakes are mechanical latch mechanisms which hold the brake pedal linkages in the brakes-on position to lock pressure in the brake assemblies.

a. Anti-skid system (Airplanes 5001 through 5092 and 5094 through 5096). The anti-skid system operates by relieving hydraulic pressure from the brake assemblies of skidding wheels while maintaining brake pressure on the other brake assemblies. Anti-skid circuits are controlled by an ANTI-SKID switch. When the anti-skid system is inoperative, an ANTI-SKID OUT LIGHT on the annunciator warning panel goes on.

b. Airplanes 5093, 5097, and up have a fully modulating anti-skid system. When this system detects an incipient skid on any wheel, it reduces brake pressure to that wheel proportional to its deviation from the normal braking deceleration.

32.3.1.2 BRAKE SYSTEM. (See figures 32-20 and 32-20A.) The brake system contains four brake pedals, two dual-brake control valves, four wheel brake assemblies, four shuttle valves, four hydraulic fuses, a No. 1 system brake selector valve, a No. 2 system brake selector valve, two low pressure warning switches, one BRAKE PRESS LOW warning light, a brake system accumulator, and associated circuitry.

32.3.1.3 PARKING BRAKES. A PARK BRAKE tee handle on the main instrument panel controls catches on the pilot's brake pedal linkages. Applying the brakes and pulling the tee handle positions the catches to engage with two pawls attached to the airplane structure to hold the brake pedals in the brakes-on position. Reapplying the brakes releases the PARK BRAKE tee handle and releases the brakes. As there are two separate catches on the pilot's pedal linkages, one on the left and one on the right, it is possible to set the parking brakes on one side only. This condition should be obvious by noting the relative positions of the brake pedals after the parking brakes are set.

32.3.1.4 ANTI-SKID SYSTEM (AIRPLANES 5001 THROUGH 5092, AND 5094 THROUGH 5096). (See figure 32-20.) The anti-skid system contains two anti-skid control boxes, four anti-skid valves, four skid detectors, an ANTI-SKID OUT warning light, and an ANTI-SKID control switch. These components operate in conjunction with the normal brake system to provide skid protection.

a. (Airplanes 5093, 5097, and up) (See figure 32-20A). The anti-skid system contains a control box, two dual anti-skid valves, four wheel velocity transducers, a system control switch, and an anti-skid panel containing a test switch and four indicator lights.

32.3.1.5 BRAKE SYSTEM OPERATION. (See figures 32-20 and 32-20A.) During normal brake operation, hydraulic pressure passes through the landing gear selector valve, and through a check valve, to the brake system accumulators and the No. 1 brake selector valve, and to the two dual-brake control valves. From the dual-brake control valves, the fluid goes through the anti-skid valves, through the brake shuttle valves, and on to the brake assemblies. During standby brake operation, hydraulic pressure passes through the No. 2 brake selector valve, through the dual-brake control valves, through hydraulic fuses, through the shuttle valves, and on to the brake assemblies. The hydraulic fuses prevent the loss of all standby system fluid if there is uncontrolled leakage downstream of the fuses. These fuses normally allow unrestricted flow of approximately 10 cubic inches of fluid. Normally, the brake assemblies are pressurized and the fuse remains open. If a leak develops downstream of the fuse, the first 10 cubic inches of fluid are not enough to equalize pressure on both sides of the fuse, so the fuse actuates to block the passage of additional fluid. After a fuse blocks fluid passage, releasing the brake pedal to relieve pressure on the input port of the fuse allows the pressure to equalize and the fuse to reset itself.

a. On airplanes 5041 and up, a landing gear isolation valve permits nose steering and main wheel brake operation even though the landing gear selector valve is not in the gear down position. This valve is in the left wheel well area, and is manually controlled by an LG ISOLATION VALVE tee handle in front of the left wheel well box inside the airplane. Pulling this tee handle positions the valve to route normal hydraulic power to the nose steering and wheel brake systems regardless of the position of the landing gear selector valve. This valve also opens the gear actuating system to the normal system return lines for assistance in releasing the landing gear during emergencies.

b. Airplanes 5041 and up have a spring detent in the landing gear selector valve assembly to hold the valve in the gear-down position to insure brake and steering pressure even though the gear control circuit is de-energized. The spring detent automatically releases when the valve is energized for gear up operation.

32.3.1.6 ANTI-SKID SYSTEM OPERATION (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096). (See figure 32-20.) The anti-skid system contains four skid detectors. Each detector is a permanent magnet type, DC generator and is driven by the hub of its respective wheel. Each detector's output is balanced with the discharge of a capacitor. When a wheel

32.3.1.6 (Cont).

approaches a skid, the output of its detector drops suddenly, causing a higher discharge of the capacitor. This action energizes the anti-skid valve for the skidding wheel. Energizing the anti-skid valve not only blocks hydraulic pressure from the brake assembly, but also relieves the pressure within the brake assembly to release the brake on the skidding wheel. As the wheel regains rotational speed, the anti-skid valve returns braking power to the wheel.

a. If a skid signal exceeds approximately 2.7 seconds, the anti-skid control box circuit receiving the signal de-energizes to return full manual braking control to the wheel with which that circuit is concerned. When this happens, the remaining wheels still have anti-skid protection although the ANTI-SKID OUT light on the annunciator warning panel is on. After an ANTI-SKID OUT warning, the control switch must be switched off then on again to reset the anti-skid system.

32.3.1.7 ANTI-SKID SYSTEM OPERATION (AIRPLANES 5093, 5097, AND UP). (See figure 32-20A.) The wheel velocity transducers are mounted in the wheel axles and are driven by the hub caps. The information inputs to the control box from the transducers are frequency modulated signals which continuously provide an instantaneous measure of the velocity of each wheel. The control box operates from this information to provide fully modulated brake pressure control including four adaptive control loops of velocity threshold, rate threshold, pressure bias modulation, and proportioning control. The control box also continuously monitors the transducers, valves and wiring continuity, and partial or complete loss of system power. The control box contains touchdown protection circuits, through the squat switches, to insure that the pilot cannot land with wheels braked.

a. (Airplanes 5093, 5097, and up). Malfunctions are indicated by four monitoring lights, one for each wheel, on the Anti-Skid Panel and by the ANTI-SKID OUT light on the annunciator panel. Any malfunction that causes brake pressure to be withheld from any wheel for three seconds will cause the circuits for that wheel to go fail-safe and revert the associated wheel to manual braking. The monitor light for that wheel and the ANTI-SKID OUT light will illuminate. The other wheels will still have anti-skid protection. To reset any circuits that have gone fail-safe, the anti-skid system switch must be switched off and then on again.

b. (Airplanes 5093, 5097, and up). The anti-skid system incorporates a self-contained test and checkout circuit. The Anti-Skid Panel located on the copilot's forward wainscot, contains in addition to the four wheel monitor lights, a three-position INBOARD-OFF-OUTBOARD test switch. When the test switch is placed in the INBOARD position, the inboard indicator lights and the annunciator panel ANTI-SKID OUT lights illuminate; this indicates a successful test. The switch returns to OFF when released and the lights extinguish if no malfunction exists. If a fault is present, the annunciator panel light and the affected wheel light will remain illuminated. When the test switch is placed in the OUTBOARD position the results are the same except the outboard indicator lights illuminate. When the anti-skid system control switch is in the OFF position only the ANTI-SKID OUT annunciator panel light is illuminated.

c. At low rotational speeds of the wheels (10 to 20 miles per hour airplane speed) the anti-skid system is inoperative.

32.3.1.8 CONTROL SWITCHES. (See figures 32-20 and 32-20A.) A HYD SYS NORM-1 STANDBY-2 switch on the copilot's side of the main instrument panel controls hydraulic system selection for both the wing flaps and the wheel brakes (and steering, on airplanes 5046 and up). Placing the switch in the "NORM-1" position selects the normal hydraulic system; and if the ANTI-SKID switch on the pilot's control column cover is in the "ON" position, it also completes a circuit from the landing gear anti-skid circuit breaker to the anti-skid system. Placing the HYD SYS NORM-1 STANDBY-2 switch in the "STANDBY-2" position breaks the anti-skid power circuit, and selects the standby hydraulic system to operate the brakes. A two-position ("ON," "OFF") ANTI-SKID switch controls power from the landing gear anti-skid circuit breaker to the anti-skid contacts of the HYD SYS NORM-1 STANDBY-2 switch.

32.3.1.9 BRAKE PEDALS. There are two sets of brake pedals in the airplanes: a left and a right for the pilot, and a left and a right for the copilot. Pivoting the left pedal of either set positions linkage to the dual-brake control valve for the brakes on the left gear, and pivoting the right pedal of either set positions linkage to the dual-brake control valve for the brakes on the right gear. The dual-brake control valve for the left brakes is under the copilot's flooring; the dual-brake control valve for the right brakes is under the pilot's flooring. Pedal adjustment for both the pilot and copilot is provided by means of RUDDER PEDAL ADJUST knobs on the control column covers.

32.3.1.10 PARKING BRAKE TEE HANDLE. A PARK BRAKE tee handle controls catches on the pilot's brake pedal linkages. Pulling the handle when the wheel brakes are applied positions the catches to engage with pawls attached to the airplane structure to hold the pedals in the brakes-on position. Reapplying the brakes when the parking brakes are set releases the catches and returns the PARK BRAKE handle to the normal (in) position. The PARK BRAKE tee handle is on the main instrument panel, left of the pilot's control column.

32.3.1.10 (Cont).

CAUTION

When releasing the parking brakes prior to taxiing or towing, ensure that both right and left brakes are released. The normal release procedure is to pivot both rudder pedals hard forward until the parking brake handle springs back to the released position. Do not attempt to push the parking brake handle in until it is certain that the parking brake catches have released.

32.3.1.11 DUAL-BRAKE CONTROL VALVES. There are two dual-brake control valves, one for the left brakes, and one for the right brakes. The valve for the left brakes is under the copilot's flooring, and the valve for the right brakes is under the pilot's flooring. Each valve is a dual unit with two separate fluid compartments. One compartment is used for normal brake operation, and the other compartment is used for standby brake operation. Depressing a brake pedal exerts pressure on the plungers for both compartments of the valve, and the operation of both compartments is identical. Depressing the plunger in either compartment exerts pressure on a spring which opens the pressure outlet of the compartment. As pressure within the brake assemblies rises to a point where the back pressure also acts upon the pressure outlet, this back pressure acts upon the springs to allow the pilots to select the proper amount of braking.

32.3.1.12 BRAKE ASSEMBLIES. (See figures 32-21 and 32-21A.) Each wheel brake assembly is a hydraulically operated, three-rotor, self-adjusting unit consisting of four non-rotating disks separated by three disks keyed to rotate with the wheel. On the rotating disks, full-circle lining is sintered on the sides adjacent to the non-rotating disks. Twelve pistons apply a force to a piston plate. (On airplanes 5093, 5097 and up, eight pistons are used with four rotors and three non-rotating disks.) These pistons have O-ring seals which retain hydraulic fluid within the fluid chambers. A full-circle insulator between the pressure plate and the piston housing obstructs the transfer of heat from the brake heat sink to the piston housing. The force of the piston applied through the piston plate forces the rotating and non-rotating disks together to supply friction for braking action.

32.3.1.13 BRAKE ACCUMUATOR. A 50-cubic inch, piston-type accumulator in the brake system minimizes surges in the system. A check valve in the system prevents pressure in the accumulator from bleeding back into the normal hydraulic system. This accumulator has an air filler valve and a pressure gage to facilitate servicing. The brake system accumulator is in the nose wheel well area.

32.3.1.14 SHUTTLE VALVES. One shuttle valve for each wheel brake assembly shifts back and forth to permit brake operation from either the normal or the standby hydraulic system. These shuttle valves shift under pressure to admit hydraulic pressure from one system while blocking the other system lines. These valves are in the brake system lines, and are on the brake system hydraulic panel under the center flooring between fuselage stations 357 and 370.

32.3.1.15 HYDRAULIC FUSES. Four hydraulic fuses, one in each standby system brake pressure line, prevent standby system fluid from escaping from any uncontrolled leak downstream of the fuses. Each fuse normally allows 10 cubic inches of fluid to pass through it before any force is exerted on the shutoff mechanism. Normally, this is enough to pressurize the brake assembly to the desired pressure. If an uncontrolled leak should occur downstream of the fuse, more than 10 cubic inches would be required as the brakes would not pressurize to the desired degree. This additional fluid would force the shutoff mechanism to halt the passage of additional fluid through the fuse. After a fuse has shutoff fluid flow, releasing the brakes to equalize the input and output port pressures resets the fuse. These fuses are under the center flooring between fuselage stations 357 and 370.

32.3.1.16 BRAKE SELECTOR VALVES. There are two brake selector valves, a No. 1 system selector valve, and a No. 2 system selector valve. When using normal hydraulic system power to operate the brakes, the No. 1 selector valve is de-energized to admit normal system pressure to the normal side of the dual-brake control valves; the No. 2 system selector valve is energized to block standby system pressure from the brake system and to open the standby compartments of the dual-brake control valves to standby return. When using the standby system to power the brakes, the No. 1 system selector valve is energized to block the normal system pressure from the brake system and to open the normal compartments of the dual-brake control valves to the system return side; and the No. 2 system selector valve is de-energized to admit standby system power to the brake system. The brake selector valves are in the nose wheel well.

32.3.1.17 LOW-PRESSURE WARNING SWITCHES. One low-pressure warning switch in the normal brake line and one in the standby system brake lines control a single BRAKE PRESS LOW warning light on the annunciator warning panel. The HYD SYS NORM-1 STANDBY-2 switch parallels system operation and low-pressure warning light switch control; that is, only the warning light in the operating system is given continuity to the warning light. Each switch is set to make the warning light go on at 1,250 PSI minimum and go out at 1,500 PSI maximum. Both low-pressure warning switches are in the nose wheel well.

32.3.1.18 ANTI-SKID CONTROL BOXES.

a. (Airplanes 5001 through 5092 and 5094 through 5096.) There are two anti-skid control boxes, one for the inboard wheel brakes, and one for the outboard wheel

32.3.1.18 (Cont).

brakes. These boxes are inside the lower right wing-to-fuselage fairing between fuselage stations 450 and 470. Each control box contains capacitors, relays, rectifiers, resistors, and transistors in circuits whose function it is to energize the anti-skid valves in the brake lines of skidding wheels. A fail-safe circuit within the control boxes insures braking control by returning full brake control to the pilots after a skid signal continues for over 2.7 seconds. The ANTI-SKID OUT warning light on the annunciator warning panel goes on whenever either control box is inoperative.

b. (Airplanes 5093, 5097, and up). The anti-skid control box is located in the floor well between fuselage stations 330 and 350. Access to the unit may be obtained by lifting the walkway forward panels (see Figure 53-2). The control box contains solid state circuits to control pressure to the brakes, detect malfunctions, test the system, and revert the wheels to manual braking if pressure is withheld for three seconds. When a circuit is inoperative, the monitor light for the associated wheel and the ANTI-SKID OUT light on the annunciator panel illuminate.

32.3.1.19 ANTI-SKID VALVES. One anti-skid control valve in the brake pressure line to each brake assembly controls brake pressure according to signals from the control boxes. When the valve is energized, it not only blocks hydraulic pressure from the brake assembly, but also relieves the pressure within the brake system. The anti-skid valves are under the center flooring between fuselage stations 357 and 370.

32.3.1.20. SKID DETECTORS.

a. (Airplanes 5001 through 5092 and 5094 through 5096). One skid detector on each wheel is driven by the hub of that wheel. Each skid detector is a small permanent-magnet-type DC generator whose voltage increases as wheel speed increases. The voltage generated by these wheel skid detectors control the anti-skid circuits within the control boxes. As inboard and outboard wheels rotate in different directions in relation to the axle, skid detectors are wired differently for these wheels.

b. (Airplanes 5093, 5097, and up.) One skid detector on each wheel is driven by the hub cap of that wheel. Each detector is a non-polarized variable reluctance transducer whose output is an AC signal whose frequency is proportional to wheel velocity without a corresponding amplitude change.

32.3.1.21 ANTI-SKID OUT WARNING LIGHT. An ANTI-SKID OUT warning light on the annunciator warning panel goes on when the anti-skid system is inoperative, whether it is due to the ANTI-SKID switch being in the "OFF" position, or whether it is due to the fail-safe circuit in either of the control boxes. It is possible to have the ANTI-SKID OUT light on with a portion of the anti-skid system still operating.

a. (Airplanes 5093, 5097, and up.) In addition to the warning light on the annunciator panel there is an indicator light for each wheel on the Anti-Skid Panel. If a malfunction causes a fail-safe circuit to operate, the indicator light for the associated wheel illuminates as well as the annunciator panel light. These lights are also used for testing the anti-skid system (see paragraph 32.3.1.7b). When the anti-skid system switch is in the OFF position, only the annunciator panel light is illuminated.

32.3.1.22 LANDING GEAR ISOLATION VALVE. Airplanes 5041 and up, and airplanes having Service Bulletin 329-25 incorporated, have a manually controllable landing gear isolation valve. This valve is above the flooring forward of the left wheel well, and is operated by an LG ISOLATION VALVE tee handle on the front of the left wheel box inside the airplane. Pulling this tee handle positions the valve to bypass the landing gear selector valve to route hydraulic power to the nose steering and main wheel brake systems, and it also opens the landing gear actuating system to return. Actuating this valve ensures nose steering and main wheel brake pressure even though the landing gear selector valve may not be in the gear down position.

32.3.1.23 CHECK VALVE. A check valve is installed in the return line plumbing and positioned so as to protect the brake system from transient surges in the normal (No. 1) system return lines.

32.3.1.24 SNUBBER ASSEMBLIES. Airplanes 5073 and up, and airplanes having Service Bulletin 329-225 incorporated, have a snubber assembly installed upstream of each brake low pressure warning switch. The purpose of the snubber is to protect the switch mechanism from system pressure surges.

32.3.2 OPERATIONAL CHECKOUT.

32.3.2.1 WHEEL BRAKE SYSTEM.

a. Make sure that the following circuit breakers are engaged: landing gear brakes, system No. 1 hydraulic pressure, system No. 2 hydraulic pressure, warning lights, landing gear control, and main inverter control (5001 thru 5009) or No. 1 main inverter control (5010 and up).

b. Precharge the normal hydraulic system accumulator, the standby hydraulic system accumulator, and the brake system accumulator as needed according to the instructions in Chapter 12.

32.3.2.1 (Cont).

c. Service the normal and standby system reservoirs as needed according to the instructions in Chapter 12.

d. Install a hydraulic pressure gage capable of measuring up to 2000 PSI in each brake assembly bleed port.

e. Install an air pressure gage in the air filler valve of the brake system accumulator. This gage should be capable of measuring up to 3,000 PSI in increments of 100 PSI.

f. Make sure that the HYD SYS NORM-1 STANDBY-2 switch is in the "NORM-1" position.

g. Install a rig pin in the rudder pedals to maintain the pedals in the neutral position.

h. Make sure that the landing gear control handle is in the "LG DOWN" position and that the landing gear ground safety locks are installed on all gears.

i. Make sure that the wheels are chocked and that the parking brakes are released.

j. Connect 28-volt, DC power to the airplane service connections.

k. Connect a source of hydraulic power to the normal hydraulic system ground test connections, and adjust it to deliver 3 gallons per minute at 2,700 PSI, and no flow at 3,000 PSI.

l. Apply 28-volt, DC power to the airplane and place the main inverter switch (5001 thru 5009) or the No. 1 main inverter switch (5010 and up) in the "ON" position with the AC SY SW in the "NORMAL" position. Observe the BRAKE PRESS LOW warning light on the annunciator warning panel.
RESULT: The BRAKE PRESS LOW warning light should go on if the brake system pressure is below 1,100 PSI.

m. Slowly apply 3,000 PSI to the normal hydraulic system, and observe the BRAKE PRESS LOW warning light and the pressure gages in the brake assemblies.
RESULT: If the BRAKE PRESS LOW warning light was on in step (l), it should go out. The brake pressure gages should show no more than 10 PSI.

n. With 3,000 PSI on the normal hydraulic system, observe the pointers of the dual-pressure indicator on the copilot's side of the main instrument panel.
RESULT: The No. 1 pointer should show 3,000 (±150) PSI, and the No. 2 pointer should show 0 (±150) PSI.

o. Depress the pilot's left brake pedal until 325 PSI is indicated on the gages in the left brake assemblies, and observe the pressure gages in the bleed port of the right brakes.
RESULT: The gages on the right brakes should not show over 10 PSI.

p. Depress the left brake pedal until the locknuts on the metering valve contact the valve body.
RESULT: The left brake pressure gages should read 775 to 825 PSI for airplanes 5001 thru 5092 and 5094 thru 5096. For airplanes 5093 and 5097 and up, gages should read 1400 PSI minimum.

q. Release the left brakes and try to move the brake disks on each left wheel.
RESULT: The brake disks should have some in-and-out clearance along the axle.

r. Repeat steps (o) through (q), using the pilot's right brake pedal.

s. With the brakes released, observe the pressure gages in the brake assemblies.
RESULT: None of the gages should show more than 10-PSI brake pressure.

t. Repeat steps (o) through (r) using the copilot's brake pedals.

u. Apply the pilot's brakes and set the parking brakes by pulling the PARK BRAKE tee handle and then releasing the brakes. Observe the brake pressure gages.
RESULT: The pressure gages should show 420-520 PSI airplanes 5001 thru 5092, and 5094 thru 5096 725-975 PSI airplanes 5093 and 5097 and up.

v. Apply, then release the brakes without touching the PARK BRAKE tee handle.
RESULT: The PARK BRAKE tee handle should return to the normal (in) position with no assistance, and the brakes should release.

w. Remove the rig pin installed in step (g) and, without applying the brakes, apply full left and full right rudder while observing the gages in the brake assemblies.
RESULT: At no time should rudder operation affect the brake pressure.

x. Set the parking brakes and allow the wheel brake pressures to stabilize.

y. Record the brake system accumulator pressure as shown on the gage installed in the air filler valve and then after a 10 minute period, record the pressure again.
RESULT: During the 10 minute period, the accumulator pressure should not drop more than 250 PSI.

z. Place the HYD SYS NORM-1 STANDBY-2 switch in the "STANDBY-2" position and then back to the

32.3.2.1 (Cont).

"NORM 1" position while observing the pressure gages in the brake assemblies.
RESULT: The pressures should drop when the switch is placed in the "STANDBY-2" position, and should rise again when the switch is returned to the "NORM-1" position.

aa. Release the parking brakes and reduce the output of the external hydraulic power source to zero.

ab. Slowly reduce hydraulic pressure in the normal hydraulic system by operating the brakes. Observe the BRAKE PRESS LOW warning light and the gage in the air filler valve of the accumulator.
RESULT: The BRAKE PRESS LOW warning light should go on before the gage in the air filler valve of the accumulator reaches 1,100 PSI.

ac. With the normal hydraulic system pressure at zero, disconnect the hydraulic power source from the normal system ground test connections and connect it to the standby system ground test connections.

ad. Apply 3,000 PSI to the standby hydraulic system, and observe the dual-pressure indicator on the copilot's side of the main instrument panel.
RESULT: With 3,000 PSI on the standby system, the No. 1 pointer should show 0 (±150) PSI, and the No. 2 pointer should show 3,000 (±150) PSI.

ae. Place the HYD SYS NORM-1 STANDBY-2 switch in the "STANDBY-2" position, and observe the BRAKE PRESS LOW warning light on the annunciator warning panel.
RESULT: The BRAKE PRESS LOW warning light should go out as the HYD SYS NORM-1 STANDBY-2 switch is placed in the "STANDBY-2" position.

af. Repeat steps (n) through (r) using the standby hydraulic system to power the brake system.

ag. While maintaining 3,000 PSI on the standby hydraulic system, set the parking brakes and visually inspect the entire brake system for evidence of leakage.
RESULT: There should be no evidence of external leakage from any tube, connection, or valve in the entire brake system.

ah. Move the HYD SYS NORM-1 STANDBY-2 switch from "STANDBY-2" to "NORM-1," and then back to "STANDBY-2." Observe the brake pressure gages in the brake assemblies and the BRAKE PRESS LOW warning light.
RESULT: When the switch is placed in the "NORM-1" position, the BRAKE PRESS LOW warning light should go on, and the brake pressure gages in the brake assemblies should show a drop in pressure. When the switch is returned to the "STANDBY-2" position, the brake pressure should be reapplied, and the BRAKE PRESS LOW warning light should go out.

ai. Reduce the output of the external hydraulic power source to zero, and then slowly reduce the standby system pressure to zero while observing the BRAKE PRESS LOW warning light and the dual-pressure indicator on the copilot's side of the main instrument panel.
RESULT: The BRAKE PRESS LOW warning light should go on before the No. 2 pointer of the dual-pressure indicator shows 1,100 PSI.

aj. Remove all test equipment and restore the brake system to normal. If needed, bleed the brake system according to the instructions in Chapter 29.

32.3.2.2 ANTI-SKID SYSTEM (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096). To drive the skid detector in this checkout, obtain a drill motor (or similar device) and attach a rubber tube approximately 1/4 inch in inside diameter and 2 inches in length to the motor drive. Cut a slot in the end of the tube approximately 1/8-inch wide by 1/2-inch deep to fit the drive of the skid detector.

a. Connect external electrical power to the airplane.

b. Connect external hydraulic power to the normal hydraulic system and adjust the power source to deliver zero gallons per minute at 3,000 PSI and 10 gallons per minute at 2,700 PSI.

c. Install a pressure gage and adapter on each brake assembly bleed port.

d. Remove the main wheel hub caps so that the skid detector drives can be rotated.

e. Engage the following circuit breakers: landing gear anti-skid, landing gear brakes, warning lights, and landing gear control.
RESULT: With these circuit breakers engaged and power on the airplane, the ANTI-SKID OUT light should go on when the ANTI-SKID switch is in the "OFF" position.

f. Apply hydraulic pressure to the normal hydraulic system.

g. Be sure the HYD SYS NORM-1 STANDBY-2 switch is in the "NORM-1" position, and then place the ANTI-SKID switch on the copilot's control column in the "ON" position.
RESULT: The ANTI-SKID OUT warning light should go out.

h. Apply the brakes and observe the brake pressure gages in the bleed ports of the brake assemblies.
RESULT: All of the brake pressure gages should indicate brake pressure.

32.3.2.2 (Cont).

i. With the brakes still applied, disengage the landing gear anti-skid circuit breaker and observe the ANTI-SKID OUT warning light.
RESULT: The ANTI-SKID OUT light should go on when the circuit breaker is disengaged.

j. Engage the landing gear anti-skid circuit breaker and observe the ANTI-SKID OUT warning light.
RESULT: The warning light should go out.

k. Release the brakes.

Note

The ANTI-SKID OUT warning light should not go on during steps (l) and (m).

l. Rotate the left outboard skid detector at 150 RPM or higher in the direction of forward wheel rotation, and then apply the brakes and suddenly stop the skid detector rotation.
RESULT: The left outboard anti-skid valve should energize as soon as the skid detector rotation is stopped. This should be evidenced by a drop in left outboard brake pressure. Other brake pressures should not be affected.

m. With the brakes still applied, and within 2 seconds after the skid detector has stopped, start rotating the skid detector at approximately 150 RPM again. Observe the pressure gages in the brake assembly bleed ports.
RESULT: The left outboard brake pressure gage should show an increase in pressure and should equalize with the pressures on the other brakes.

n. Repeat steps (l) and (m) for the left inboard skid detector. Results should be the same except that the left inboard brake pressure should be affected.

o. Repeat step (l) through (n) for the right anti-skid detectors. Results for the right brakes should be the same as those for the left brakes though for different wheels.

Note

It is not necessary to keep the brakes applied during steps (p) through (r).

p. Remove the electrical connector from the left outboard wheel anti-skid control valve.
RESULT: After approximately 2.7 seconds the ANTI-SKID OUT warning light should go on.

q. Reconnect the electrical connector for the left outboard wheel anti-skid control valve, and reset the system by first placing the ANTI-SKID switch in the "OFF" position and then back to the "ON" position.
RESULT: The ANTI-SKID OUT light should go off.

r. Repeat steps (p) and (q) for the left inboard, right outboard, and right inboard anti-skid control valves.
RESULT: Same as step (q).

s. Release the brakes, reduce hydraulic pressure to zero, and then cycle the ailerons or elevators to reduce pressure in the normal hydraulic system to zero. Operate the wheel brakes until the pressure in the wheel brake system is zero.

t. With no pressure in the brake system, remove the adapters and pressure gages from the bleed ports of the brake assemblies, and return the brake and anti-skid systems to normal.

32.3.2.3 SKID DETECTOR POLARITY CHECK (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096).

CAUTION

Do not disassemble the skid detectors. To do so may disturb the magnetization of the detectors.

a. With the ANTI-SKID switch in the "OFF" position, disconnect the electrical connectors from both the inboard and the outboard skid control boxes.

b. Remove the skid detector drive assemblies to enable the detector drives to be rotated by hand.

c. Rotate the right outboard skid detector drive in the direction of normal wheel rotation, and check the polarity of pins D and G of the outboard control box connector.
RESULT: Pin D should show negative polarity; pin G should show positive polarity.

d. Rotate the right inboard skid detector drive in the direction of normal wheel rotation, and check the polarity of pins D and G of the inboard control box connector.
RESULT: Pin D should show negative polarity; pin G should show positive polarity.

e. Rotate the left outboard skid detector drive in the direction of normal wheel rotation, and check the polarity of pins G and H of the outboard control box connector.
RESULT: Pin G should show positive polarity; pin H should show negative polarity.

f. Rotate the left inboard skid detector drive in the direction of normal wheel rotation, and check the polarity of pins G and H of the inboard control box connector.
RESULT: Pin G should show positive polarity; pin H should show negative polarity.

g. Replace the drive assemblies; reconnect the connectors to the control boxes, and return the anti-skid system to normal.

3.2.3.2.4 QUICK CHECK OF ANTI-SKID SYSTEM (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096).

a. With external electrical power connected to the airplane, engage the anti-skid circuit breaker, and place the ANTI-SKID switch in the "ON" position.

b. Remove the hub caps from the main gear wheels.

c. Flip the drive of each skid detector with the finger in the direction of forward rotation.
RESULT: The anti-skid control valve solenoid should actuate, causing an audible click.

d. Replace hub caps.

32.3.2.5 ANTI-SKID SYSTEM (AIRPLANES 5093, 5097, AND UP). The anti-skid control box contains circuits for checking out the anti-skid system. The test switch and four indicator lights are on the Anti-Skid Panel located on the copilot's wainscot.

a. Connect external electrical power to the airplane.

b. Be sure wheel chocks are in place and parking brakes are released.

c. Connect external hydraulic power to the normal hydraulic system, and adjust the power source to deliver zero gallons per minute at 3,000 psi and 10 gallons per minute at 2,700 psi.

d. Install a pressure gage and adapter on each brake assembly bleed port.

e. Engage the following circuit breakers: anti-skid, anti-skid test, brakes, warning lights, landing gear control, No. 1 main inverter control, and inverter out light.
RESULT: With these circuit breakers engaged and power on the airplane, the ANTI-SKID OUT light should be illuminated with the ANTI-SKID switch in the OFF position.

f. Make sure the HYD SYS NORM-1 STANDBY-2 switch is in the NORM-1 position. Place the ANTI-SKID switch on the copilot's control column in the ON position and the NO. 1 MAIN INVERTER switch on the overhead panel in the ON position.
RESULT: The ANTI-SKID OUT light should extinguish.

g. Disengage the anti-skid circuit breaker, and observe the ANTI-SKID OUT warning light.
RESULT: The ANTI-SKID OUT light should illuminate.

h. Engage the anti-skid circuit breaker, and observe the ANTI-SKID OUT warning light.
RESULT: The ANTI-SKID OUT light should extinguish.

i. Apply the brakes and observe the brake pressure gages in the bleed ports of the brake assemblies.
RESULT: All the brake pressure gages should indicate brake pressure.

j. With the brakes still applied, place the ANTI-SKID TEST switch in the INBD TEST position.
RESULT: The RIGHT and LEFT INBD indicator lights and the ANTI-SKID OUT light should illuminate and the pressure gages on the inboard wheels should indicate a momentary decrease of brake pressure. The brake pressure on the outboard wheels should not be affected.

k. Release the ANTI-SKID TEST switch.
RESULT: The test switch should return to the NORM position and the right and left inboard indicator lights and the ANTI-SKID OUT light should extinguish.

l. With the brakes still applied, place the ANTI-SKID TEST switch in the OUTBD TEST position.
RESULT: The RIGHT and LEFT OUTBD indicator lights and the ANTI-SKID OUT light should illuminate and the pressure gages on the outboard wheels should indicate a momentary decrease of brake pressure. The brake pressure on the inboard wheels should not be affected.

m. Release the ANTI-SKID TEST switch.
RESULT: Same as step k, except right and left outboard indicator lights.

n. Release the brakes and place the HYD SYS NORM-1 STANDBY-2 switch in the STANDBY-2 position.
RESULT: The ANTI-SKID OUT light and all four of the wheel indicator lights should illuminate.

o. Place the HYD SYS NORM-1 STANDBY-2 switch in the NORM-1 position.
RESULT: The ANTI-SKID OUT light and the four wheel indicator lights should remain illuminated.

p. Reset the system by replacing the ANTI-SKID switch in the OFF position, then back to the ON position.
RESULT: The ANTI-SKID OUT light and the four wheel indicator lights should extinguish.

q. Rotate the left outboard wheel transducer at a steady speed (200 to 1000 RPM) in either direction. (Use a variable speed drill motor with a suitable adapter for rotating the transducer.)
RESULT: Pressure shall be removed from the right outboard wheel brake.

Note

During this test the system detects a locked wheel condition on the wheel opposite to the wheel being turned and releases pressure on the locked wheel.

r. Slowly reduce left outboard wheel transducer rotating speed.
RESULT: Pressure shall be restored to right outboard wheel brake as transducer speed is reduced.

s. Rotate the left outboard wheel transducer speed at 200 to 1000 RPM, and then stop abruptly.
RESULT: Brake pressure shall be instantly removed from the left outboard wheel when the transducer is stopped but pressure will be restored within 1.5 seconds.

Note

During this test the system detects a skid condition on the wheel being turned and releases pressure to prevent skid.

t. Repeat steps q, r, and s, left inboard wheel, right inboard and right outboard.

u. Reduce hydraulic pressure to zero, and then cycle the ailerons or elevators to reduce pressure in the normal hydraulic system to zero. Operate the wheel brakes until the pressure in the brake system is zero.

v. With no pressure in the brake system, remove the adapters and pressure gages from the bleed ports of the brake assemblies, and return the brake and anti-skid systems to normal.

Note

If the operational checkout of the anti-skid system is made with the airplane on jacks, the results of steps e through p will be the same with the exception of the brake pressure in steps i, j, and l. In step i, all gages should indicate no brake pressure. In step j, the gages on the inboard wheels should indicate a momentary increase of brake pressure. In step l, the gages on the outboard wheels should indicate a momentary increase of brake pressure.

32.3.3 TROUBLESHOOTING.

32.3.3.1 WHEEL BRAKE SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
SIMULTANEOUS HYDRAULIC SYSTEM PRESSURE FLUCTUATION DETECTED ON SYSTEMS ONE AND TWO INDICATOR POINTERS, DURING THE BRAKE AND ANTI-SKID SYSTEM TEST WITH NUMBER ONE AND TWO PRESSURE NORMAL.		
Number two, brake selector valve.	Confirm that the valve solenoid is de-energized and the valve has not failed open.	Repair or replace valve.
EXCESSIVE HYDRAULIC PRESSURE ON BRAKE ASSEMBLIES.		
Defective or improperly adjusted dual-brake control valves.	Check the operation and the adjustment of the valves.	Replace or adjust the valves as needed.
INSUFFICIENT HYDRAULIC PRESSURE ON BRAKE ASSEMBLIES.		
Insufficient hydraulic power available.	Check the hydraulic power source.	Obtain proper hydraulic power.
Improperly adjusted dual-brake control valves.	Check the operation and the adjustment of the valves.	Replace or adjust the valves as needed.
HYDRAULIC PRESSURE ON BRAKE ASSEMBLIES FAILS TO DROP TO 10 PSI OR LESS WHEN CORRESPONDING PEDAL IS RELEASED.		
Parking brakes set.	Check that the PARK BRAKE tee handle is fully in.	Release tee handle.
Improperly adjusted dual-brake control valve.	Check the operation and adjustment of the valve.	Replace or adjust the valve as needed.
Dirty system return line filter.	Check button on filter case.	Clean filter element if necessary.
BRAKE PRESSURE LOW WARNING LIGHT REMAINS ON WITH OVER 1,400 PSI PRESSURE IN THE HYDRAULIC SYSTEM POWERING THE BRAKES.		
Defective low pressure sensing switch.	Check the operation of the switch.	Replace the switch as needed.
Short in warning light circuit.	Check continuity of circuit.	Repair circuit as needed.
HYD SYS NORM-1 STANDBY-2 switch not wired properly.	Check the wiring of the switch.	Correct wiring as needed.
SLOW AND INEFFECTIVE BRAKING.		
Adjusting screws not properly torqued.	Check screws for correct torque.	Install screws with correct torque.

32.3.3.2 ANTI-SKID SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ANTI-SKID OUT LIGHT FAILS TO GO ON WHEN THE ANTI-SKID SWITCH IS PLACED IN THE "OFF" POSITION.		
Defective bulb.	Check bulb.	Replace defective bulb.
Open warning circuit.	Check continuity of circuit.	Repair circuit as needed.
ANTI-SKID OUT LIGHT GOES ON APPROXIMATELY THREE SECONDS AFTER THE ANTI-SKID SWITCH IS PLACED IN THE "ON" POSITION (AIRPLANES 5001 THRU 5092 AND 5094 THRU 5096 ONLY).		
False skid signal to the control boxes.	Check the continuity of the airplane wiring, and also check the operation of the control boxes.	Repair circuit or replace components as needed.

32.3.3.2 ANTI-SKID SYSTEM (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ANTI-SKID OUT LIGHT GOES ON APPROXIMATELY THREE SECONDS AFTER THE ANTI-SKID SWITCH IS PLACED IN THE "ON" POSITION (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096 ONLY) (CONT).		
Defective anti-skid control valve.	Check operation of anti-skid control valves.	Repair or replace components as needed.
ANTI-SKID OUT WARNING LIGHT STAYS ON WHEN THE ANTI-SKID SWITCH IS PLACED IN THE "ON" POSITION (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096 ONLY).		
Defective circuit, control box, or anti-skid valve.	Check the continuity of the circuits and the valve solenoid.	Replace components or repair circuits as needed.
LIGHT COMES ON IMMEDIATELY AFTER BECOMING AIRBORNE (AIRPLANES 5001 THROUGH 5092 AND 5094 THROUGH 5096 ONLY).		
Brakes applied or tire rubbing in wheel well. Binding or defective bearing or hub cap spinner.	Check wheel well for evidence of tire rubbing. Check bearings and hub cap spinners.	Repair or replace components as needed.
ANTI-SKID OUT LIGHT AND ONE OR MORE WHEEL LIGHTS COME ON AT ANY TIME AFTER ANTI-SKID SWITCH IS PLACED IN THE "ON" POSITION (AIRPLANES 5093, 5097, AND UP ONLY).		
Loss of system power or defective wiring.	Check operation of circuit breaker and continuity of wiring.	Repair or replace components as needed.
Open or shorted anti-skid valve.	Check operation of valve.	Replace valve.
Defective circuit in control box.	Check operation of control box.	Replace control box.
Loss of touchdown protection. (Airborne only)	Check operation and continuity of squat switches. Check continuity of wiring. Check operation of control box.	Replace components or repair circuits as needed.
Open or shorted transducer.	Check operation and continuity of transducer and wiring.	Replace transducer or repair circuit as needed.

32.3.4 REMOVAL AND REPLACEMENT.

Note

See the following B.F. Goodrich Maintenance and Overhaul Manuals for detailed instructions on maintenance and overhaul of the brakes:

a. Maintenance and Overhaul Manual 667 for -8 JetStars.

b. Maintenance and Overhaul Manual for Part No. 2-798-4 and 2-798-5 for -6 JetStars.

32.3.4.1 REMOVAL OF THE WHEEL BRAKE ASSEMBLY.

a. Reduce brake pressure to zero by operating the brakes and flight controls.

b. Remove the wheel assembly according to the instructions in paragraph 32.0.4.13.

c. Disconnect the hydraulic line to the brake assembly.

d. Plug the disconnected hydraulic line and the brake assembly inlet port.

e. Remove the six brake attachment bolts.

f. Remove the brake assembly from the axle.

32.3.4.2 REPLACEMENT OF THE WHEEL BRAKE ASSEMBLY.

a. Place the brake assembly on the axle and install the six brake assembly bolts.

Note

The nuts shall be Standard Pressed Steel number 68457-624, Kaynar H29649-6, or Esna L55LHEB220-064 torqued to 400-450 inch-pounds.

b. Remove the inlet plug from the brake assembly, uncap the brake pressure line, and connect the brake line to the brake assembly inlet port.

c. Bleed the brake system after securing the hydraulic connection.

d. Replace the tire and wheel assembly.

e. After the bearing spacer and the wheel are properly installed, and while the airplane is still on the jacks, rotate the wheel to check for brake drag.

32.3.4.3 REMOVAL OF DUAL-BRAKE CONTROL VALVES.

WARNING

Chock the wheels, depressurize the normal and standby hydraulic systems, de-energize the auxiliary hydraulic pump circuits, and warn personnel not to move the brake or rudder pedals.

a. Remove the access panels inside the nose wheel well to gain access to the valve.

Note

The valve for the right wheel brakes is on the left side of the airplane; the valve for the left wheel brakes is on the right side of the airplane.

b. Disconnect the six hydraulic lines from the valve. Plug or cap the openings.

c. Remove the two nuts, washers, and bolts holding the valve assembly to the valve support, and remove the valve.

32.3.4.4 REPLACEMENT OF DUAL-BRAKE CONTROL VALVES.

a. Reverse the removal procedures.

b. Bleed the brake system according to the instructions in Chapter 29.

c. Check brake system operation, and adjust the brake system as needed.

32.3.5 ADJUSTMENT.

32.3.5.1 DUAL-BRAKE CONTROL VALVE ADJUSTMENT. For these procedures, refer to figure 32-22.

32.3.5.2 INSPECTION.

32.3.5.3. INSPECTING BRAKE LINING WEAR. Check the wheel brakes for lining wear as follows:

a. Dash 6 aircraft - (See figure 32-21A).

At each routine ground check, visually inspect for excessive corrosion, broken or cracked parts, loose bolts, screws and fittings. Check all lock-wired parts, and replace wire if broken or loose. Tighten all loose threaded parts to proper torque. Check the position of the pins (24, 25) in the adjuster as an indication of lining wear. If pins are flush with or below the surface of the adjuster housing (27) when brake pressure is applied, worn lining is indicated and brake should be removed for further lining wear check. Check for leaks from inlet and bleeder ports with brake pressure applied.

b. Dash 8 aircraft - (See figure 32-21.)

At each routine ground check, visually inspect for excessive corrosion, broken, or cracked parts, loose bolts, screws and fittings. Check all lock-wired parts, and replace wire if broken or loose. Tighten all loosely threaded parts to proper torque. Check the position of the pins (30) in the adjusters next to the fittings (17, 19) as an indication of lining wear. If pins are even with or below top surface of screw (24), worn lining is indicated and brake should be removed for further lining wear check. Check for leaks from inlet and bleeder ports with brake pressure applied.

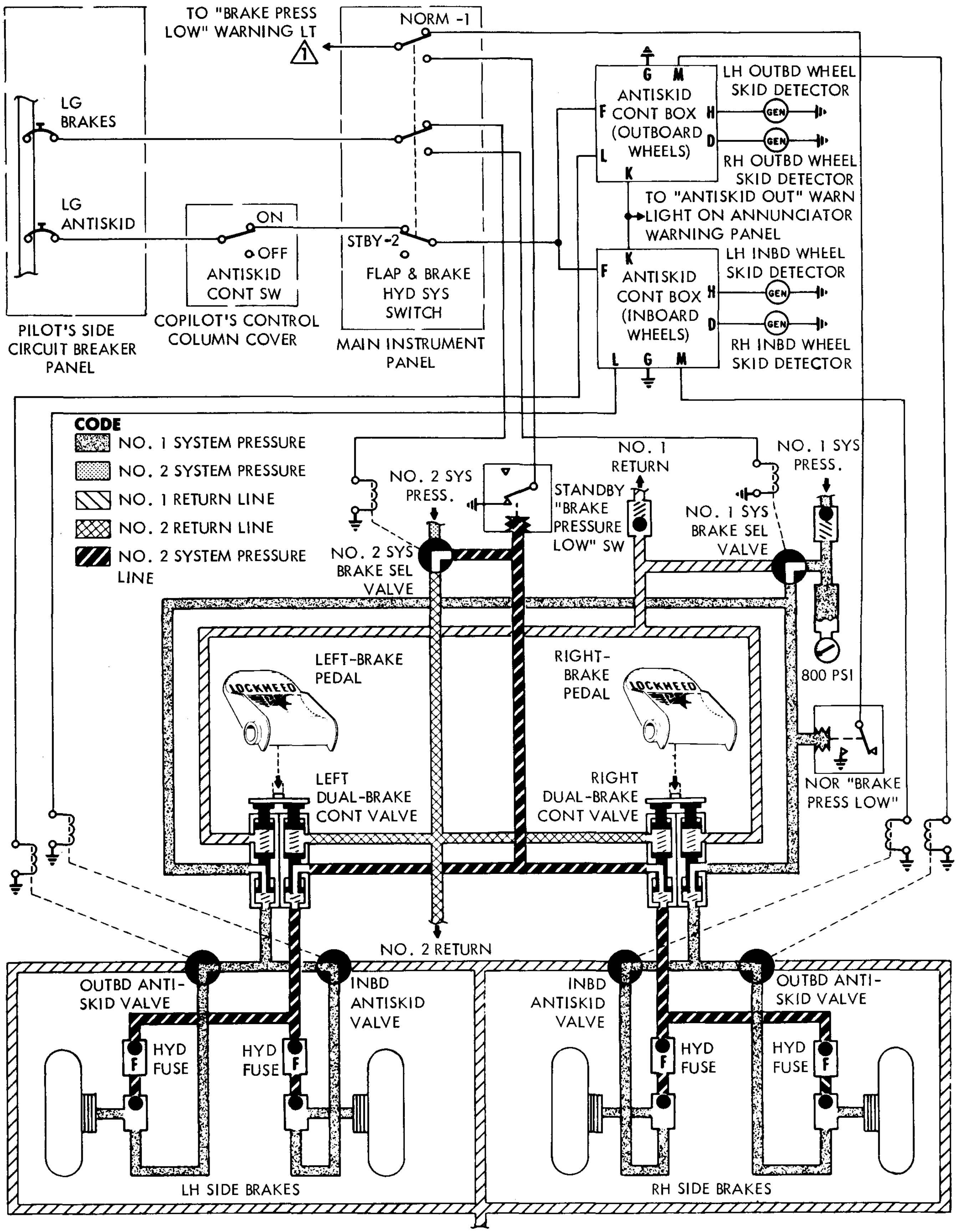

TO "BRAKE PRESS LOW" WARNING LT
1
NORM -1
LG BRAKES
LG ANTISKID
ON
OFF
ANTISKID CONT SW
STBY-2
FLAP & BRAKE HYD SYS SWITCH
PILOT'S SIDE CIRCUIT BREAKER PANEL
COPILOT'S CONTROL COLUMN COVER
MAIN INSTRUMENT PANEL
G M
ANTISKID CONT BOX (OUTBOARD WHEELS)
F H D L K
LH OUTBD WHEEL SKID DETECTOR
RH OUTBD WHEEL SKID DETECTOR
GEN
TO "ANTISKID OUT" WARN LIGHT ON ANNUNCIATOR WARNING PANEL
K
F
ANTISKID CONT BOX (INBOARD WHEELS)
H D
L G M
LH INBD WHEEL SKID DETECTOR
RH INBD WHEEL SKID DETECTOR
CODE
NO. 1 SYSTEM PRESSURE
NO. 2 SYSTEM PRESSURE
NO. 1 RETURN LINE
NO. 2 RETURN LINE
NO. 2 SYSTEM PRESSURE LINE
NO. 2 SYS PRESS.
NO. 2 SYS BRAKE SEL VALVE
STANDBY "BRAKE PRESSURE LOW" SW
NO. 1 RETURN
NO. 1 SYS PRESS.
NO. 1 SYS BRAKE SEL VALVE
800 PSI
LEFT-BRAKE PEDAL
RIGHT-BRAKE PEDAL
LOCKHEED
LEFT DUAL-BRAKE CONT VALVE
RIGHT DUAL-BRAKE CONT VALVE
NOR "BRAKE PRESS LOW"
NO. 2 RETURN
OUTBD ANTI-SKID VALVE
INBD ANTISKID VALVE
INBD ANTISKID VALVE
OUTBD ANTI-SKID VALVE
HYD FUSE
F
HYD FUSE
HYD FUSE
HYD FUSE
LH SIDE BRAKES
RH SIDE BRAKES

Figure 32-20. Wheel Brake and Anti-Skid System Schematic Diagram (Airplanes 5001 thru 5092 and 5094 thru 5096)

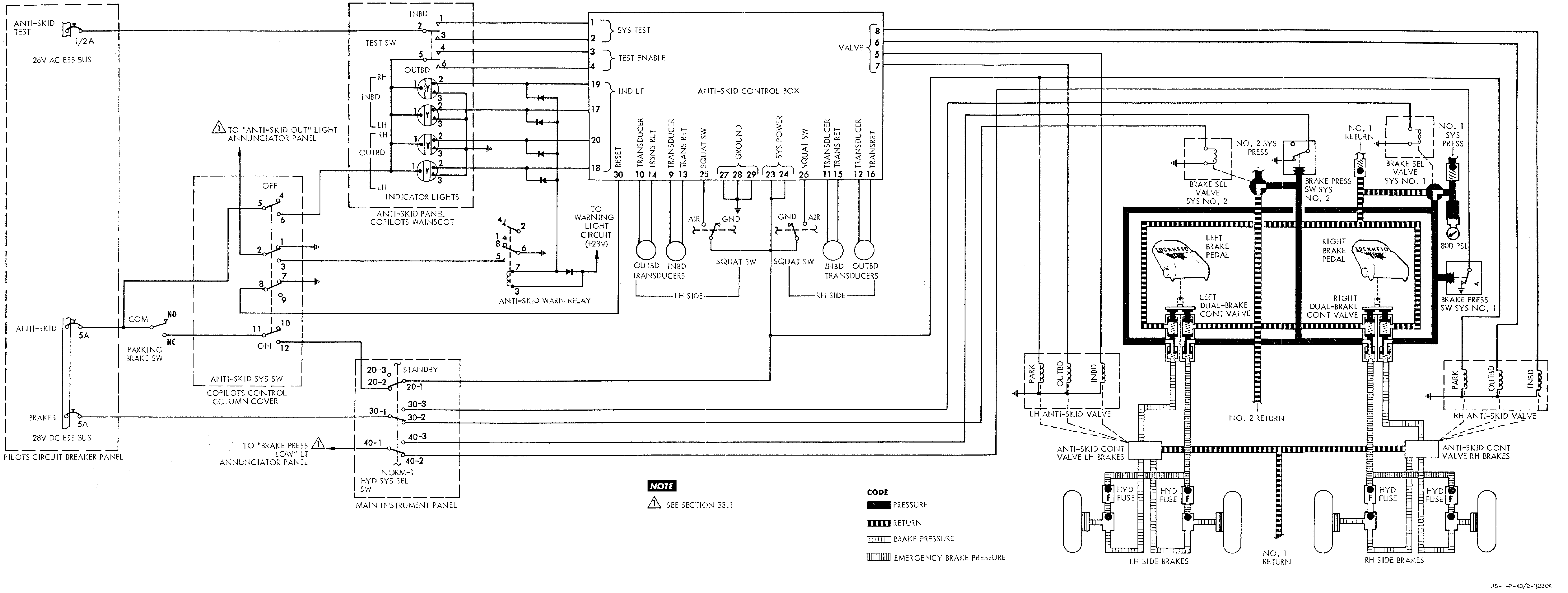
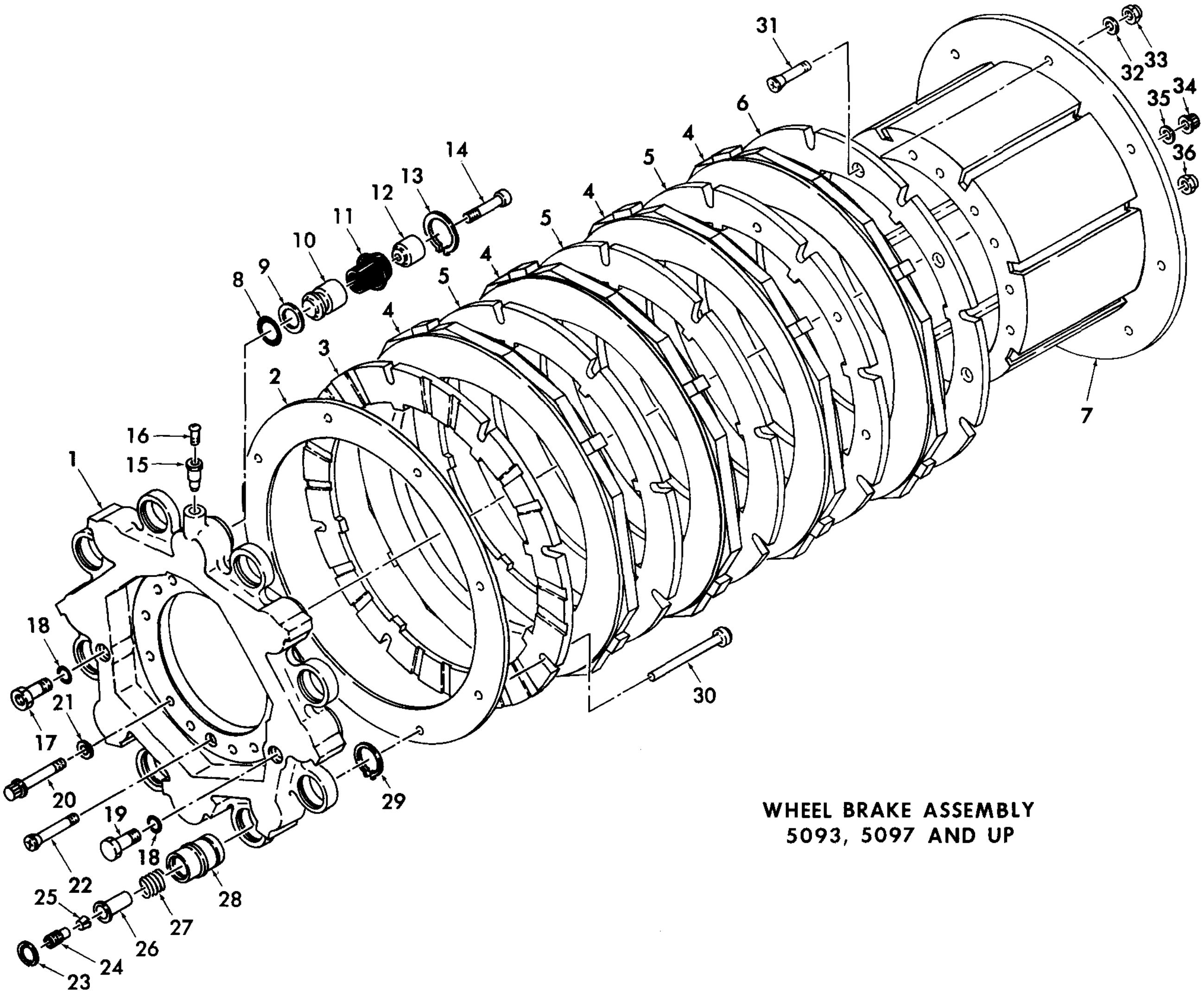


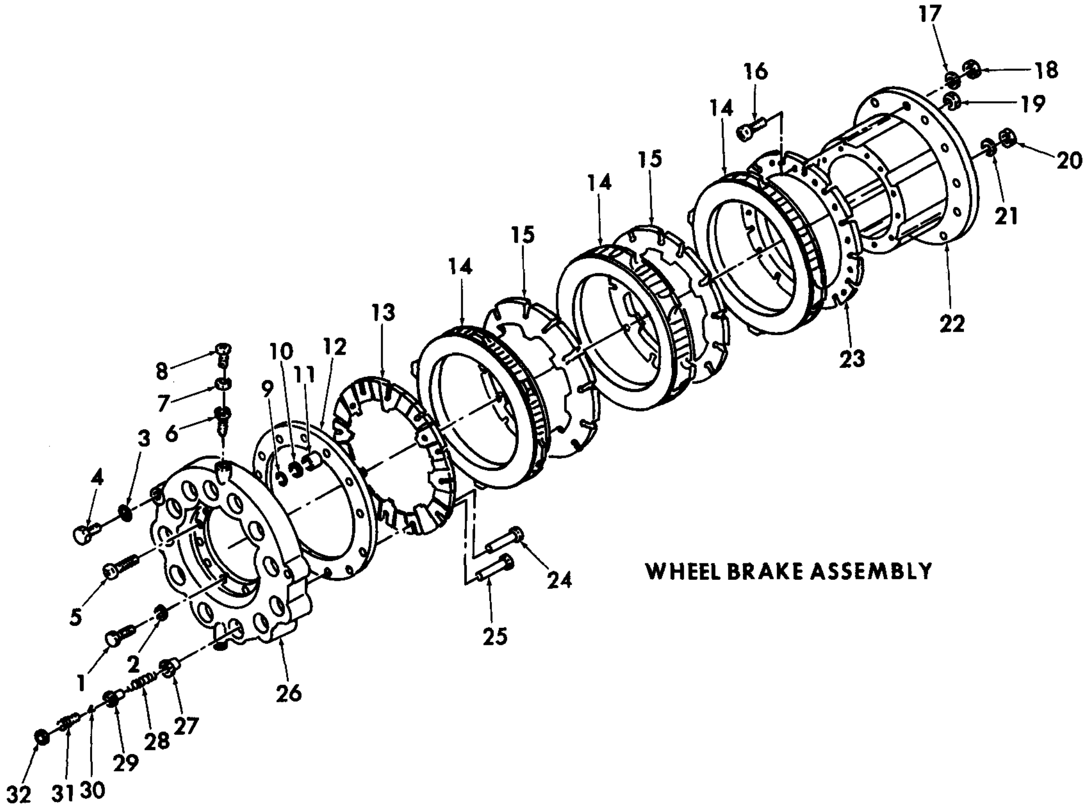
Figure 32-20A. Wheel Brake and Anti-Skid System Schematic Diagram (Airplanes 5093, 5097 and Up)



WHEEL BRAKE ASSEMBLY
5093, 5097 AND UP

WHEEL BRAKE COMPONENTS LIST	
1. PISTON HOUSING	19. MACHINE THREAD PLUG
2. HEAT SHIELD	20. MACHINE BOLT
3. PRESSURE PLATE	21. FLAT WASHER
4. CARRIER AND LINING ASSEMBLY	22. MACHINE SCREW
5. BRAKE DISK	23. RETAINING RING
6. WEAR PLATE	24. MACHINE SCREW
7. TORQUE PLATE	25. BRASS COLLET
8. PREFORMED PACKING	26. ADJUSTER BODY
9. PACKING BACKUP RING	27. COMPRESSION SPRING
10. ALUMINUM ALLOY PISTON	28. ADJUSTER HOUSING
11. DUST BOOT	29. RETAINING RING
12. PISTON INSULATOR	30. STRAIGHT PIN
13. RING	31. MACHINE SCREW
14. CAP SCREW	32. FLAT WASHER
15. BLEEDER VALVE	33. SELF-LOCKING NUT
16. MACHINE SCREW	34. SELF-LOCKING NUT
17. FITTING	35. FLAT WASHER
18. PREFORMED PACKING	36. SELF-LOCKING NUT

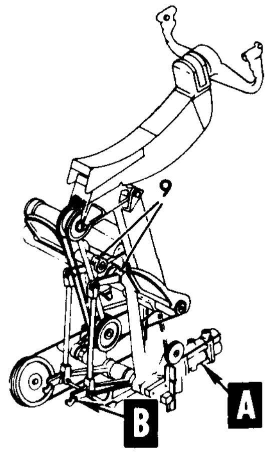
Figure 32-21. Wheel Brake Assembly



WHEEL BRAKE COMPONENTS LIST

1. MACHINE BOLT
2. FLAT WASHER
3. PREFORMED PACKING
4. MACHINE THREAD PLUG
5. MACHINE SCREW
6. BLEEDER VALVE
7. LOCK WASHER
8. MACHINE SCREW
9. PREFORMED PACKING
10. PACKING BACKUP RING
11. ALUMINUM ALLOY PISTON
12. THERMAL INSULATION BOARD
13. STEEL PLATE
14. CARRIER AND LINING ASSEMBLY
15. BRAKE DISK
16. MACHINE SCREW
17. FLAT WASHER
18. SELF-LOCKING NUT
19. SELF-LOCKING NUT
20. SELF-LOCKING NUT
21. FLAT WASHER
22. TORQUE WASHER
23. WEAR PLATE
24. STRAIGHT PIN
25. STRAIGHT PIN
26. PISTON HOUSING
27. ADJUSTER HOUSING
28. COMPRESSION SPRING
29. ADJUSTER BODY
30. BRASS COLLET
31. MACHINE SCREW
32. RETAINING RING

Figure 32-21A. Wheel Brake Assembly (5001 thru 5092, 5094 thru 5096)

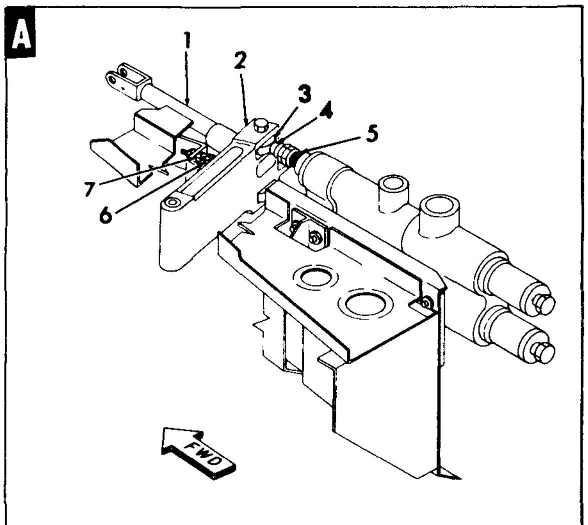


STEP 1

REMOVE THE ACCESS PANEL FORWARD OF THE NOSE WHEEL WELL; THEN REMOVE THE FOUR ACCESS PANELS INSIDE.

NOTE

THE FOLLOWING PROCEDURES ARE FOR THE RIGHT WHEEL BRAKE CONTROL VALVE LOCATED ON THE LEFT SIDE OF THE AIRPLANE. ADJUSTMENT PROCEDURES FOR THE LEFT WHEEL BRAKE VALVE ARE IDENTICAL EXCEPT FOR THE PARKING BRAKE ADJUSTMENT.



VALVE FOR RIGHT WHEEL BRAKES SHOWN.
LEFT WHEEL BRAKE VALVE IS A MIRROR OPPOSITE.

STEP 2

LOOSEN THE NUT (7) TO UNLOCK THE ADJUSTABLE STOP (6).

STEP 3

ADJUST THE FORWARD BOLTS (4) ON THE PLUNGERS (5) TO CONTACT THE BEARING (3) SIMULTANEOUSLY AND ALSO TO GIVE 0.55 (±0.03) INCH VALVE PLUNGER STROKE.

STEP 4

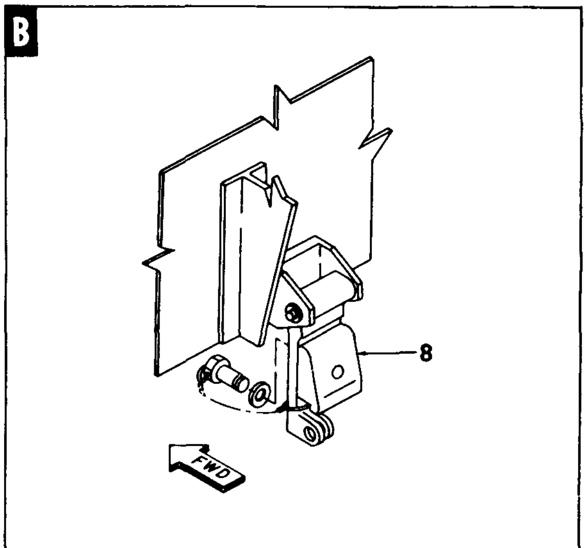
WITH THE BRAKES OFF, ADJUST THE STOP (6) TO GIVE 0.005 TO 0.015 INCH CLEARANCE BETWEEN THE STOP (6) AND THE LEVER (2), THEN LOCK THE STOP IN PLACE BY TIGHTENING THE NUT (7).

STEP 5

ADJUST THE ROD (1) TO ALIGN THE WITNESS MARKS (9) ON THE PEDAL LINKS (IN THE PILOT'S COMPARTMENT) WITHIN 0.12 INCH.

STEP 6

ADJUST THE PARKING PAWLS (8) TO GIVE 420 TO 520 PSI (5001-92, 94-96) 725-975 PSI (5093, 97 AND UP) BRAKE PRESSURE WITH THE PARKING BRAKES ON AND THE HYDRAULIC SYSTEM PRESSURIZED TO NORMAL PRESSURE.



ONE PAWL SHOWN,
OTHER PAWL IDENTICAL.

JS-I-2-X0/8-3222

Figure 32-22. Dual-Brake Control Valve Adjustment

SECTION 32.4
NOSE STEERING SYSTEM

Contents Page

DESCRIPTION AND OPERATION 32-87
  General 32-87
  Steering Wheel 32-88
  Steering Selector Valve (Airplanes 5001 through 5045) 32-88
  Hydraulic Fuse (Airplanes 5046 and up, and airplanes having Service Bulletin 329-214 incorporated) 32-88
  Steering Selector Valves (Airplanes 5046 and up, and airplanes having Service Bulletin 329-214 incorporated) 32-88
  Dual Shuttle Valve (Airplanes 5046 and up, and airplanes having Service Bulletin 329-214 incorporated) 32-88
  Steering Control Valve 32-88
  Steering Actuating Cylinder 32-88
  Nose Steering Torque Arms 32-88
  Nose Gear Scissors Switch 32-88
  Nose Steering Index Marks 32-88A
  Check Valves 32-89

OPERATIONAL CHECKOUT 32-89
  Nose Gear Steering System (Airplanes 5001 through 5045) 32-89
  Nose Gear Steering System (Airplanes 5046 and up) 32-89
  Nose Steering System Pulling Checkout 32-90

TROUBLESHOOTING 32-90
  Nose Gear Steering System 32-90

REMOVAL AND REPLACEMENT 32-91
  Removal of the Nose Steering Control Valve 32-91
  Replacement of the Nose Steering Control Valve 32-91

ADJUSTMENT 32-91
  Nose Steering System Rigging Instructions 32-91
  Nose Steering Control Valve Adjustment 32-91

FIGURES
  32-23. Nose Steering System Schematic Diagram (Airplanes 5001 through 5045) 32-92
  32-24. Nose Steering System Schematic Diagram (Airplanes 5046 and up) 32-93
  32-25. Nose Steering System Rigging Instructions 32-94
  32-26. Adjustment of Nose Landing Gear Steering Control Valve No. JL1351-7 32-96
  32-26A. Repacking Nose Steering Cylinder Assembly 32-96A

32.4.1 DESCRIPTION AND OPERATION.

32.4.1.1 GENERAL. (See figures 32-23 and 32-24.) The nose wheels of the JetStar airplanes are hydraulically actuated to steer the airplane on the ground. The steering system can be powered by the normal or auxiliary hydraulic systems only (on airplanes 5001 thru 5045). On airplanes 5046 and up, and airplanes incorporating Service Bulletin 329-214, the standby system can also be used. Hydraulic components in the steering system include a steering selector valve, a steering control valve, and a steering cylinder. For airplanes 5046 and up, a second selector valve, a hydraulic fuse, and a dual shuttle valve are also included. When the airplane is on the ground, the steering selector valve is energized by a nose gear scissors switch to admit hydraulic power to the steering control valve. For airplanes 5046 and up electric power from the nose gear scissors switch goes to the HYD SYS NORM-1 STANDBY-2 switch where it is directed to either the normal steering selector valve or the standby selector valve. Hydraulic power from the energized selector valve is directed to the steering control valve through the dual shuttle valve. The steering control valve is positioned by mechanical linkage to the steering wheel to admit hydraulic power to the proper sides of the actuating cylinder to turn the steering collar. The steering collar acts through torque arms, or scissors, to turn the nose wheels according to steering input commands. Maximum nose wheel travel is 53 degrees in either direction. Airplanes 5001 thru 5040 must have the landing gear selector valve energized in the gear-down position in order to admit hydraulic power to the nose steering and main wheel brake systems. On airplanes

32.4.1.1 (Cont).

5041 and up, and on all previous airplanes equipped with a valve manufactured by the Parker Aircraft Company an internal spring detent holds the valve in the gear-down position after the No. 2 solenoid is de-energized. Airplanes 5041 and up have a manually controllable landing gear isolation valve which can be positioned to admit hydraulic power to the nose steering system and the main wheel brake system. This valve is controlled by an LG ISOLATION VALVE tee handle attached to the left wheel well inside the airplane. (See Section 32.1 for use of the isolation valve.)

32.4.1.2 STEERING WHEEL. The steering wheel assembly is attached to a shaft mounted on the pilot's side inclined panel. The shaft has a sprocket and chain mechanism that positions control cables to the steering control valve when the wheel is rotated. A mechanism in the aft. left area of the nose wheel well allows control valve positioning when the gear is down. and disengages the control valve from steering input commands when the gear is up. This mechanism also locks the steering input system in the neutral position while the gear is up.

32.4.1.3 STEERING SELECTOR VALVE (AIRPLANES 5001 THRU 5045). A steering selector valve in the aft area of the nose wheel well energizes when the nose gear is on the ground to admit hydraulic power to the nose steering control valve. This valve is a three-way. two-position. single-solenoid valve and is controlled by the nose gear scissors switch. When the nose gear is off the ground. the steering selector valve is de-energized to block hydraulic power from the steering control valve.

32.4.1.3.1 HYDRAULIC FUSE (AIRPLANES 5046 AND UP, AND AIRPLANES HAVING SERVICE BULLETIN 329-214 INCORPORATED). A hydraulic fuse is located between the standby steering selector valve and the standby system pressure main line. The fuse is mounted under the cabin floor at fuselage sta. 251 and RBL 10. The normal flow is 120 cubic inches of fluid before shutoff occurs. The fuse prevents loss of all stand-by system fluid in the event of uncontrolled leakage in the stand-by steering system. (See paragraph 32.3.1.15 for further description of fuse operation.)

32.4.1.4 STEERING SELECTOR VALVES (AIRPLANES 5046 AND UP, AND AIRPLANES HAVING SERVICE BULLETIN 329-214 INCORPORATED). The normal steering selector valve is located on the aft left side of the nose wheel well and the standby steering selector valve is located on the aft right side of the nose wheel well. These valves are three-way. two-position single-solenoid pilot operated and are controlled by the nose gear scissors switch and the HYD SYS NORM-1 STANDBY-2 switch. When the nose gear is off the ground both selector valves are de-energized to block hydraulic power from the steering control valve. When the nose gear is on the ground the valve selected by the HYD SYS NORM-1 STANDBY-2 switch energizes to admit either normal or standby hydraulic power to the nose steering control valve.

32.4.1.5 DUAL SHUTTLE VALVE (AIRPLANES 5046 AND UP, AND AIRPLANES HAVING SERVICE BULLETIN 329-214 INCORPORATED). A dual shuttle valve is located on the aft right side of the nose wheel well, and connects the normal and standby steering selector valves to the steering control valve. The dual shuttle valve is pressure operated and functions so as to direct the hydraulic pressure from the energized selector valve to the control valve. and also directs the return flow from the control valve to the system supplying pressure.

32.4.1.6 STEERING CONTROL VALVE. A steering control valve assembly near the top of the nose gear trunnion routes hydraulic power to the actuating cylinder according to input signals from the steering wheel. Positioning the steering wheel positions a spool in the control valve to route hydraulic flow to the proper sides of the actuating cylinder to turn the nose wheels. When no input signals are present. the spool is centered by means of springs. This centering action holds the nose wheels in the centered position and also maintains pressure on both sides of the steering cylinder to act as a shimmy damper during non-steering operations. A spring accumulator in the steering control valve assembly minimizes surges in the steering system and also helps to maintain pressure on both sides of the actuating cylinder during non-steering operation.

32.4.1.7 STEERING ACTUATING CYLINDER. A steering actuating cylinder positions the steering collar on the strut. which in turn positions the scissors to turn the nose wheels. The steering cylinder is a double-acting cylinder attached to the rear of the nose strut assembly. Due to cylinder rotation during steering operations. a steering flapper door in the strut door is spring-loaded closed. but opens when the steering cylinder contacts the rubbing strip on the door. The steering control valve controls actuation of the actuation cylinder.

32.4.1.8 NOSE STEERING TORQUE ARMS. Torque arms. or scissors. transmit the motion of the steering collar to the nose strut piston to turn the nose wheels. The scissors fold and unfold as the strut piston extends and retracts. This action rotates a cam on the lower half of the scissors. The cam positions the nose scissors switch to control the steering selector valve. The upper and lower halves of the scissors can be disconnected during towing operation. To do this. the pin at the scissors joint must be removed and reinserted through the upper half of the scissors and into the strut to hold the upper half of the scissors in the horizontal position.

32.4.1.9 NOSE GEAR SCISSORS SWITCH. The nose gear scissors switch is a microswitch attached to the nose strut near the scissors. This switch is positioned by a cam driven by the scissors. The

32.4.1.9 (Cont).

scissors switch makes and breaks circuits to the steering selector valves according to strut extension. As the airplane weight is removed from the nose wheels. the strut extends and the switch breaks the circuit to the steering selector valves. As the airplane weight is placed on the nose wheels. the strut is compressed and the switch makes a circuit to energize the selector valves and admit hydraulic power to the steering control valve.

32.4.1.10 NOSE STEERING INDEX MARKS. One set of index marks on the nose strut assembly. and one set of index marks on the steering wheel (See figure 32-25.) show the positions of steering components without need of special tools. An index mark on the top spoke of the steering wheel aligns with a painted mark on the pilot's side inclined panel when the steering input system is in the neutral position. On the strut assembly, there are five index marks on the strut and one on the steering collar adjacent to the strut marks. When the one on the collar is aligned with either outside mark on the strut, the nose wheels are turned 53 degrees. When the steering collar

32.4.1.10 (Cont).

mark is aligned with the center index mark on the strut, the nose wheels are centered. The two inside marks near the center mark on the strut are four degrees from the center mark.

32.4.1.11 CHECK VALVES. Airplanes 5046 and up, and airplanes having Service Bulletin 329-214 incorporated, have two check valves located in the hydraulic return systems mounted downstream of the normal (No. 1) system and standby (No. 2) system steering selector valves. The primary purpose of these valves is to prevent repositioning of the steering system shuttle valves by transient surges in the airplane system return hydraulic lines.

32.4.2 OPERATIONAL CHECKOUT.

32.4.2.1 NOSE GEAR STEERING SYSTEM (AIRPLANES 5001 THRU 5045).

a. Make sure that the normal hydraulic system accumulator is properly serviced according to the instructions in Chapter 12.

b. Make sure that all of the landing gear ground safety locks are installed on the gear.

c. Jack the nose of the airplane, and then place two steel plates with grease between them under the nose wheels. Lower the airplane so that the nose wheels rest upon the steel plates to prevent scuffing of the tires during this test.

d. Make sure that the landing gear control handle is in the "LG DOWN" position.

e. Connect 28-volt, DC power to the airplane and make sure that the nose steering and landing gear control circuit breakers are engaged.

f. Connect external hydraulic power to the normal hydraulic system ground test connections, and adjust the power source to deliver no flow at 3,000 PSI and 3 gallons per minute at 2,700 PSI.

g. Turn the steering wheel from full left to full right six complete cycles to make sure that the hydraulic system is free of air.

h. Center the steering wheel by aligning the index mark on the top spoke of the wheel with the mark on the pilot's side inclined panel. Check the position of the nose wheels with the steering wheel in this position.
RESULT: The nose wheels should be within 2 degrees of the centered position.

Note

The two index marks on the strut which are near the center index mark are 4 degrees from the center marks.

i. Turn the steering wheel from full left to full right while observing steering system operation.
RESULT: The steering cylinder should bottom in each direction, and the index marks on the strut should indicate full travel (approximately 53 degrees) in each direction. The rubbing block on the steering flapper door should be serviceable, and should be positioned so that the cylinder touches the rubbing block instead of the door itself.

j. Remove the plug from the center of the steering wheel, and place a socket and torque wrench on the retaining nut of the steering wheel. Using the torque wrench, turn the nose wheels full left and then full right, and observe the amount of torque required to obtain this travel.
RESULT: The torque required to obtain full nose wheel travel should not exceed 45 inch-pounds.

k. With the nose wheels either full left or full right, jack the airplane until the nose wheels clear the ground. Observe the position of the nose wheels after they clear the ground.
RESULT: The nose wheels should center within 2 degrees as they leave the ground, and the nose steering wheel should center within 1/2-inch as measured at the index marks on the steering wheel and the inclined panel.

l. Move the steering wheel from full right to full left, and observe the action of the nose wheels.
RESULT: The nose wheels should not move.

m. Remove the greased plates from under the nose wheels, and lower the airplane to the ground.

n. Reduce hydraulic pressure to zero, disconnect the hydraulic power source, disconnect the electrical power source, replace the plug removed in step (j), and return the steering system to normal.

32.4.2.2 NOSE GEAR STEERING SYSTEM (AIRPLANES 5046 AND UP).

a. Make sure that the normal and standby hydraulic system accumulators are properly serviced according to the instructions in Chapter 12.

b. Accomplish steps b, c, d, and e of paragraph 32.4.2.1.

c. Connect external hydraulic power to the normal and standby hydraulic systems ground connections and adjust the power source to deliver no flow at 3,000 PSI and 3 gallons per minute at 2,700 PSI.

Note

Since the normal and standby systems are tested separately, only one ground test stand is necessary.

32.4.2.2 (Cont).

d. Place the HYD SYS NORM-1 STANDBY-2 switch in the "NORM-1" position and apply hydraulic power to the normal hydraulic system.

e. Accomplish steps (g), (h), (i), (j), (k), and (l) of paragraph 32.4.2.1 and then lower the airplane so that the nose wheels again rest on steel plates.

f. Place the HYD SYS NORM-1 STANDBY-2 switch in the "STANDBY-2" position and apply hydraulic power to the standby hydraulic system.

g. Accomplish steps (g), (h), (i), (k), (l), (m), and (n) of paragraph 32.4.2.1.

Note

> When operating on the STANDBY-2 system, seven to ten full consecutive actuations of the nose gear steering cylinder will cause the hydraulic fuse to close. Equalizing the hydraulic pressure for a brief period of time will allow the fuse to reset.

32.4.2.3 NOSE STEERING SYSTEM PULLING CHECKOUT. Occasionally during taxiing the airplane may tend to pull to one side or the other and opposite steering is required to maintain directional control. In the event that this occurs, corrective action should be undertaken as follows until the problem is cleared.

a. The nose wheel tires on the airplane should be matched/paired per manufacturer, tread pattern, tire pressure, and amount of wear. Any mis-match of the above features can cause the pull condition and can be corrected easily by tire/wheel replacement of tire pressure adjustment. Check the steering selector valve adjustment/rigging per paragraph 32.4.5.

b. Steering cable tension and wheel rigging are important. See figure 32-25 for rigging instructions.

c. Steering system pads in the nose wheelwell should be examined to be sure rigging is correct, and that spacing between pads is proper (no gap). Pads and shims may be identified in the IPB. Should a gap develop between these pads in service, steering can become loose and the airplane may tend to wander while taxiing. Any gap should be corrected by the addition of shims, P/N JL398-6, under each pad. Use the same number and thickness of shims under each side.

d. Check or replace the hydraulic system return filter. A dirty return line filter may restrict return fluid flow, particularly during high flow periods such as: thrust reverser, flap, steering and brake operation simultaneously upon landing. With restricted return fluid flow, brake pressure is not as rapidly relieved causing uneven brake drag which affects steering.

Note

> Production tolerances limit lateral movement of the upper scissor link to the steering collar to 0.012 inch to 0.024 inch maximum.
> Production tolerances at the apex of the two links are limited to 0.000 inch to 0.010 inch maximum,.
> Tolerances for the lower link to piston are 0.012 inch to 0.024 inch maximum.

e. Check the installation of nose gear scissors. There may be shims installed for lateral spacing because of in-service wear. Shimming should be accomplished equally on each side of the scissors links. If scissors links have been removed and reinstalled recently or have exececssive side clearance, this should be suspect.

32.4.3 TROUBLESHOOTING.

32.4.3.1 NOSE GEAR STEERING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NOSE WHEELS FAIL TO CENTER WHEN OFF THE GROUND.		
Steering system improperly rigged.	Check the rigging of the nose steering system.	Rig the nose steering system as needed.
Steering input mechanism from the steering wheel to the steering control valve binding or out of adjustment.	Check the operation and the adjustment of the steering input system.	Adjust or repair the mechanism as needed.
Nose strut centering cams defective, or strut piston binding.	Check the strut piston for freedom of rotation.	Replace or repair strut as needed.

32.4.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
STEERING SYSTEM OPERATIVE WITH NOSE WHEELS OFF THE GROUND AND WITH THE SCISSORS CONNECTED.		
Nose gear scissors switch defective or out of adjustment.	Check operation and adjustment of the nose gear scissors switch.	Replace or adjust the switch as needed.
Strut piston not extending properly.	Check to see if the strut lock is removed. Check extension of the strut.	Remove the strut lock if it is installed. Repair or replace the strut as needed.
Defective steering selector valve(s).	Check operation of the steering selector valve(s).	Repair circuit or replace valve as needed.
INSUFFICIENT NOSE WHEEL STEERING TRAVEL.		
Nose steering system improperly rigged.	Check the rigging of the nose steering system.	Rig the nose steering system as needed.
Defective steering control valve.	Check the operation of the steering control valve.	Repair circuit or replace defective valve as needed.
Defective steering actuating cylinder.	Check operation of the steering actuating cylinder.	Repair circuit or replace defective valve as needed.
Binding or interference in the steering system.	Check the entire steering system for binding or interference.	Relieve binding or interference as needed.
NOSE STEERING INOPERATIVE AFTER LANDING, BUT CONTROL RETURNS AFTER AIRPLANE HAS WARMED UP.		
Nose scissors switch contaminated by ice or moisture.	Check the operation of the switch. Check potting of the switch.	Replace switch as needed. Readjust switch if no defects are found.
Ice around steering pulley under flooring.	Check for moisture around the steering pulley under the floor.	Dry the area around the steering pulley, and clean the drain holes as needed.
NOSE STEERING SYSTEM INOPERATIVE.		
Disengaged or defective nose gear steering circuit breaker.	Check for disengaged or defective circuit breaker.	Engage or replace circuit breaker as needed.
Defective or improperly adjusted nose gear scissors switch.	Check the operation and adjustment of the nose gear scissors switch.	Replace or adjust the nose steering switch as needed.
STEERING PULLS IN ONE DIRECTION.		
Airplane moves right or left when steering should be straight.	Perform pulling checkout per paragraph 32.4.2.3.	Perform corrective action as indicated.

32.4.4 REMOVAL AND REPLACEMENT.

32.4.4.1 REMOVAL OF THE NOSE STEERING CONTROL VALVE.

WARNING

Depressurize the normal and standby hydraulic systems (refer to paragraph 32.4.1.1), and de-energize the auxiliary hydraulic pump circuits.

a. Remove the nut, washer, and bolt holding the steering crank to the valve body, being careful not to lose the bushing in the crank or the spacer between the crank and the valve body.

b. Remove the nut, washers, and bolt holding the steering control arm to the input shaft of the valve, being careful not to lose the spacer between the arm and the valve or the bushing in the arm.

c. Disconnect the four hydraulic lines from the valve. Cap or plug all openings.

d. Remove the nuts, washers, and bolts holding the valve to the strut, and remove the valve.

Note

Bertea Products, Pasadena, California Manual No. 32-50-2, Overhaul Instructions with Parts List, Nose Landing Gear Steering Valve Assembly, Part No. 50100 (Lockheed Part No. JL1351-9), gives complete overhaul and maintenance instructions for the nose landing gear steering valve.

32.4.4.2 REPLACEMENT OF THE NOSE STEERING CONTROL VALVE.

a. Reverse the removal procedure.

b. Make sure that all spacers and bushings are properly installed.

c. Bleed the steering system.

d. Perform an operational checkout of the nose steering system, and adjust the steering system as needed.

Note

After replacing the valve, make sure that the hydraulic lines to the valve do not touch the strut door bracket.

32.4.5 ADJUSTMENT.

32.4.5.1 NOSE STEERING SYSTEM RIGGING INSTRUCTIONS. For these instructions, see figure 32-25.

32.4.5.2 NOSE STEERING CONTROL VALVE ADJUSTMENT. (See figure 32-26.) Adjust steering valve assemblies, part number JL1351-7, according to these instructions whenever steering becomes sluggish in either direction. Steering valve assemblies, part number JL1351-9, are not adjustable.

a. Remove the valve end cover and the valve stem jamnut, and insert an Allen wrench in the socket at the valve stem end.

b. Adjust the valve stem travel to obtain a differential of less than 0.030 inch between stop lug strokes X and X_1 (see figure 32-26).

c. Install a new jamnut, and adjust the torque of the nut to 12 to 15 inch-pounds.

d. Hold the end cover in place and install the end cover screws. Adjust the torque of the screws to obtain 18 to 20 inch-pounds, and safetywire them together.

e. Pressurize the nose steering system and check for evidence of leaks.

f. Perform an operational checkout of the nose steering system.

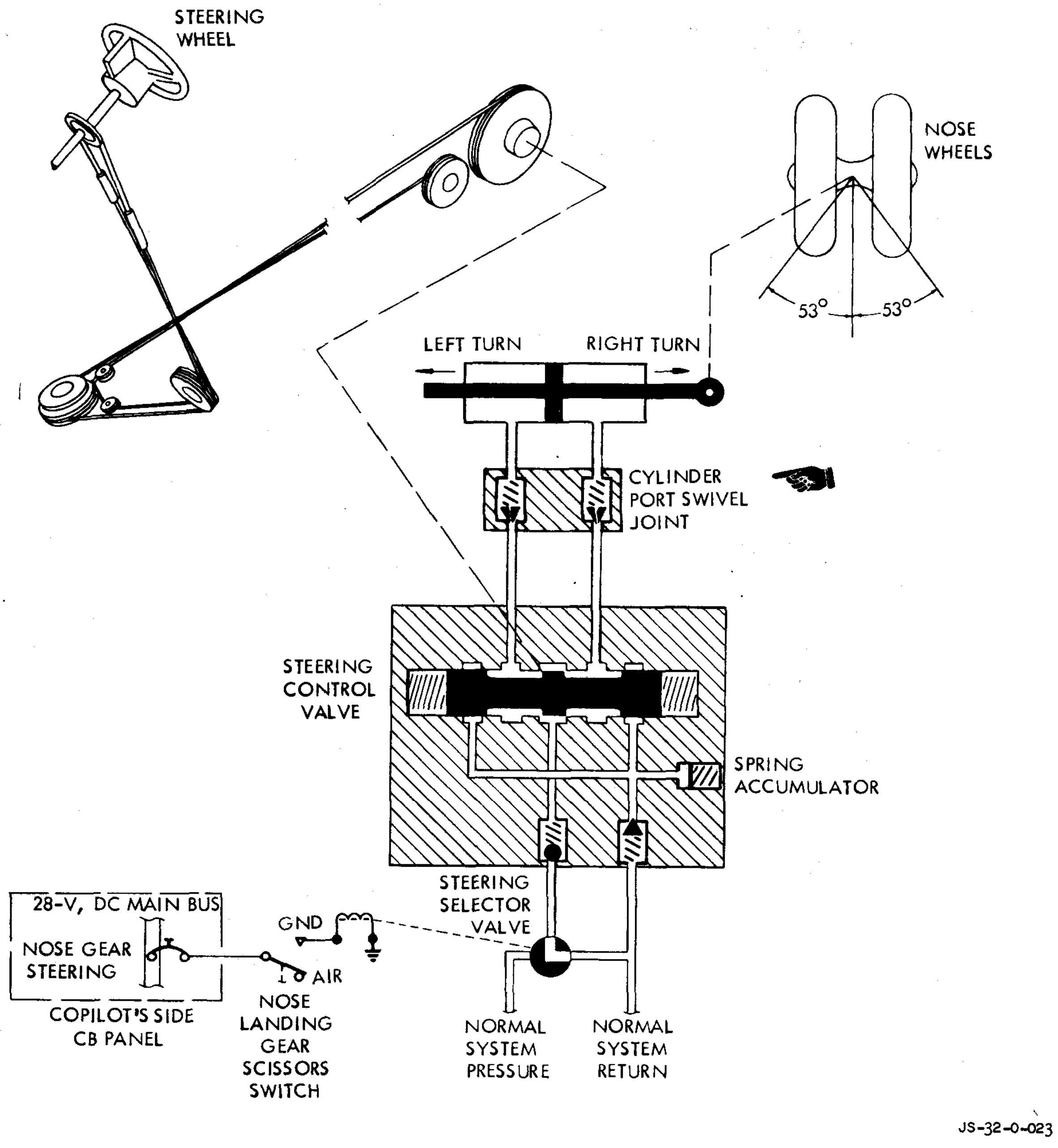


Figure 32-23. Nose Steering System Schematic Diagram (Airplanes 5001 thru 5045)

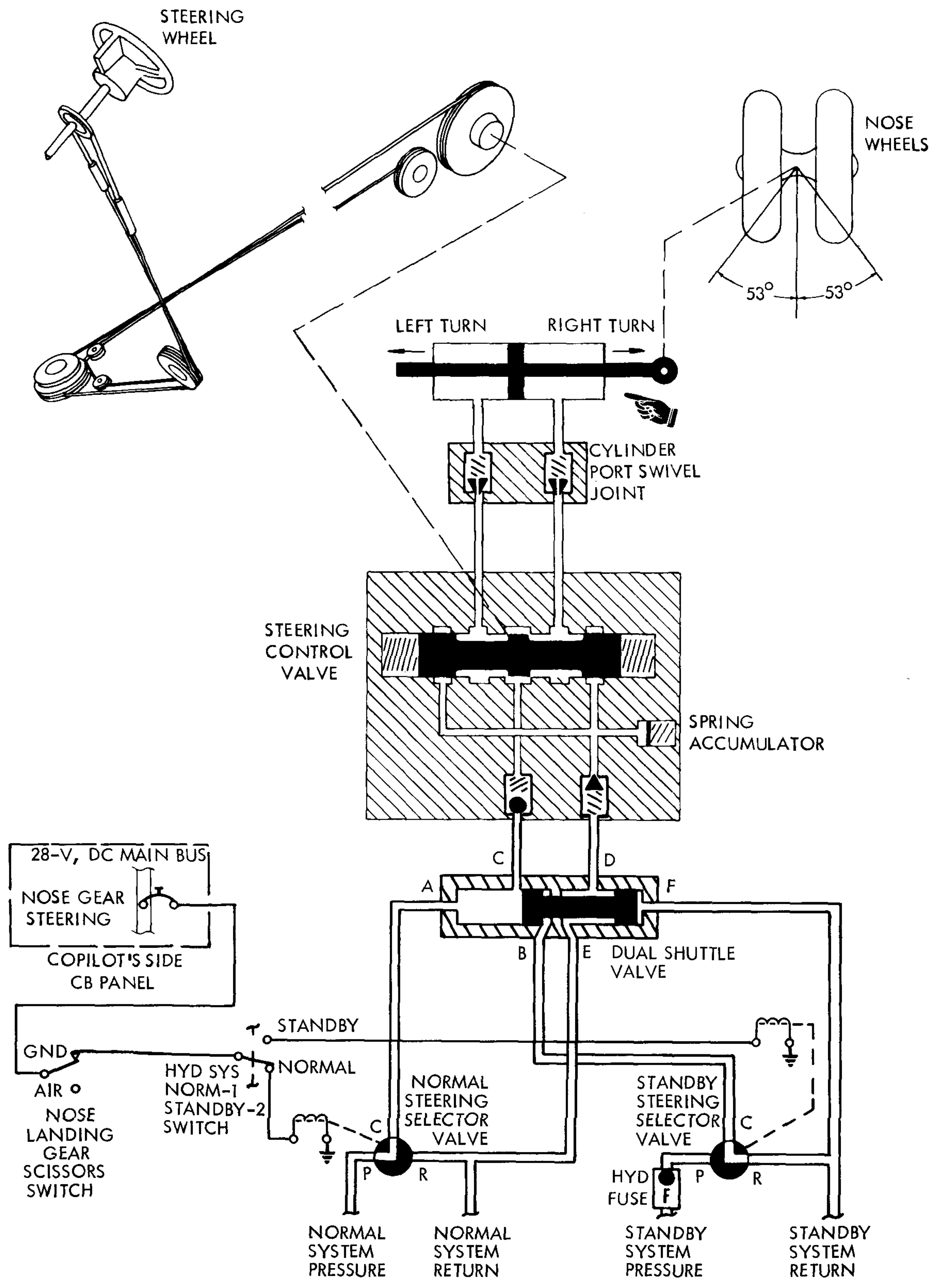


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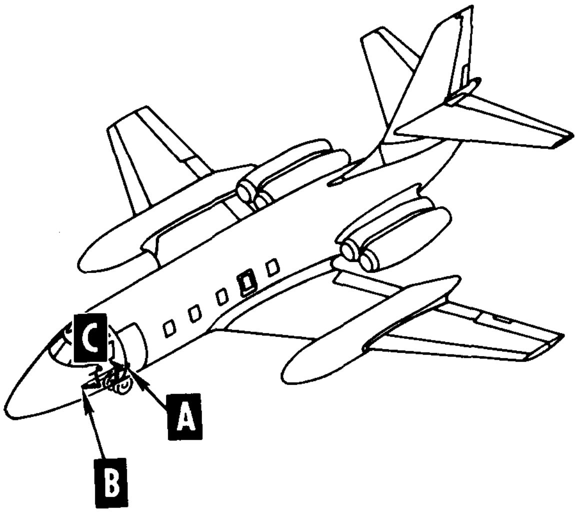
Figure 32-24. Nose Steering System Schematic Diagram (Airplanes 5046 and Up)



STEP 1

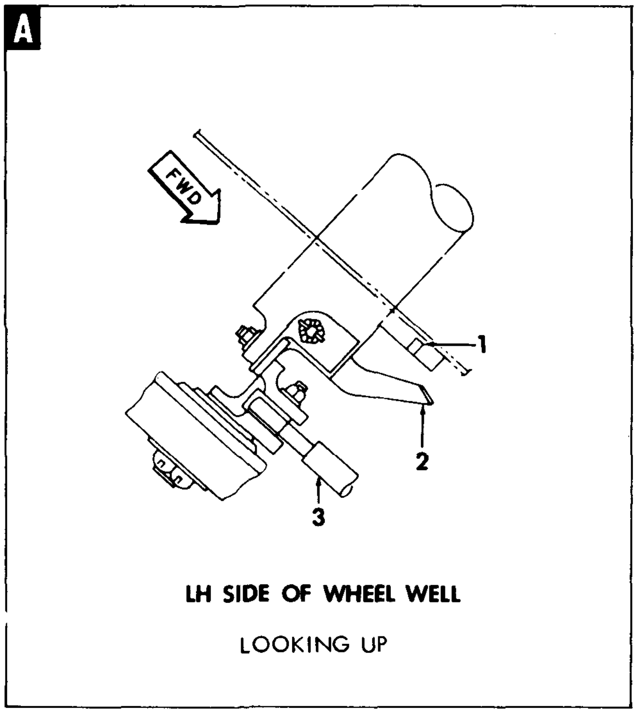
REMOVE THE FLOOR PANELS DIRECTLY OVER THE STEERING CABLES.

STEP 2

JACK THE NOSE OF THE AIRPLANE UNTIL THE NOSE WHEELS CLEAR THE GROUND.

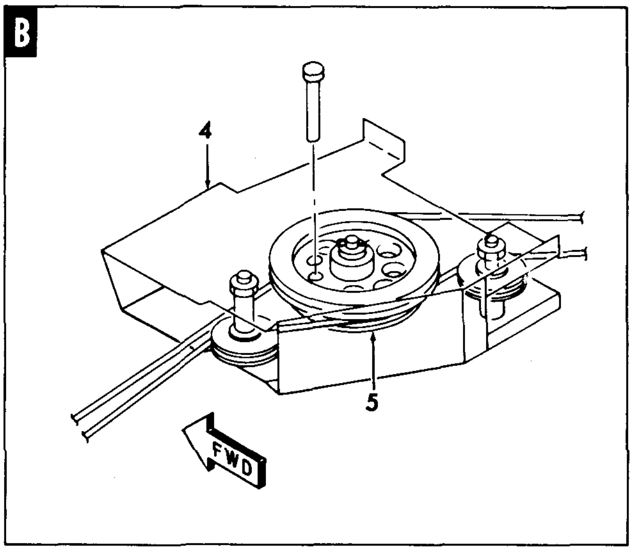
STEP 3

REMOVE THE NOSE GEAR GROUND SAFETY LOCK, AND MANUALLY RETRACT THE NOSE GEAR UNTIL THE STEERING DETENT LEVER (2) ENGAGES IN THE SLOT OF THE LOCKING PLATE (1).



STEP 4

INSTALL A 3/16-INCH DIAMETER RIG PIN THROUGH THE HOLE IN THE BRACKET (4) AND THE FORWARD PULLEY (5).



STEP 5

RIG THE LOWER CABLES (8) AND (11) TO A TENSION OF 50 (± 5) POUNDS AT 70°F OR EQUIVALENT.

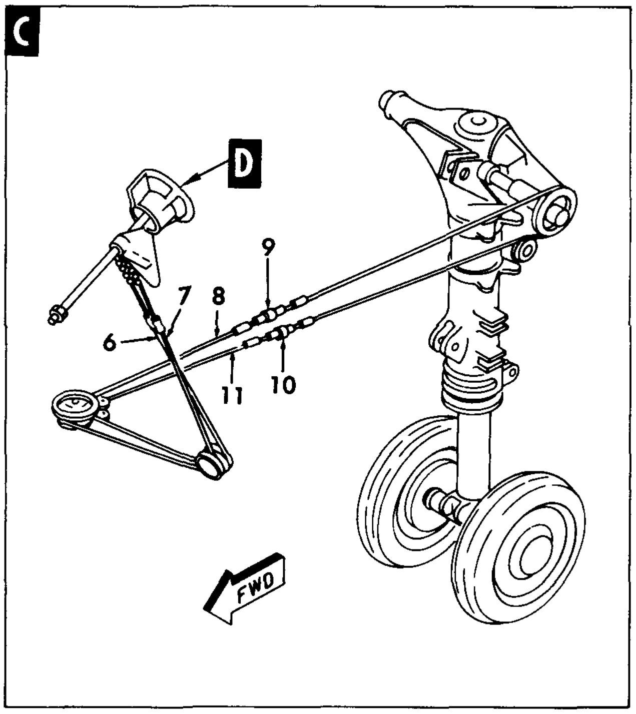


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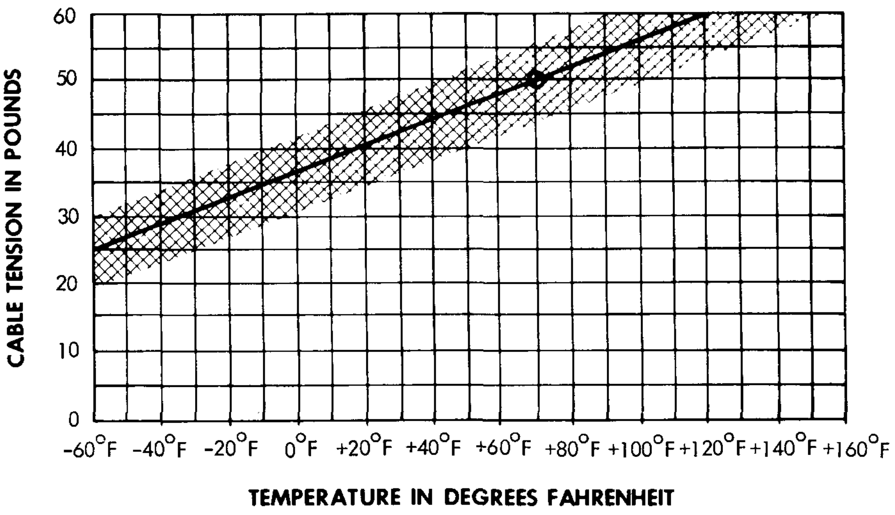
Figure 32-25. Nose Steering System Rigging Instrutions (Sheet 1 of 2)



NOTE

THIS CHART VALID ONLY WHEN TEMPERATURES OF THE CABLES AND THE STRUCTURE ARE STABILIZED.

STEP 6

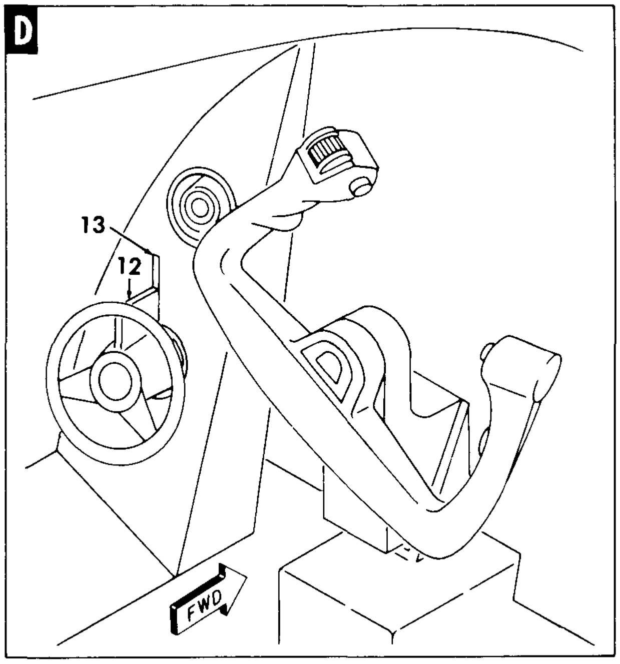
REPEAT STEP 5 FOR THE UPPER CABLES (6) AND (7).

STEP 7

ALIGN THE INDEX MARK (12) AS NEARLY AS POSSIBLE WITH THE MARK ON THE INSTRUMENT PANEL BY REMOVING THE STEERING WHEEL AND REPLACING IT IN THE DESIRED POSITION.

STEP 8

REMOVE THE RIG PIN INSTALLED IN STEP 4 . IF NECESSARY, MAKE MINOR ADJUSTMENTS ON THE LOWER TURNBUCKLES (9) AND (10) TO ALIGN THE INDEX MARKS (12) AND (13) PERFECTLY.



NOTE

MAINTAIN THE PROPER CABLE TENSION DURING ADJUSTMENT.

STEP 9

EXTEND THE NOSE GEAR TO THE DOWN-AND-LOCKED POSITION, AND REPLACE THE NOSE GEAR GROUND SAFETY LOCK.

STEP 10

CHECK THE ALIGNMENT OF THE LOCK PLATE (1) WITH THE STEERING DETENT LEVER (2). IF NECESSARY, ADJUST ALIGNMENT BY ADJUSTING THE LENGTH OF THE INPUT ROD (3).

STEP 11

RECHECK THE WHEEL INDEX MARK (12) FOR ALIGNMENT WITH THE MARK (13) ON THE PANEL.

STEP 12

LOCK-WIRE ALL TURNBUCKLES AND FITTINGS AS REQUIRED.

STEP 13

PERFORM AN OPERATIONAL CHECKOUT OF THE NOSE WHEEL STEERING SYSTEM.

STEP 14

LOWER THE NOSE OF THE AIRPLANE FROM THE JACKS, REPLACE THE FLOOR PANELS, AND RETURN THE STEERING SYSTEM TO NORMAL OPERATING CONDITION.

Figure 32-25. Nose Steering System Rigging Instrutions (Sheet 2 of 2)

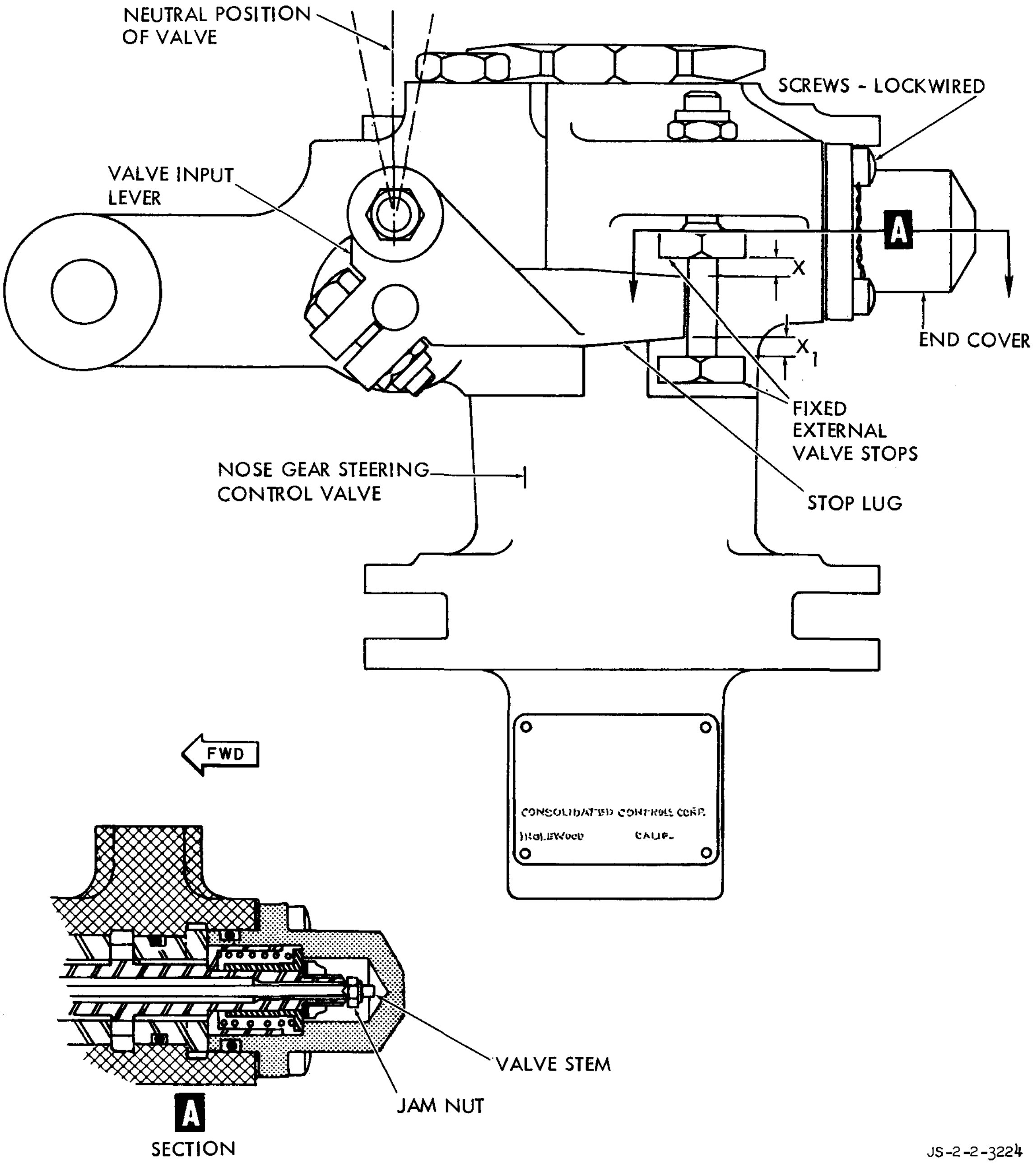
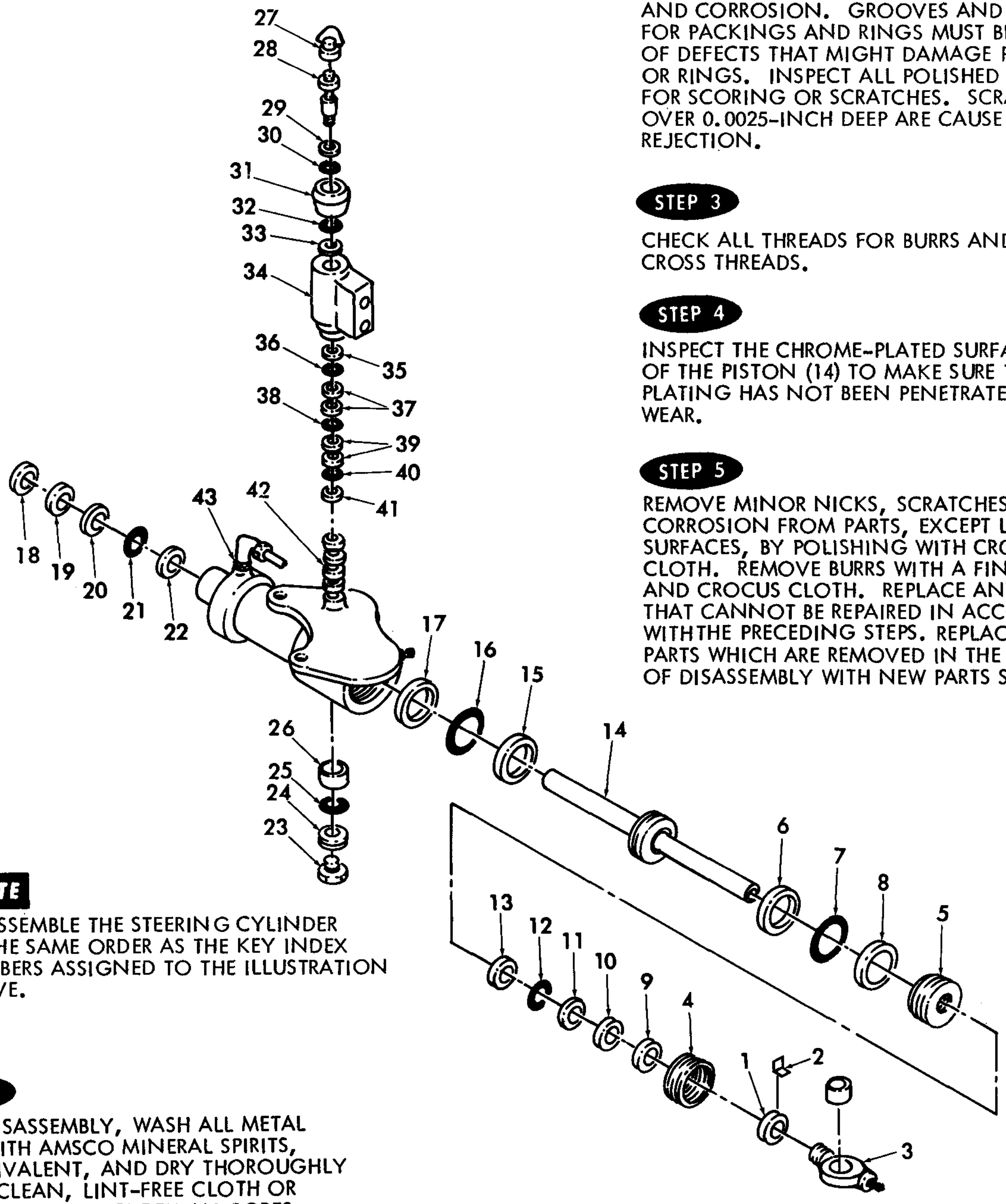


Figure 32-26. Adjustment of Nose Landing Gear Steering Control Valve No. JL1351-7



NOTE

DISASSEMBLE THE STEERING CYLINDER IN THE SAME ORDER AS THE KEY INDEX NUMBERS ASSIGNED TO THE ILLUSTRATION ABOVE.

STEP 1

AFTER DISASSEMBLY, WASH ALL METAL PARTS WITH AMSCO MINERAL SPIRITS, OR EQUIVALENT, AND DRY THOROUGHLY WITH A CLEAN, LINT-FREE CLOTH OR COMPRESSED AIR. CHECK ALL BORES AND PASSAGES FOR CLEANLINESS.

WARNING

AVOID BREATHING SOLVENT VAPORS, AND DO NOT ALLOW SOLVENT TO CONTACT THE SKIN. OBSERVE FIRE PRECAUTIONS.

STEP 2

USING A STRONG LIGHT AND MAGNIFYING GLASS, INSPECT ALL PARTS FOR CRACKS AND CORROSION. GROOVES AND SEATS FOR PACKINGS AND RINGS MUST BE FREE OF DEFECTS THAT MIGHT DAMAGE PACKINGS OR RINGS. INSPECT ALL POLISHED SURFACES FOR SCORING OR SCRATCHES. SCRATCHES OVER 0.0025-INCH DEEP ARE CAUSE FOR REJECTION.

STEP 3

CHECK ALL THREADS FOR BURRS AND CROSS THREADS.

STEP 4

INSPECT THE CHROME-PLATED SURFACE OF THE PISTON (14) TO MAKE SURE THE PLATING HAS NOT BEEN PENETRATED BY WEAR.

STEP 5

REMOVE MINOR NICKS, SCRATCHES, AND CORROSION FROM PARTS, EXCEPT LAPPED SURFACES, BY POLISHING WITH CROCUS CLOTH. REMOVE BURRS WITH A FINE FILE AND CROCUS CLOTH. REPLACE ANY PART THAT CANNOT BE REPAIRED IN ACCORDANCE WITHTHE PRECEDING STEPS. REPLACE ALL PARTS WHICH ARE REMOVED IN THE PROCESS OF DISASSEMBLY WITH NEW PARTS SUPPLIED.

STEP 6

DURING ASSEMBLY, ENSURE THAT HEAD (5) IS SEATED AGAINST SHOULDER OF CYLINDER (42). INSTALL NUT (4) FINGER TIGHT AND SAFETY.

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Figure 32-26A. Repacking Nose Steering Cylinder Assembly (Sheet 1 of 2)

NOSE STEERING CYLINDER COMPONENTS

1. NUT
2. TAB
3. END
4. NUT
5. HEAD
6. RING
7. PACKING
8. RING
9. RING
10. SCRAPER
11. FELT
12. PACKING
13. RING
14. PISTON
15. RING
16. PACKING
17. RING
18. RING
19. SCRAPER
20. FELT
21. PACKING
22. RING
23. LOCK
24. INSERT
25. PACKING
26. SLEEVE
27. CAP
28. BOLT
29. WASHER
30. PACKING
31. BODY
32. PACKING
33. GASKET
34. VALVE HOUSING
35. RING
36. PACKING
37. RING
38. PACKING
39. RING
40. PACKING
41. RING
42. CYLINDER ☛
43. GASKET

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Figure 32-26A. Repacking Nose Steering Cylinder Assembly (Sheet 2 of 2)

SECTION 32.5
EMERGENCY GEAR EXTENSION SYSTEM (AIRPLANES 5086 AND UP)

Contents Page

DESCRIPTION AND OPERATION 32-97
- General 32-97
- Nose Gear Door Release Pull Ring 32-97
- Nose Gear Up-Latch Release and Air Bottle Lever Access Plate 32-97
- LG Isolation Valve Tee Handle 32-97
- Main Landing Gear Emergency Release Lever 32-97
- Inspection Windows 32-98
- Emergency Gear Extension Air Bottles 32-98
- Dump Valves 32-98
- Vent Valves 32-98
- Check Valves 32-98

OPERATIONAL CHECKOUT 32-98
- Emergency Gear Release System 32-98

TROUBLESHOOTING 32-99
- Emergency Gear Release System 32-99

REMOVAL AND REPLACEMENT 32-99
- Removal of the Air Bottles 32-99

FIGURE
- 32-27. Landing Gear Emergency Extension System (Airplanes 5086 and Up) 32-100

32.5.1 DESCRIPTION AND OPERATION FOR AIRPLANES 5086 AND UP.

32.5.1.1 GENERAL. (See figure 32-27) The emergency gear extension system is actuated at each main gear and nose gear separately. The main gear is actuated by pulling a lever, which mechanically sequences door uplock release and main gear release. Separate handles are used for discharge of air bottles into the down side of the main gear actuating cylinders and nose gear actuating cylinder. The nose gear door is released by a pull ring with a tee handle which releases the nose gear uplock. No rigging adjustments required.

32.5.1.2 NOSE GEAR DOOR RELEASE PULL RING. A pull ring in the floor next to the pilot's seat is pulled to release the nose gear forward door uplock. This pull ring is connected to the nose gear door uplock by a cable.

32.5.1.3 NOSE GEAR UP-LATCH RELEASE AND AIR BOTTLE LEVER ACCESS PLATE. A rectangular plate in the floor of the pilots' compartment is removed to permit pulling the tee handle to release the nose gear. After the nose gear has been released, the air bottle is discharged into the downside of the nose gear actuator by turning the lever.

32.5.1.4 LG ISOLATION VALVE TEE HANDLE. An LG ISOLATION VALVE tee handle on the left wheel well is linked to a four-way, two-position hydraulic selector valve forward of the left wheel well. Pulling the handle positions the valve to by-pass the landing gear and main landing gear door selector valves to supply hydraulic power to the wheel brakes and nose steering system. It also opens the gear and gear door hydraulic lines to the system return line to prevent hydraulic locks in the gear actuating system during emergency gear extension.

32.5.1.5 MAIN LANDING GEAR EMERGENCY RELEASE LEVER. This lever is located at the forward side of each wheel well and is attached to a torque shaft mounted across the top of the wheel well. When in the stowed position, the lever rests against the stop attached to the forward side of the wheel well structure. This assembly sequences the release of the door up-locks and main gear up-lock so that the door is released before the main gear. The forward door up-lock is released by a rod and lever system and the aft door up-lock by a pull cable. The main gear is released by a cam lever action pulling the up-lock open. The right main landing gear, left main landing gear, and nose landing gear function mechanically, and are each operated separately.

32.5.1.6 INSPECTION WINDOWS. One inspection window is located in each main wheel well, in the inboard sides of the wheel well boxes. These windows provide a means of inspecting the main gear and door up-locks during flight.

32.5.1.7 EMERGENCY GEAR EXTENSION AIR BOTTLES. There are three air bottles, one for each landing gear. These are discharged into the downside of the landing gear actuators. The air bottle for the nose gear is operated by a lever in the pilots' compartment floor. The two air bottles for the main landing gears, one in each wheel well, are operated by a tee handle on the forward side of each wheel well. A pressure gauge and a filler valve are provided for each air bottle assembly to facilitate servicing. All bottles are equipped with relief valves. The relief valves protect bottles from overcharging and from the results of thermal expansion.

CAUTION

Neither No. 1 nor No. 2 air bottle should be discharged unless both main landing gears have had their door up-locks and gear up-lock released.

Either No. 1 air bottle or No. 2 air bottle will actuate both main landing gears together. The unused air bottle (No. 1 or No. 2) will then become a back-up air bottle.

32.5.1.8 DUMP VALVES. Hydraulic fluid dump valves are provided for the landing gear up and the main landing gear door close pressure lines. Opening the valve will relieve the pressure or pressure blockage in the associated line. The dump valves are located on the inboard wall of the right hand main wheel well and are accessible through the removable panel that supports the inspection window.

CAUTION

The dump valves are to be used only in accordance with instructions in the JetStar Airplane Flight Manual (SMR 150).

32.5.1.9 VENT VALVES. Vent valve assemblies are installed in the plumbing lines for the nose landing gear and main landing gear emergency extension air bottles. The vent valves are provided to facilitate maintenance following discharge of an air bottle for any reason.

32.5.1.10 CHECK VALVES. Several check valve assemblies are installed in the emergency landing gear release system hydraulic plumbing, positioned to prevent undesirable hydraulic fluid flow.

a. Check valve assemblies are installed in the discharge line from each of the three emergency extension air bottle assemblies. These check valves protect the bottle assemblies from reverse flow into the bottle, whether air or fluid, and prevent loss of air from one bottle through the vent of another bottle.

b. A check valve assembly is installed in the return line downstream of the isolation valve and protects against transient surges in the normal (No. 1) return system.

c. A check valve is installed in the pressure line downstream of the isolation valve, downstream of the tap off for steering control. This valve prevents loss of normal (No. 1) system brake pressure through the isolation valve plumbing.

d. Check valves are installed in the normal (No. 1) system pressure lines to the brake and steering systems. These check valves prevent loss of isolation valve pressure through normal (No. 1) system plumbing.

32.5.2 OPERATIONAL CHECKOUT.

32.5.2.1 EMERGENCY GEAR RELEASE SYSTEM. Before starting this checkout, make sure the main gear air bottles in the right and left wheel wells and the nose gear air bottle in the nose wheel well are charged to 1800 (+200, −0) psi, as shown by the air pressure gages on the bottle assemblies.

a. Jack the airplane in accordance with Chapter 7, using wing and fuselage jacks. The wheels must clear the ground.

b. Connect external hydraulic power to the normal hydraulic system, and connect external 28-volt DC power to the airplane. Adjust the output to no flow at 3,000 psi and 8 gallons per minute at 2,700 psi.

c. Remove the landing gear ground safety locks and retract the gear.

d. Shut down the hydraulic unit.

e. Deplete the normal hydraulic system pressure to zero, by cycling the aileron and elevator control systems with the BOOST HYD PRESS switches in the "ON" position.

f. With no pressure on the normal hydraulic system, place the gear control handle in the "LG DOWN" position.
RESULT: The landing gear actuating system should not be affected.

g. Pull the EMERGENCY NOSE GEAR DOOR RELEASE ring in the pilots' floor area.
RESULT: Nose gear doors should unlock.

h. Remove the EMERGENCY NOSE GEAR UPLATCH RELEASE access panel and pull the NOSE GEAR REL tee handle. If the nose gear falls into its downlock, mechanically release it from the downlock and leave the gear in the partially down position. Then rotate the NOSE GEAR AIR BOTTLE RELEASE lever.
RESULT: Nose gear should be down and locked.

32.5.2.1 (Cont).

i. Pull the EMERGENCY GEAR RELEASE levers on the front side of the wheel wells. Either lever can be pulled first.
RESULT: The door up-locks should release before the main gear uplock releases. The main gear should free fall.

CAUTION

Do not release the EMERGENCY GEAR RELEASE lever after pulling it to check the emergency extension of the main landing gears. Always manually return the lever to the stowed position.

CAUTION

BOTH MAIN LANDING GEARS MUST BE RELEASED BEFORE EITHER NO. 1 OR NO. 2 AIR BOTTLE IS RELEASED.

j. Pull either No. 1 or No. 2 air bottle MLG EXTENSION tee handle.
RESULT: Both main landing gears should be down and locked and the remaining air bottle should be full.

CAUTION

After returning the discharged air bottle lever to the normal position, bleed the air pressure trapped in the down side of the nose and main gear actuators through the bleed valves provided.

k. With all personnel clear of the gear doors, pressurize the normal hydraulic system to approximately 3,000 psi.
RESULT: The main gear inboard doors should close and lock.

CAUTION

After an emergency gear extension, always pressurize the down side of the gear system before retracting the gear. This will reset the shuttle valves to the normal operating position.

l. Cycle the landing gear six complete cycles to bleed all air from the system.

CAUTION

Cycle the gear according to the instructions for bleeding the normal and auxiliary hydraulic systems in Chapter 29. If excessive pressure is used during bleeding of the gear system, uplock damage could result.

m. Retract the landing gear.

n. Deplete hydraulic pressure in the brake system by operating the wheel brakes. Observe the wheel brakes, and observe the BRAKE PRESS LOW warning light on the annunciator warning panel.
RESULT: The BRAKE PRESSURE LOW warning light should go on.

o. Pull the LG ISOLATION VALVE tee handle forward of the left wheel well and observe the BRAKE PRESS LOW warning light.
RESULT: The warning light should go out indicating that brake pressure is available.

p. With the landing gear control handle in the "LG UP" position, follow the procedure as outlined in paragraphs g. thru l., except use the unused main landing gear air bottle for this operational check.

q. Return the pull ring and all tee handles to their normal (in) positions, and replace the nose gear uplock access plate.

r. Place the gear control handle in the "LG DOWN" position, reapply hydraulic pressure to the normal system, and then bleed the landing gear system according to the instructions in Chapter 29.

s. Install the landing gear ground safety locks, disconnect all external power, and lower the airplane to the ground.

t. Service the normal system reservoir and all three air bottles according to the instructions in Chapter 12.

32.5.3 TROUBLESHOOTING.

32.5.3.1 EMERGENCY GEAR RELEASE SYSTEM. Due to the simplicity of the system, any failures or malfunctions can be remedied by replacement or repair of the discrepant part or parts.

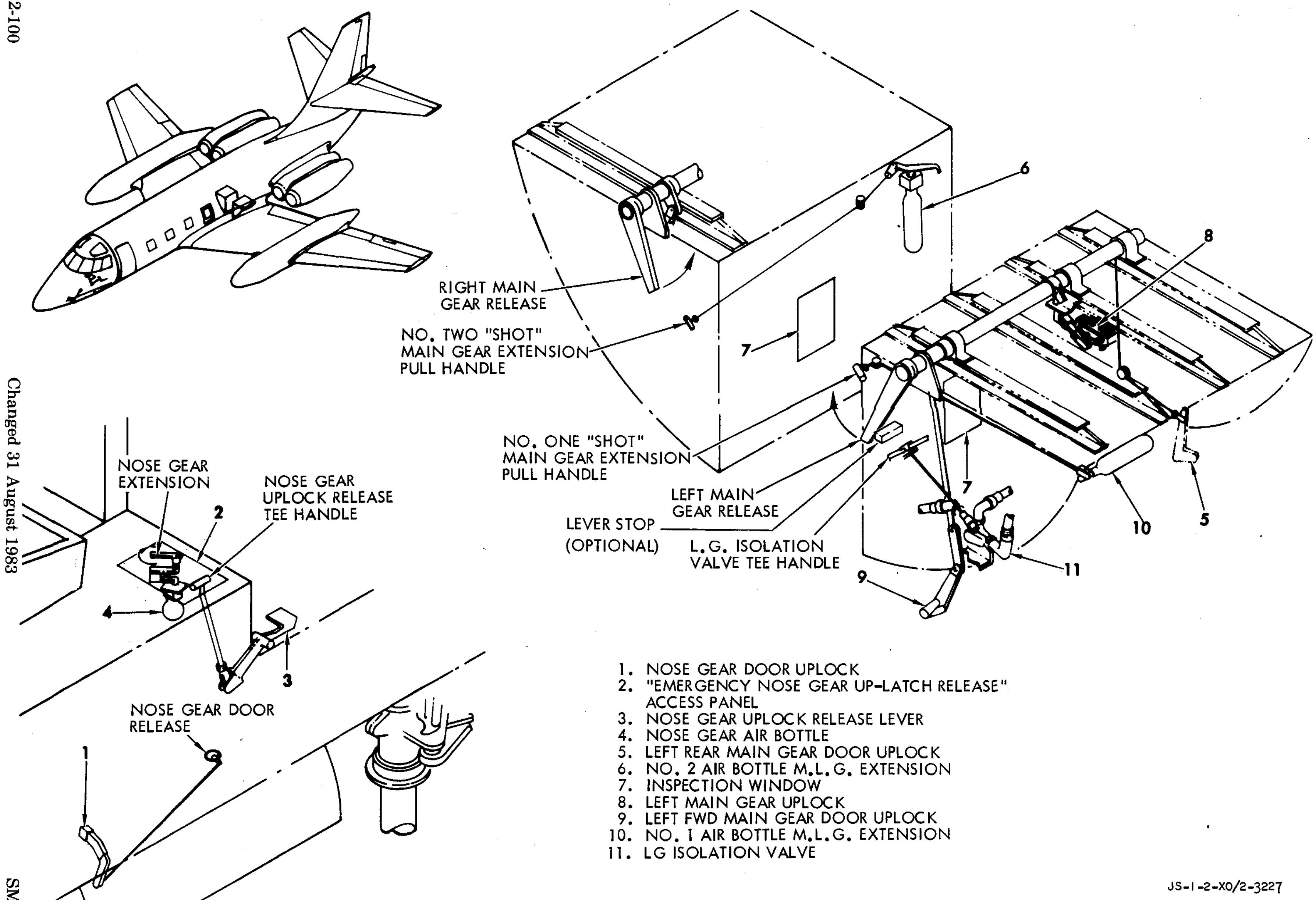
32.5.4 REMOVAL AND REPLACEMENT.

32.5.4.1 REMOVAL OF THE AIR BOTTLES.

WARNING

Keep personnel clear of the air discharge line during the bottle discharge.

a. Disconnect the emergency gear extention air line near the bottles and discharge by slowly opening the control valve. All connections are readily accessible for any removal or installation.


RIGHT MAIN GEAR RELEASE
NO. TWO "SHOT" MAIN GEAR EXTENSION PULL HANDLE
NO. ONE "SHOT" MAIN GEAR EXTENSION PULL HANDLE
LEFT MAIN GEAR RELEASE
LEVER STOP (OPTIONAL)
L.G. ISOLATION VALVE TEE HANDLE
NOSE GEAR EXTENSION
NOSE GEAR UPLOCK RELEASE TEE HANDLE
NOSE GEAR DOOR RELEASE
1
2
3
4
5
6
7
7
7
8
9
10
11
JS-1-2-X0/2-3227

1. NOSE GEAR DOOR UPLOCK
2. "EMERGENCY NOSE GEAR UP-LATCH RELEASE" ACCESS PANEL
3. NOSE GEAR UPLOCK RELEASE LEVER
4. NOSE GEAR AIR BOTTLE
5. LEFT REAR MAIN GEAR DOOR UPLOCK
6. NO. 2 AIR BOTTLE M.L.G. EXTENSION
7. INSPECTION WINDOW
8. LEFT MAIN GEAR UPLOCK
9. LEFT FWD MAIN GEAR DOOR UPLOCK
10. NO. 1 AIR BOTTLE M.L.G. EXTENSION
11. LG ISOLATION VALVE

Figure 32-27. Landing Gear Emergency Extension System (Airplanes 5086 and Up)

CHAPTER 33
LIGHTING

TABLE OF CONTENTS

Section Page

33.0 AIRCRAFT LIGHTING 33-3

33.1 WARNING CIRCUITS 33-15

SECTION 33.0
AIRCRAFT LIGHTING

Contents Page

DESCRIPTION AND OPERATION 33-3
  Exterior Lighting Systems 33-3
    Wing Inspection Light 33-3
    Anti-Collision Lights 33-3
    Navigation Position Lights 33-4
    Landing Lights 33-4
    Auxiliary Landing Lights 33-4
    Taxi Lights 33-4
  Interior Lighting Systems 33-4
    Map Lights 33-4
    Overhead Panel Floodlight 33-4
    Instrument and Control Panel Lights 33-4
    Vestibule Lights 33-5
    Aft Compartment Lights 33-5
    Ordinance Lights 33-5
    Cabin Lights 33-5

OPERATIONAL CHECKOUT 33-5

REMOVAL AND REPLACEMENT 33-6
  Exterior Lighting Systems 33-6
  Auxiliary Landing Light Clearance Check 33-6
  Landing Light Cover Assembly Removal 33-6
  Landing Light Cover Assembly Installation 33-6A
  Landing and Taxi Light Adjustment 33-7
  Auxiliary Landing Light Adjustment 33-7
  Landing Light and Taxi Light Adjustment 33-7

FIGURES

33-1. Exterior Lights-Locations and Controls 33-8
33-2. Exterior Lights Schematic Diagram 33-9
33-3. Lighting - Pilots' Compartment Schematic Diagram (Airplanes 5001 through 5088) 33-10
33-3A Lighting, Pilots' Compartment Schematic Diagram (Airplanes 5089 and Up) 33-10A
33-4. Lighting - Pilots' Compartment Location and Controls 33-11
33-5. Cabin Lighting Schematic Diagram (Airplanes 5001 thru 5140) 33-12
33-5A Cabin Lighting Schematic Diagram (Airplanes 5141 and Up) 33-13
33-6. Aft Compartment Lights Schematic Diagram 33-14
33-7. Landing Light Cover 33-14
33-8. Static Discharge Dissipating Strip Installation 33-14A
33-9. Landing Light Cover Sanding Drum Fixture 33-14B
33-10. Landing and Taxi Light Rigging 33-14C
33-11. Exterior Lights Replacement List 33-14D

33.0.1 DESCRIPTION AND OPERATION.
(See figures 33-1 and 33-2.)

33.0.1.1 EXTERIOR LIGHTING SYSTEMS. Included in the exterior lighting systems are a wing inspection light, two anti-collision (beacon) lights, three navigation position lights, two landing lights, one auxiliary landing light, and two taxi lights. The wing inspection light, the anti-collision (beacon) lights, and the navigation position lights are all controlled by switches on the exterior lights control panel. The landing lights, the auxiliary landing light, and the taxi lights are controlled by switches on the center console. All lights receive 28-volt DC through the copilot's wainscot circuit breaker panel from the main DC bus, except the anti-collision lights, which receive 28-volt DC power from the essential DC bus.

33.0.1.1.1 WING INSPECTION LIGHT. A wing inspection light on the side of the fuselage between the door and the first window illuminates the leading edge of the left wing. Power for this clear light is routed through a WING INSP EXTERIOR LIGHTS circuit breaker. The light is controlled by a two-position ("ON", "OFF") WING INSP light switch on the exterior lights control panel.

33.0.1.1.2 ANTI-COLLISION LIGHTS. Two red anti-collision (beacon) lights are provided; one atop the vertical stabilizer and one on the underside of the fuselage between the wings. The top light oscillates and the bottom light rotates. Each light assembly consists of a pair of lamps, a motor, and a noise filter. Both anti-collision lights are supplied 28-volt DC through a BEACON EXTERIOR LIGHTS circuit

33.0.1.2 (Cont).
breaker. The anti-collision lights are controlled by a two position ("ON", "OFF") BEACON lights switch on the exterior lights control panel.

33.0.1.1.3 NAVIGATION POSITION LIGHTS. The navigation position lights system consists of a red light on the left wing tip, a green light on the right wing tip, and a white light on the aft end of the fuselage. The white light assembly on the aft end of the fuselage contains two bulbs. All three lights of the navigation position lights system are controlled by a two-position ("ON," "OFF") NAV POS light switch on the exterior lights control panel. Power is supplied through a POS EXTERIOR LIGHTS circuit breaker.

33.0.1.1.4 LANDING LIGHTS. A landing light is mounted in the bottom forward tip of each external tank. Power is supplied through a LDG LH and a LDG RH EXTERIOR LIGHTS circuit breaker. Control is provided by three-position ("LANDING ON," "OFF," "TAXI ON") L and R LANDING-TAXI LIGHTS switches. When the L or the R LANDING-TAXI LIGHTS switch is placed in the "LANDING ON" position, a corresponding landing light relay is energized to complete the circuit to the landing light.

CAUTION

Do not exceed the duty cycle of 1 minute on and 5 minutes off for the normal landing lights while the airplane is on the ground. To exceed this duty cycle may result in excessive heating and damage to the light cover. This limitation does not apply to the auxiliary landing light.

33.0.1.1.5 AUXILIARY LANDING LIGHT. An auxiliary landing light is attached to the nose landing gear door. The auxiliary landing light swings down with the door as the door opens for landing gear extension. Power is supplied through an AUX LDG LT circuit breaker on the copilot's wainscot circuit breaker panel. Control is provided by a two-position ("ON," "OFF") AUX LDG LT switch on the center console and by the nose gear downlock switch. The nose gear downlock switch prevents operation of the light when the nose gear is not down and locked. When the AUX LDG LT switch is in the "ON" position, and the nose gear is down and locked, an auxiliary landing light relay is energized to complete the circuit to the light. Normally, all three landing lights may be used for landing. However, if ice accumulation on the nose of the external tanks renders the normal lights ineffective, the auxiliary landing light is adequate for landing.

33.0.1.1.6 TAXI LIGHTS. A taxi light in the top forward tip of each external tank is supplied power through a TAXI EXTERIOR LIGHTS circuit breaker. Control is provided by three-position ("LANDING ON," "OFF," "TAXI ON") L and R LANDING-TAXI LIGHTS switches. When the L or R LANDING-TAXI LIGHTS switch is placed in the "TAXI ON" position, a corresponding taxi light relay is energized to complete the circuit to the light.

33.0.1.2 INTERIOR LIGHTING SYSTEMS. Provision is made in the interior lighting systems for the instrument and control panel lighting, and for the pilots' compartment, cabin, and galley illumination. The cabin lights receive 28-volt, DC power at the galley light control panel, supplied by the main DC bus through a cabin power current limiter in the electrical power shield.

33.0.1.2.1 MAP LIGHTS. Both the pilot and copilot are provided with an overhead map light for utility purposes. The two lights are supplied 28-volt, essential DC bus power from the ignition bus through a MAP INTERIOR LIGHTS circuit breaker on the pilot's side circuit breaker panel. Each light is controlled by a switch which is part of the light assembly (see figure 33-3).

33.0.1.2.2 OVERHEAD PANEL FLOODLIGHT. The overhead flood light and the thunderstorm lights on the glareshield are supplied 28-volt essential DC power from the ignition bus through a DOME INTERIOR LIGHTS circuit breaker on the pilot's side circuit breaker panel.

a. (Airplanes 5001 through 5088.) A white and a red lamp are in the overhead panel floodlight assembly. The lights are controlled by a three-position ("WHITE," "OFF," "RED") FLT STA LIGHTS switch on the interior lights control panel. The FLT STA LIGHTS switch, when placed in the "WHITE" position, turns on the thunderstorm lights as well as the white dome light (see figures 33-3 and 33-4).

b. (Airplanes 5089 and up.) The flight station flood light and the thunderstorm lights are controlled by a three-position ("GLARESHIELD," "OFF," "FLT STA & GLARESHIELD") switch on the interior lights control panel (see figures 33-3A and 33-4A).

33.0.1.2.3 INSTRUMENT AND CONTROL PANEL LIGHTS. Six switches and controls provide for operation of the various instrument and control panel lights (see figures 33-3 and 33-4).

a. On airplanes 5001 through 5088, the overhead instrument lights are controlled by a three-position ("BRT," "OFF," "DIM") switch on the interior lights control panel. Power is supplied at 28-Volt DC through the ENG, OVHD and PED INSTR LTS circuit breaker, on the copilot's wainscot panel, from the essential DC bus. On airplanes 5089 and up, the overhead instrument lights are controlled by a variable resistor on the interior lights control panel, through a solid state dimmer control. Essential 28-volt DC bus power is supplied through the OVHD INSTR LTS circuit breaker on the copilot's wainscot panel (see figures 33-3A and 33-4A).

b. The engine, fuel panel, and center stand instrument lights are controlled by an eight-position

33.0.1.2.3 (Cont).

("OFF" through "MED" to "BRT") switch on the interior lights control panel; 28-volt DC power is supplied through the ENG, OVHD and PED INSTR LTS circuit breaker, on the copilot's wainscot panel, from the essential DC bus.

Note

On airplanes 5089 and up, the title of the circuit breaker is ENG and PED INSTR LTS.

c. The pilot's flight instrument lights are controlled by a variable ("OFF" to "BRT") transformer on the pilot's wainscot light control panel. Power is supplied at 115-Volt AC through the PILOT FLIGHT INSTR LTS circuit breaker, on the copilot's wainscot panel, from the essential AC bus.

d. The left console and vertical panel lights are controlled by a three-position ("BRT," "OFF," "DIM") switch on the pilot's wainscot light control panel. Power is supplied at 28-Volts DC through the LH SIDE CONSOLE INSTR LTS circuit breaker, on the copilot's wainscot panel, from the main DC bus.

e. The copilots' flight instrument lights are controlled by a variable ("OFF" to "BRT") transformer on the copilot's wainscot light control panel. 115-volt AC power is supplied through the COPILOT FLIGHT INSTR LTS circuit breaker, on the copilot's wainscot panel, from the main AC bus.

f. The right console and vertical panel lights are controlled by a three-position ("BRT," "OFF," "DIM") switch on the copilot's wainscot light control panel. 28-volt, DC power is supplied through the RH SIDE CONSOLE INSTR LTS circuit breaker, on the copilot's wainscot panel, from the main DC bus.

33.0.1.2.4 VESTIBULE LIGHTS (See figure 33-5). A vestibule light fixture, installed by the interior contractor, is supplied 28-volt, DC power from the essential DC bus, through a VESTIBULE LIGHT circuit breaker in the electrical power shield. Switches for the vestibule light may be as shown in figure 33-5 or as modified by the interior contractor.

33.0.1.2.5 AFT COMPARTMENT LIGHTS. Lamps controlled by either of two switches, one in the cabin at fuselage station 570, and one in the aft compartment, provide light for the aft compartment. The aft compartment lights receive 28-volt DC power from the main DC bus, through an AFT COMPT LTS circuit breaker on the copilot's wainscot circuit breaker panel (see figure 33-6).

33.0.1.2.6 ORDINANCE LIGHTS. The ordinance lights are supplied 28-volt, essential DC bus power from the ignition bus, through an ORD INTERIOR LIGHTS circuit breaker on the pilot's side circuit breaker panel. A two-position ("ON," "OFF") NO SMOKING light switch on the interior lights control panel controls a NO SMOKING ordinance light in the cabin. A two-position ("ON," "OFF") FASTEN BELTS light switch on the interior lights control panel controls a FASTEN BELTS ordinance light in the cabin and a RETURN TO CABIN ordinance light in the lavatory. Also, when the oxygen warning relay is energized the NO SMOKING ordinance lights go on (see chapter 35). (See figures 33-4 and 33-5.)

33.0.1.2.7 CABIN LIGHTS. The cabin lights are installed by the interior contractor according to the individual requirements of the owner. 28-volt, DC power for cabin lighting is supplied from the main DC bus in the power shield (see figure 33-5).

33.0.2 OPERATIONAL CHECKOUT. The following procedure can be used to check the operation of the exterior lights:

a. Turn off all switches.

b. Connect external power to the airplane.

c. Make sure the LH LDG EXTERIOR LIGHTS circuit breaker is closed.

d. Place the L LANDING-TAXI LIGHTS switch in the "L LANDING ON" position.
RESULT: The left landing light should go on.

e. Place the L LANDING-TAXI LIGHTS switch in the "OFF" position.
RESULT: The left landing light should go out.

f. Make sure the RH LDG EXTERIOR LIGHTS circuit breaker is closed.

g. Place the R LANDING-TAXI LIGHTS switch in the "R LANDING ON" position.
RESULT: The right landing light should go on.

h. Place the R LANDING-TAXI LIGHTS switch in the "OFF" position.
RESULT: The right landing light should go out.

CAUTION

Landing light ground operation is limited to a cycle of 1 minute "ON," 5 minutes "OFF." This limitation does not apply to the auxiliary landing light.

i. Make sure the AUX LDG LT circuit breaker is closed.

33.0.2 (Cont).

j. Place the AUX LDG LT switch in the "ON" position.
RESULT: The auxiliary landing light should go on.

k. Place the AUX LDG LT switch in the "OFF" position.
RESULT: The auxiliary landing light should go out.

l. Make sure the TAXI EXTERIOR LIGHTS circuit breaker is closed.

m. Place the L LANDING-TAXI LIGHTS switch in the "L TAXI ON" position.
RESULT: The left taxi light should go on.

n. Place the L LANDING-TAXI LIGHTS switch in the "OFF" position.
RESULT: The left taxi light should go on.

o. Place the R LANDING-TAXI LIGHTS switch in the "R TAXI ON" position.
RESULT: The right taxi light should go on.

p. Place the R LANDING-TAXI LIGHTS switch in the "OFF" position.
RESULT: The right taxi light should go out.

q. Make sure the WING INSP EXTERIOR LIGHTS circuit breaker is closed.

r. Place the WING INSP light switch in the "ON" position.
RESULT: The wing inspection light should go on.

s. Place the WING INSP light switch in the "OFF" position.
RESULT: The wing inspection light should go out.

t. Make sure the BEACON EXTERIOR LIGHTS circuit breaker is closed.

u. Place the BEACON lights switch in the "ON" position.
RESULT: Both the top and bottom beacon lights should go on. The top light should oscillate and the bottom light should rotate.

v. Place the BEACON lights switch in the "OFF" position.
RESULT: Both the top and bottom beacon lights should go out and motion should stop.

w. Make sure the POS EXTERIOR LIGHTS circuit breaker is closed.

x. Place the NAV POS light switch in the "ON" position.
RESULT: The two wing tip position lights and the tail position light should go on.

y. Place the NAV POS light switch in the "OFF" position.
RESULT: The two wing tip position lights and the tail position light should go out.

z. Disconnect external power from the airplane.

33.0.3 REMOVAL AND REPLACEMENT.

33.0.3.1 EXTERIOR LIGHTING SYSTEM. Refer to the replacement list in figure 33-11.

33.0.3.2 AUXILIARY LANDING LIGHT CLEARANCE CHECK. To make sure the auxiliary landing light is installed correctly, perform the following check:

a. Disconnect the nose gear forward doors, and open the doors by hand (see figure 32-9).

b. Jack the airplane according to instructions contained in chapter 7. and retract the nose gear.

c. With the nose gear in the up-and-locked position, close the left door and check that there is clearance of 0.38 inch around the auxiliary landing light. Check clearance of the electrical leads.

d. Extend the landing gear, and reconnect the door linkage.

e. Lower the airplane and remove the jacks.

33.0.3.3 LANDING LIGHT COVER ASSEMBLY REMOVAL.

a. Remove JWT27-6 landing light cover assembly (see figure 33-7) from the airplane. (Fourteen NAS 1623-1 screws).

b. The plexiglass cover is cemented to the fiberglass cover assembly and must be cut out. However, before cutting out lens, remove the two static discharge strips covering the lens seam. If care is exercised during removal of the strips, they may be reused. Carefully cut the polysulfide sealant away from each N-440-X plastic-nylon nut and remove each N-440-½ binder head screw (11 screws and nuts). Carefully cut sealant bonding aluminum strip to cover assembly, and peel strip back to MS20470A3 rivet attach points. (See figure 33-8.)

c. Cut and remove plexiglass cover JWT27-2 from the fiberglas cover JWT27-3. A fine tooth finishing blade (20 teeth per inch) in a Sabre or hand saw used for wood, can be used for cutting the plexiglass. The plexiglass cover is formed from 0.250 thick material. If the plexiglass cover is cracked or chipped, large remaining pieces may be knocked out without damaging the fiberglass cover.

33.0.3.3 (Cont).

d. Remove the remainder of the plexiglass cover from the inside edge of the fiberglass cover using a sharp putty knife. By prying carefully under the edge of the remaining plastic lens it is possible to 'pop' sections of it loose. Be careful not to damage the JWT 27-3 fiberglass cover. A small sanding drum mounted in a drill motor can be used for final clean up on the inside opening of the fiberglass cover. A sanding drum with a 1-inch face and 1-1/2-inch long using fine to medium grit sleeves should be sufficient. (See figure 33-9.)

33.0.3.4 LANDING LIGHT COVER ASSEMBLY INSTALLATION.

Note

The JWT 27-2 plexiglass cover is made by a forming operation which requires special clamping around the edges. This can result in wrinkles on these edges. These wrinkles have no detrimental effect on the part and are removed by the layout and trim requirements that follow.

a. Lay the new plexiglass cover inside the cover over the opening. Using a grease pencil, draw a line on the cover following the outline of the opening in the front of the cover assembly.

b. Remove the plexiglass cover and measure 0.75 inch from the grease pencil line. Cut and remove any excess plexiglass from the outer edge. Use a fine tooth finishing blade (20 teeth per inch) in a Sabre or hand saw. All saw blade teeth should have uniform height. Teeth of uneven height will cause excessive chipping.

c. From the grease pencil line to the edge, mill the outside (forward) edge to a depth which will allow the plexiglass cover to fit flush with the face of the fiberglass cover. The fiberglass cover is 0.080 (±0.03) thick at mating edge. This rabbit or stepped edge can be cut with a router and end mill cutter or sanding drum. A fixture to control depth and width of cut can be made as shown in figure 33-9.

d. The surfaces to be bonded should be water-sanded using 240 to 400 grit wet or dry sandpaper.

e. After fitting the plexiglass cover to the cover, the areas surrounding the surfaces to be bonded should be protected by masking with cellulose tape.

33.0.3.4 (Cont).

Cellulose tape:

Permacel	Mystic
741 Black	6270 Black
727 Red	6280 Red
726 Yellow	6280 Yellow

f. Using MIL-A-8576B Type II acrylic plastic adhesive, cement the plexiglass cover (see figure 33-7) to the fiberglas cone. The adhesive is a three-part cement and mixed as follows:

COMPONENT	QUANTITY
Component A-Base cement (viscous liquid)	118.3 milliliters
Component B-Catalyst (powder in capsule)	2.4 grams
Component C-Promoter (liquid)	5.0 milliliters

Mix component B into component A by stirring until component B is dissolved. Add component C and stir until thoroughly mixed. The mixture should be used within 30 minutes of mixing.

g. Apply cement with a brush or rod to both the plexiglass cover and fiberglass cone, and assemble immediately. Press the cover against the cone with just enough pressure to force out air bubbles. Excess pressure should be avoided. The assembly must be held together under pressure for at least 4 hours at 70°F (minimum cure time). A sand bag can be used to maintain contact pressure. Scrape any excess cement off onto the protective tape.

h. After the cement has cured, remove the protective tape previously applied. Any excess cement extruded onto the tape will be removed at this time.

i. Reinstall static discharge strips per figure 33-8.

Note

Adhesive referred to in figure 33-8 is MIL-S-8802, Class B-1/2, sealing compound. Refer to figure 51-6.

k. Touch-up paint areas where required. DO NOT PAINT static discharge strips.

l. To clean transparent plastic cover, wash with a non-abrasive soap or detergent and water. Dry with a clean damp chamois. To remove oil or grease, use kerosene or aliphatic naphtha. Do not use gasoline, alcohol, benzene, sylene, acetone, carbon tetrachloride, fire extinguisher or de-icing fluids, lacquer thinners or window cleaning sprays on the acrylic plexiglass cover because they will soften the plastic and/or cause crazing.

m. Reinstall JWT 27-6 landing light cover assembly on airplane.

n. The MIL-A-8576B acrylic plastic adhesive may be procured from the following:

Commercial Plastic Company
352 McGrath Highway
Summerville, Mass.
Phone 867-823-2100
P/N PS-1818-18 cement

Cadillac Plastic Company
A Division of Dayco
1500 Carroll Drive
Atlanta, Ga. Phone 404-351-9800
P/N PS-18

727 West Lake, Chicago, Ill.
148 Parkway, Kalamazoo, Michigan
2305 Beverly Blvd., Los Angeles, Calif.
313 Corey Way, South, San Francisco, Calif.
3818 Red Bank Rd., Cincinnati, Ohio
4533 Willow Parkway, Cleveland, Ohio
2546 Irving Blvd., Dallas, Texas
736 Renaud Ave., Montreal (Dorval, Quebec)
35-21 Vernon Blvd., L.I. City, N.Y.
295 Buell Rd., Rochester, N.Y.
600 Seco Rd., Pittsburgh, Pa.
1152 E. Indian School Rd., Phoenix, Ariz.
15111 Second Ave., Detroit, Mich.

Rohm and Haas (PS-18)
Washington Square, Philadelphia, Pa.

33.0.4 LANDING AND TAXI LIGHT ADJUSTMENT. Jack the airplane per section 7.0 in this manual until the main landing gear tire clearance is 1.00 inch with the struts extended and the airplane level.

33.0.4.1 AUXILIARY LANDING LIGHT ADJUSTMENT.

a. Loosen the three screws securing the lamp mounting bracket to the nose wheel door.

b. Turn the adjusting bolt nuts until the desired lamp attitude is obtained. (See figure 33-10.)

c. Secure the mounting screws.

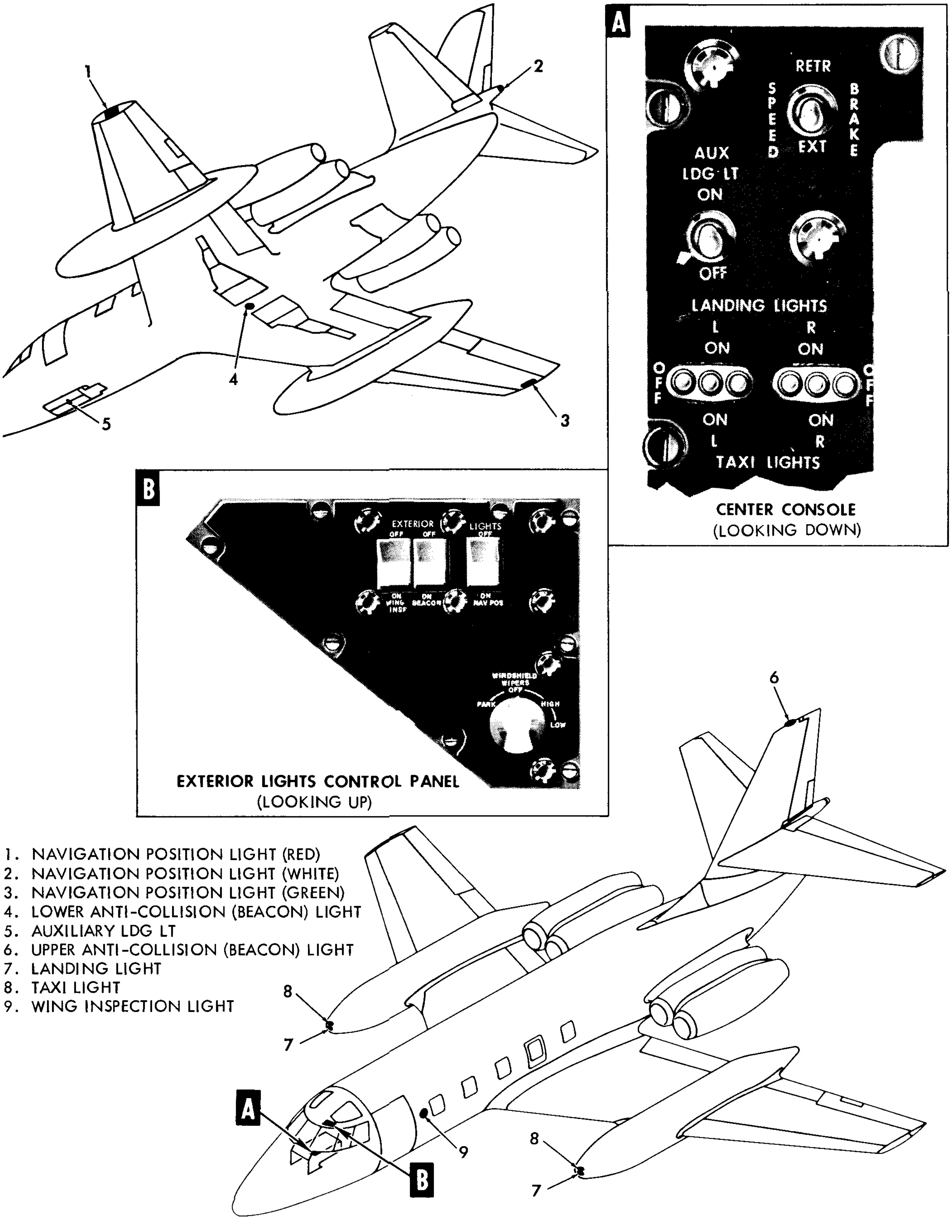
33.0.4.2 LANDING LIGHT AND TAXI LIGHT ADJUSTMENT.

a. Remove the JWT 27-6 landing light cover assembly from the airplane (See figure 33-7.)

b. The vertical pitch of the landing light (lower light) is adjusted by adding or removing AN960D10 washers to the lower mounting bolt. Horizontal adjustment is obtained by transferring washers between the two remaining mounting bolts.

c. Vertical pitch of the taxi light (upper light) is adjusted by turning the upper mounting screws. Horizontal adjustment is obtained by turning the two remaining screws in opposite directions by an equal amount. (See figure 33-10.)

d. Re-install the landing light cover assembly.



1. NAVIGATION POSITION LIGHT (RED)
2. NAVIGATION POSITION LIGHT (WHITE)
3. NAVIGATION POSITION LIGHT (GREEN)
4. LOWER ANTI-COLLISION (BEACON) LIGHT
5. AUXILIARY LDG LT
6. UPPER ANTI-COLLISION (BEACON) LIGHT
7. LANDING LIGHT
8. TAXI LIGHT
9. WING INSPECTION LIGHT

Figure 33-1. Exterior Lights—Locations and Controls

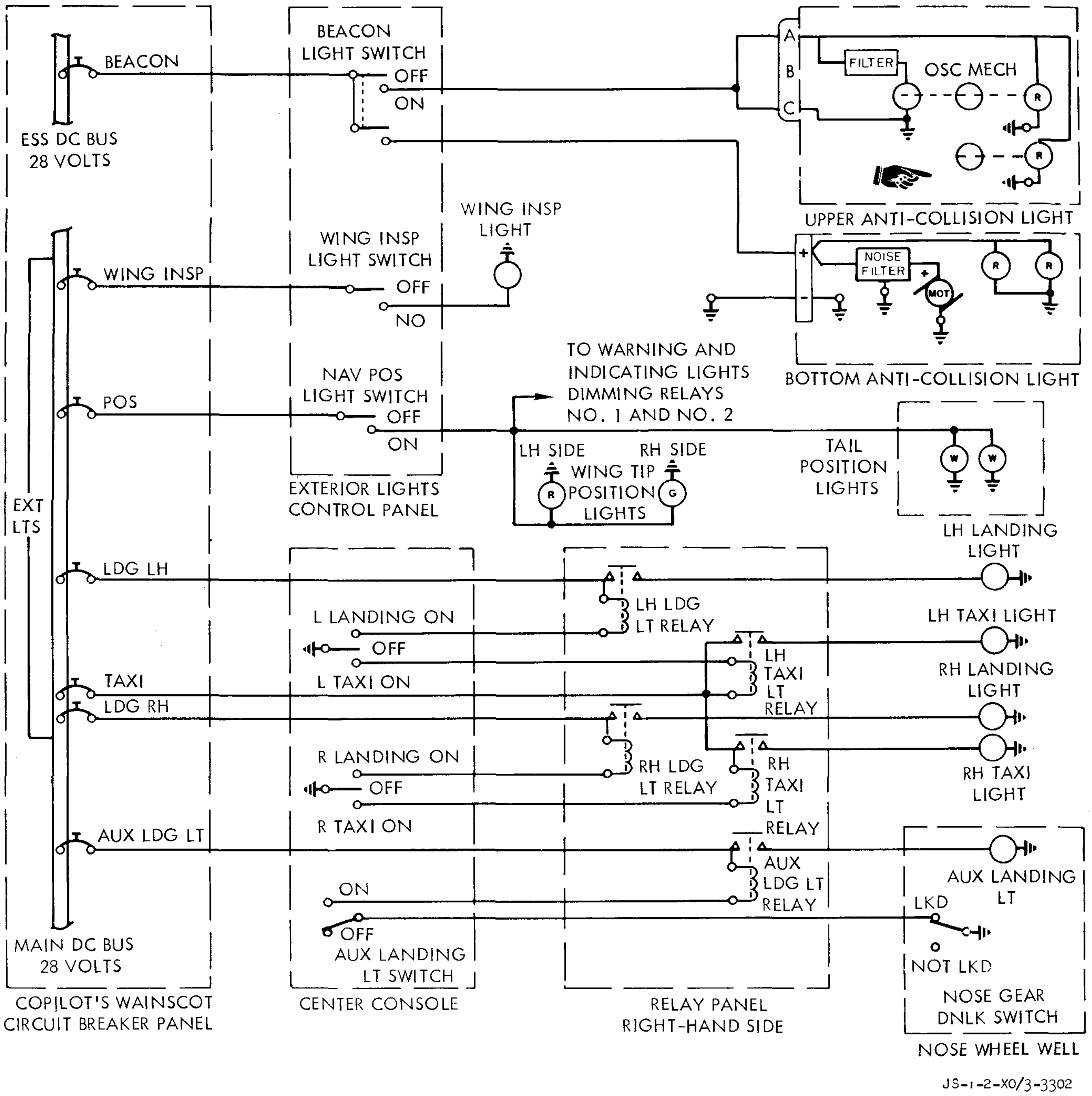


Figure 33-2. Exterior Lights-Schematic Diagram

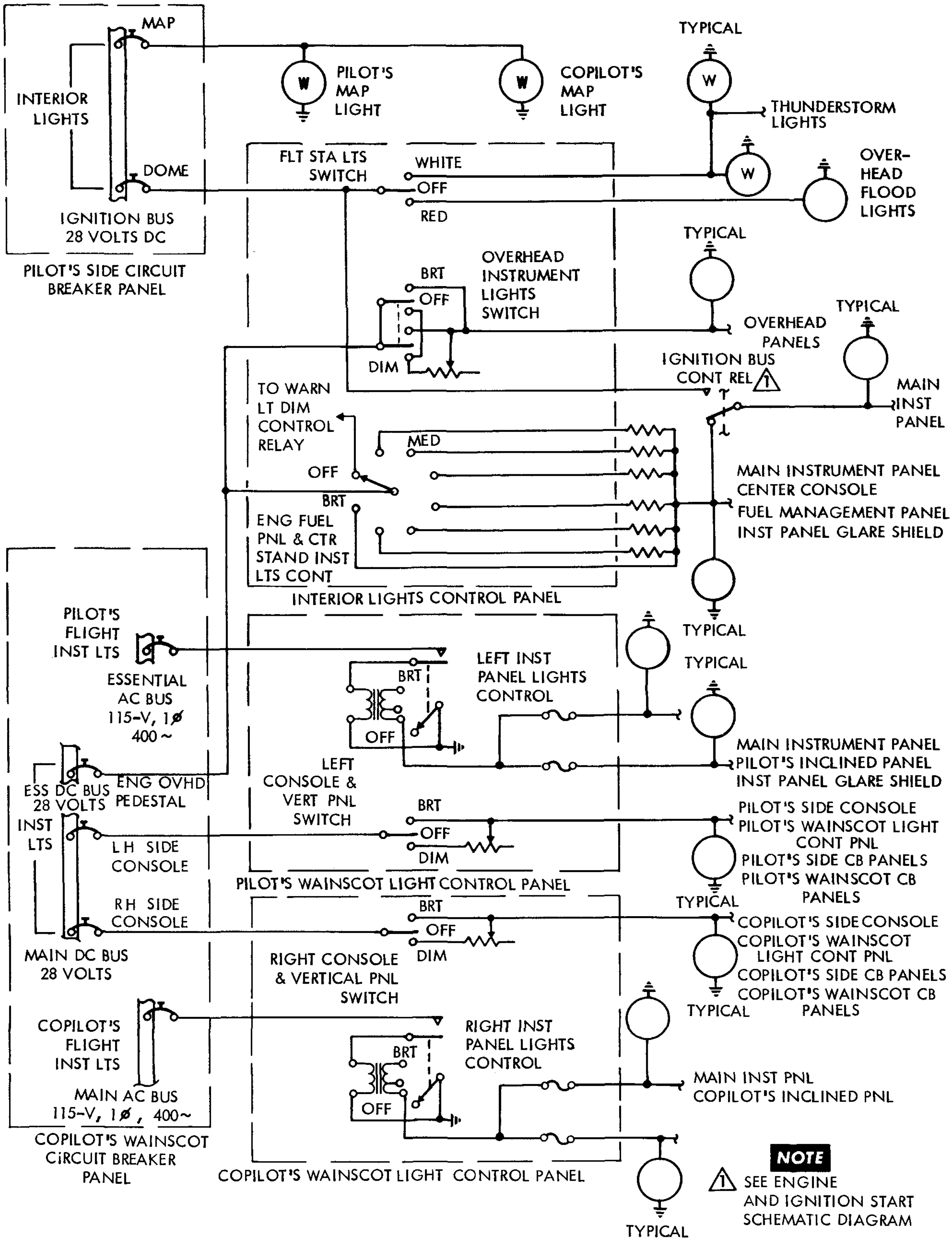
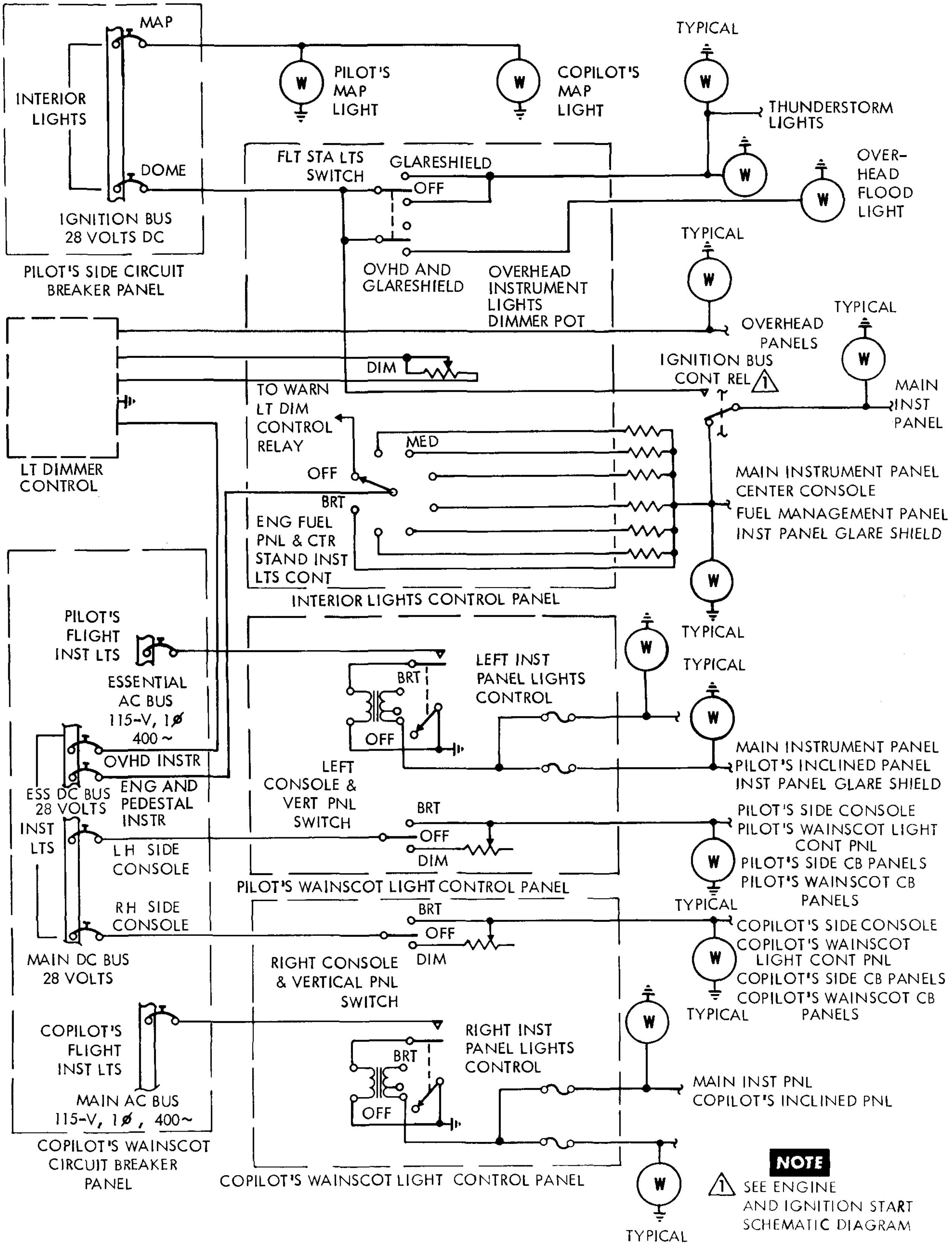


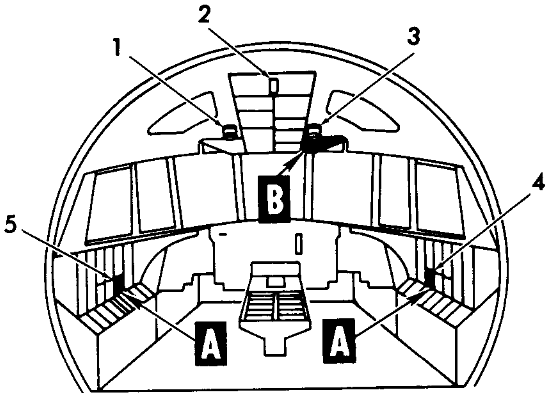
Figure 33-3. Lighting-Pilots' Compartment Schematic Diagram (Airplanes 5001 thru 5088)

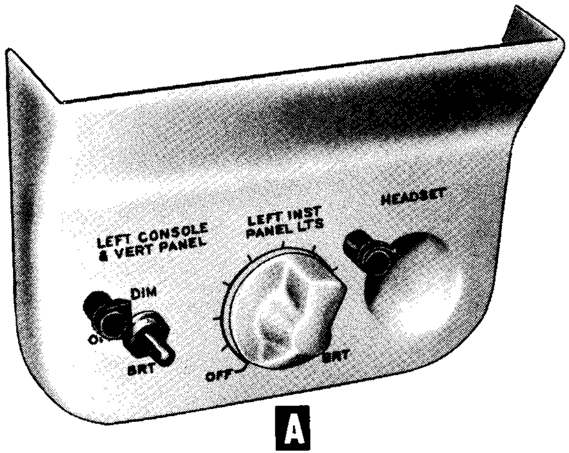


NOTE

1 SEE ENGINE AND IGNITION START SCHEMATIC DIAGRAM

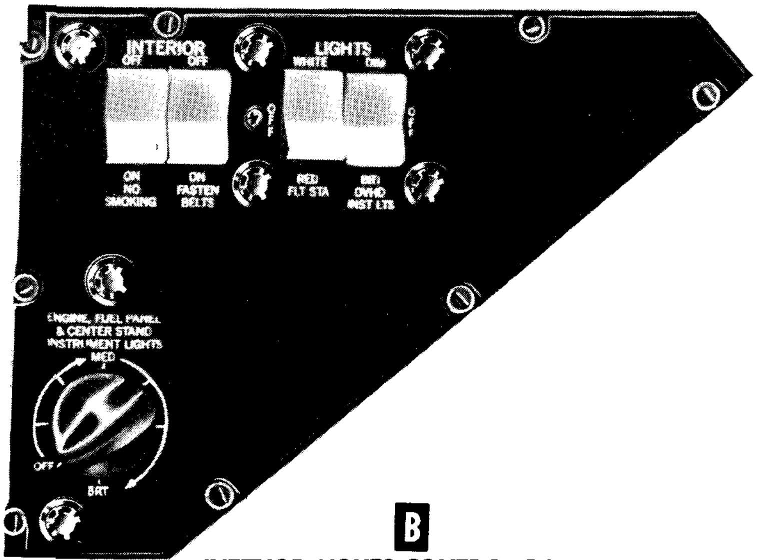
Figure 33-3A. Lighting-Pilots' Compartment Schematic Diagram (Airplanes 5089 and Up)





PILOT'S WAINSCOT LIGHT CONTROL PANEL ⚠1
(LOOKING OUTBOARD)

1. PILOT'S MAP LIGHT
2. OVERHEAD PANEL FLOOD LIGHT
3. COPILOT'S MAP LIGHT
4. COPILOT'S WAINSCOT LIGHTS CONTROL PANEL
5. PILOT'S WAINSCOT LIGHT CONTROL PANEL



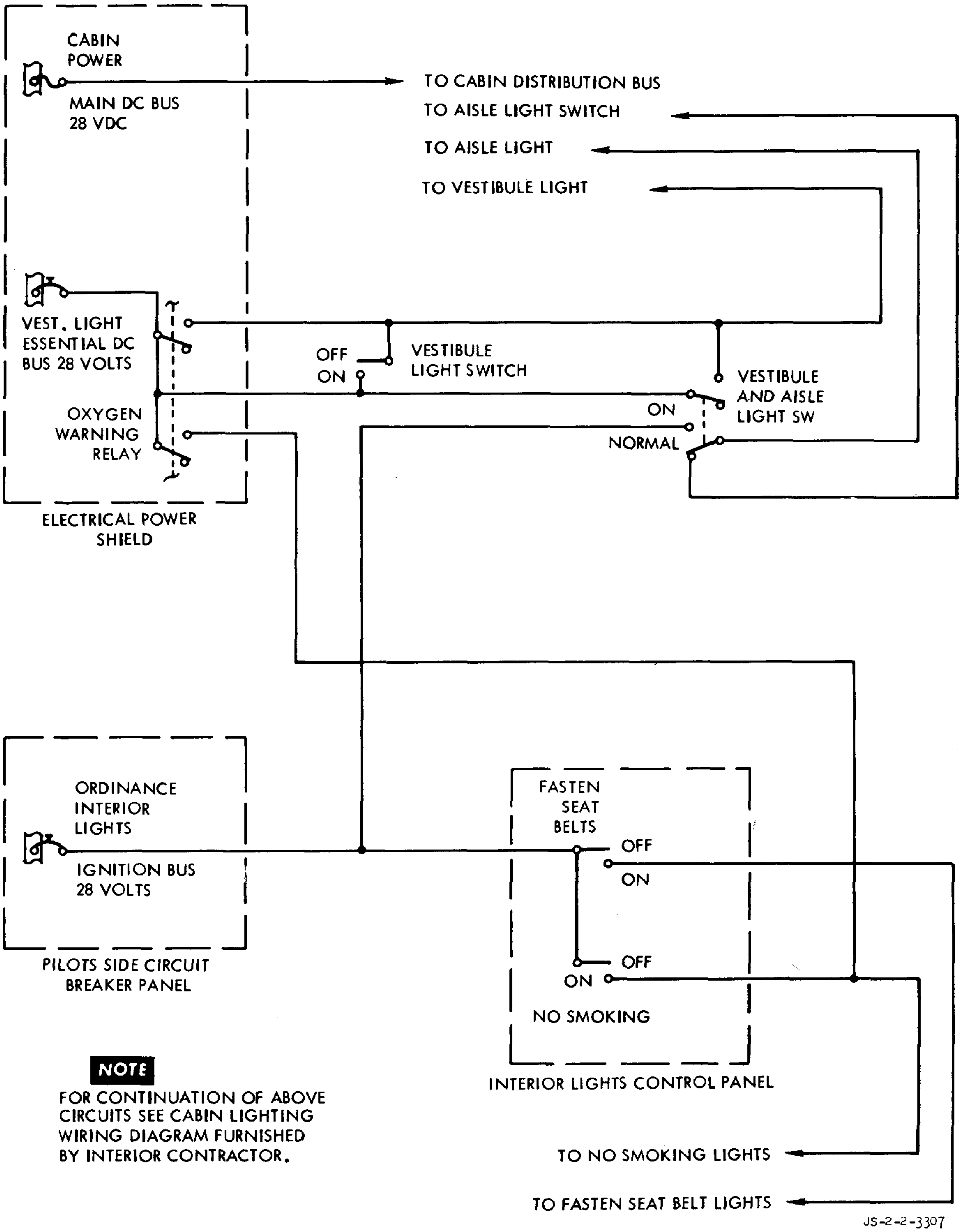
INTERIOR LIGHTS CONTROL PANEL
(LOOKING UP)

NOTE

⚠1 THE COPILOT'S WAINSCOT LIGHT CONTROL PANEL IS THE SAME AS THE PILOT'S EXCEPT THAT "LEFT" IS REPLACED WITH "RIGHT".

JS-2-2-3308

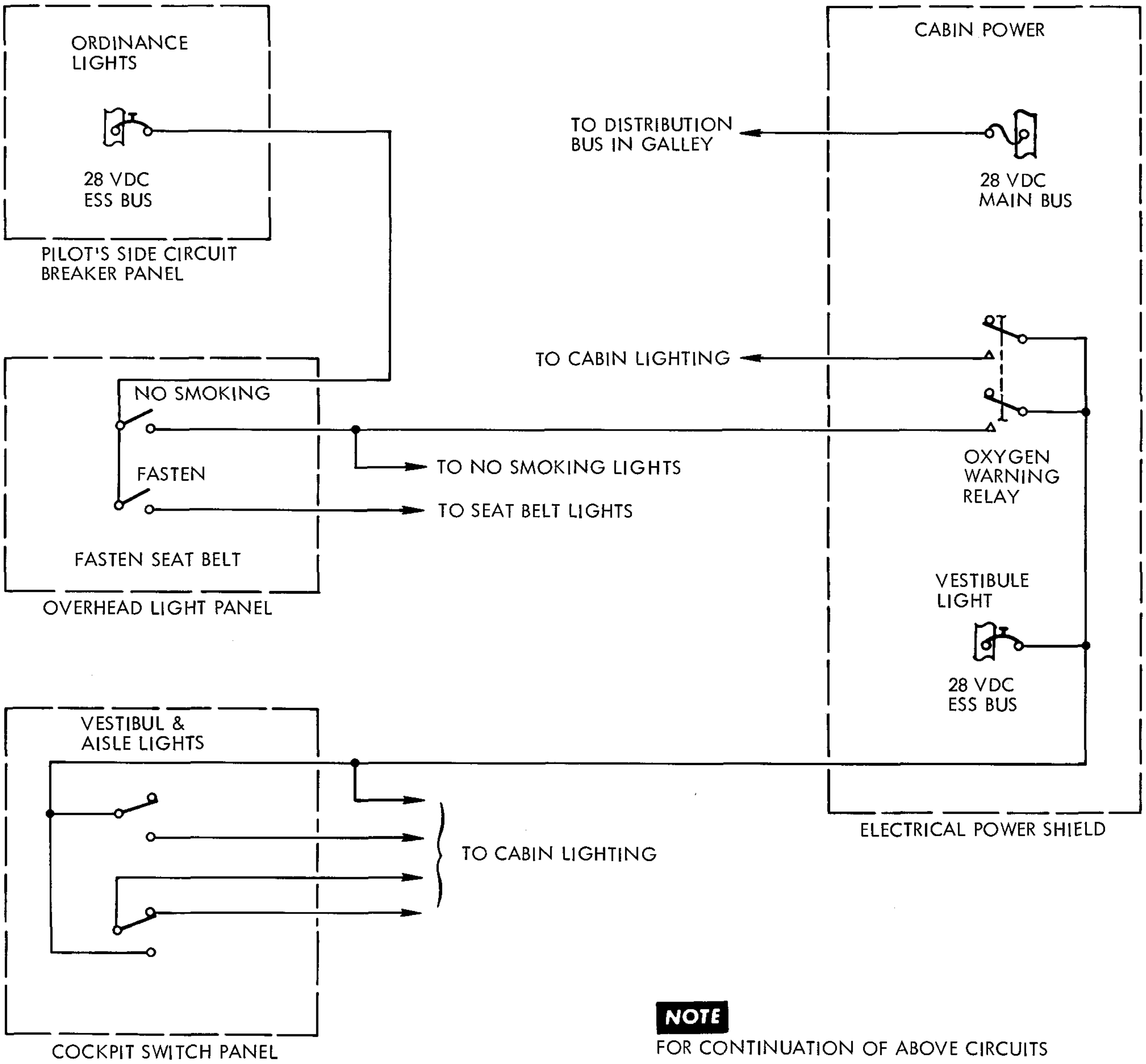
Figure 33-4. Lighting—Pilots' Compartment Location and Controls



NOTE

FOR CONTINUATION OF ABOVE CIRCUITS SEE CABIN LIGHTING WIRING DIAGRAM FURNISHED BY INTERIOR CONTRACTOR.

Figure 33-5. Cabin Lighting Schematic Diagram (Airplanes 5001 thru 5140)



NOTE

FOR CONTINUATION OF ABOVE CIRCUITS SEE CABIN LIGHTING WIRING DIAGRAMS FURNISHED BY INTERIOR CONTRACTOR

Figure 33-5A. Cabin Lighting Schematic Diagram (Airplanes 5141 and Up)

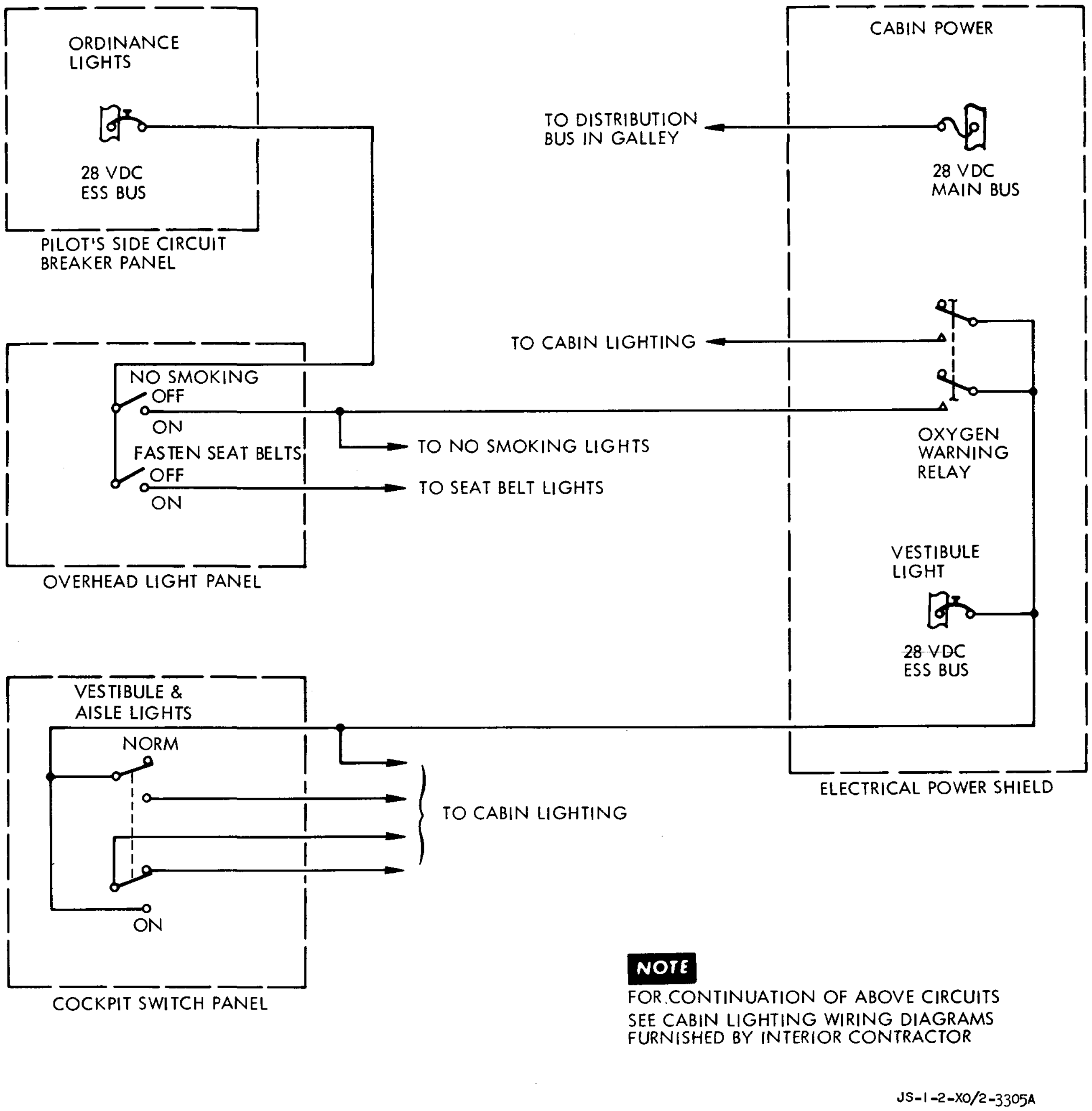
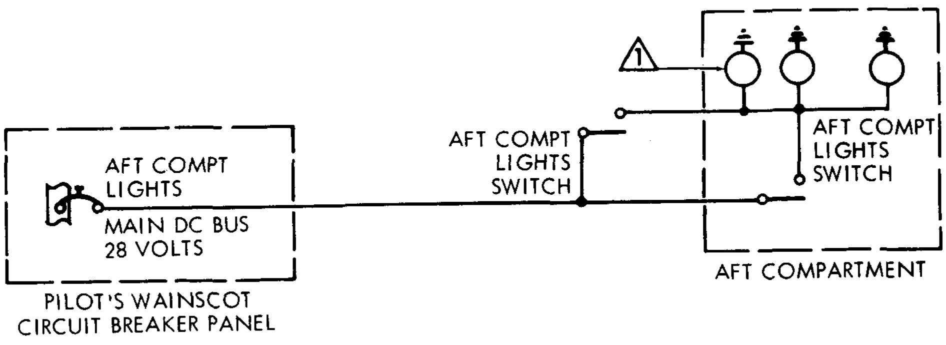


Figure 33-5A. Cabin Lighting Schematic (Airplanes 5141 and Up)

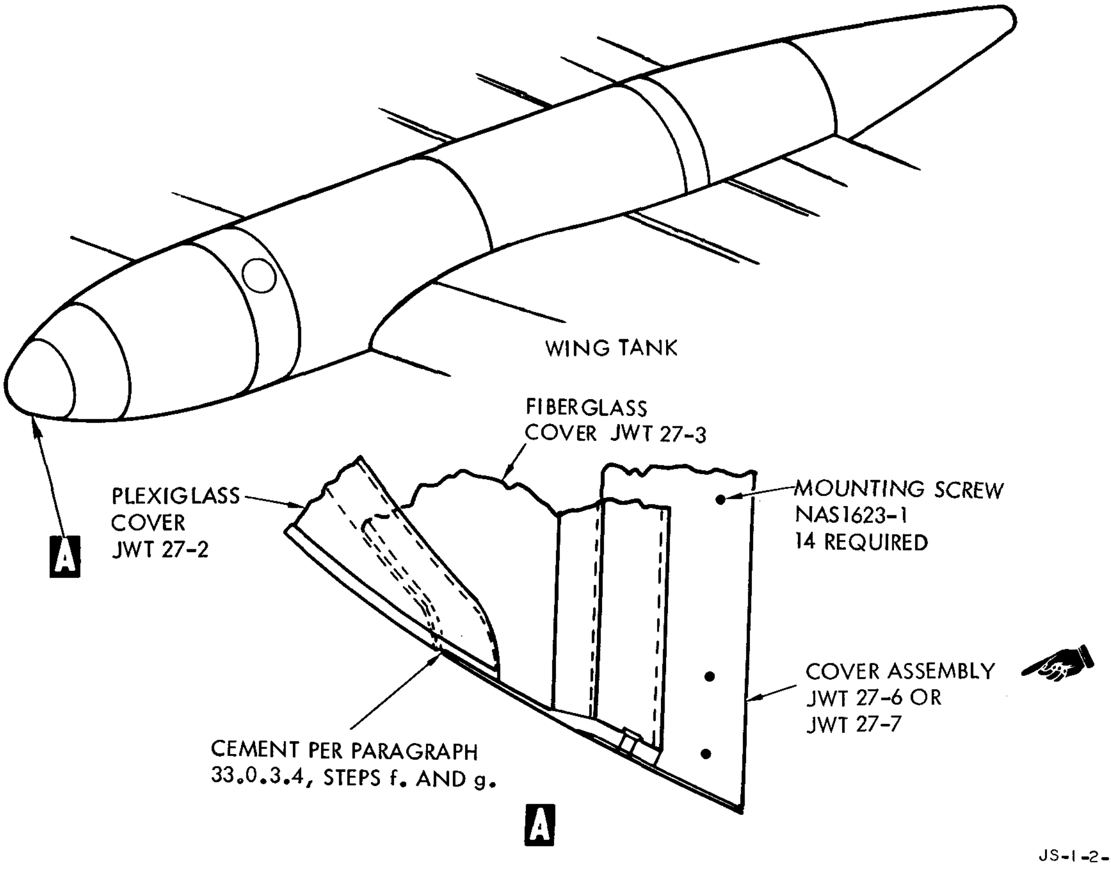


NOTE

1 THIS LIGHT INSTALLED ON AIRPLANES 5066 AND UP.

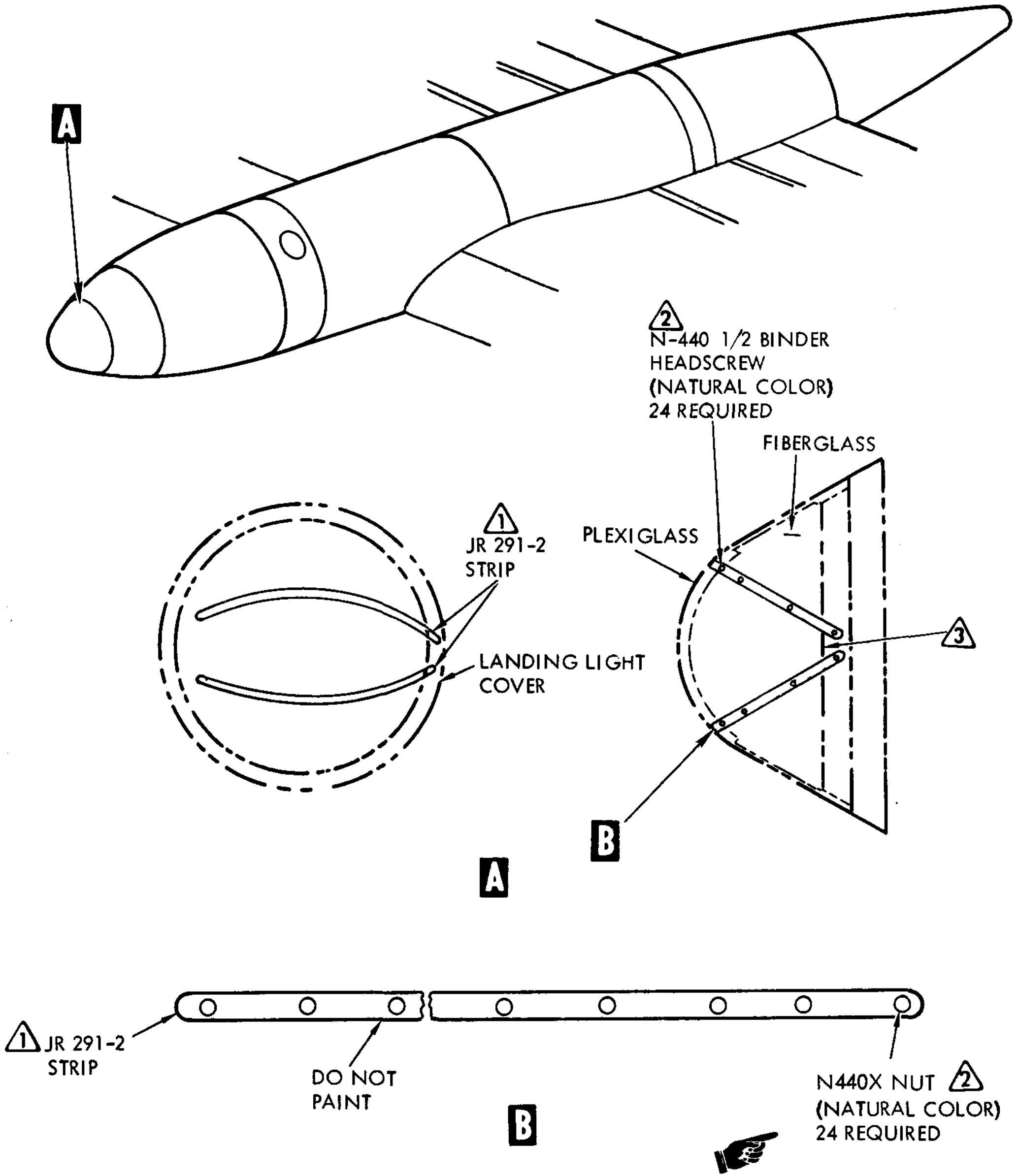
Figure 33-6. Aft Compartment Lights Schematic Diagram

JS-1-2-X0/2-3306



JS-1-2-X0/8-3307

Figure 33-7. Landing Light Cover


A
△2 N-440 1/2 BINDER HEADSCREW (NATURAL COLOR) 24 REQUIRED
FIBERGLASS
△1 JR 291-2 STRIP
PLEXIGLASS
△3
LANDING LIGHT COVER
B
A
△1 JR 291-2 STRIP
DO NOT PAINT
N440X NUT △2 (NATURAL COLOR) 24 REQUIRED
B

NOTE

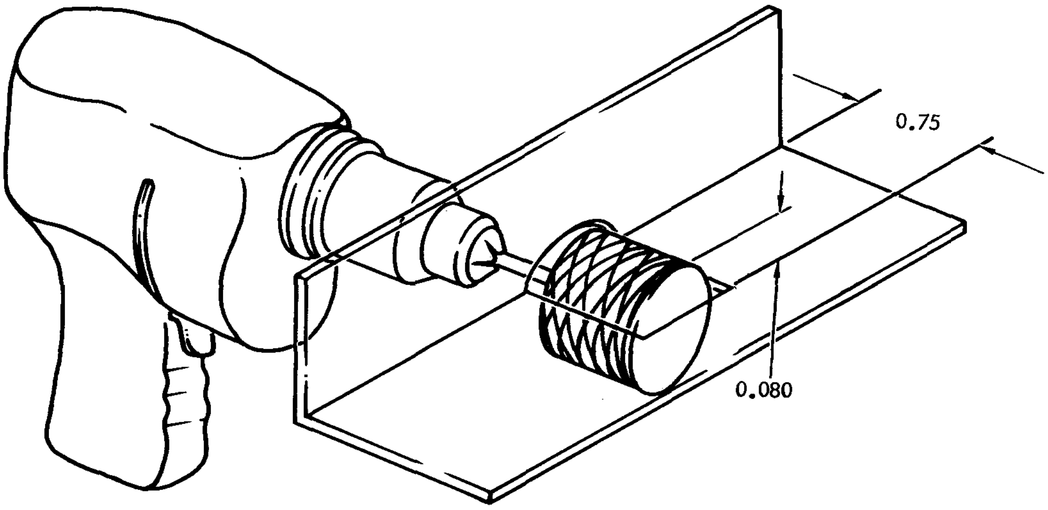
△1 CEMENT JR291-2 STRIPS TO JWT27 EXCEPT IN AREAS NOTED BY △3 .

△2 APPLY ADHESIVE TO HOLES, SCREW THREADS AND OVER NUTS.

△3 ELECTRICAL BOND FAYING SURFACE BETWEEN JR291-2 AND JWT27-5.

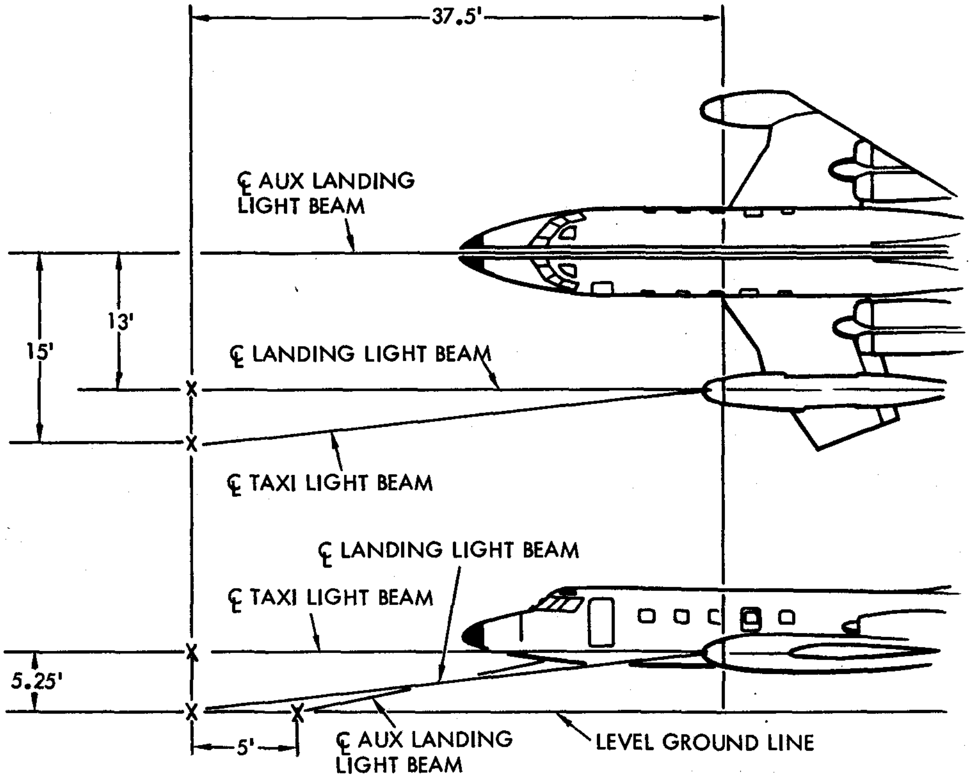
JS-1-2-X0/7-3308

Figure 33-8. Static Charge Dissipating Strip Installation



JS-1-2-X0/2-3309

Figure 33-9. Landing Light Cover Sanding Drum Fixture



NOTE

LANDING AND TAXI LIGHT RIGGING INSTRUCTION (LANDING ANGLE 7°0'0").

1. AIRPLANE JACKED TO PROVIDE 1.00" CLEARANCE ON MAIN LANDING GEAR TIRE, STRUT EXTENDED.
2. AIRPLANE TO BE LEVEL.

JS-1-2-X0/2-3310

Figure 33-10. Landing and Taxi Light Rigging

LIGHT	COLOR *	PART NUMBER
WING INSPECTION	CLEAR	GE 4594
ANTI-COLLISION	RED	A7079B24 GRIMES
NAVIGATION POSITION AFT FUSELAGE	WHITE	1683 GRIMES
LH WING TIP	RED	A7512 24 GRIMES
RH WING TIP	GREEN	A7512 24 GRIMES
LANDING	CLEAR	MS25242-4559
TAXI	CLEAR	MS25241-4553
AUXILIARY LANDING LIGHT	CLEAR	MS25241-4581

*COLOR GIVEN IS THAT OF LENS; THE BULB IS CLEAR

JS-33-0-011

Figure 33-11. Exterior Lights Replacement List

SECTION 33.1
WARNING CIRCUITS

Contents Page

DESCRIPTION AND OPERATION 33-15
General 33-15
Annunciator Warning System Sensing Devices 33-15
Aft Equipment Compartment Overheat Warning Light Circuits 33-16

OPERATIONAL CHECKOUT 33-16
Testing the Overheat Thermostats 33-16

TROUBLESHOOTING 33-16
Master Warning System Annunciator Panel Repair 33-16A

FIGURES
33-12. Annunciator Warning System Controls and Indicators (Airplanes 5001 through 5085) 33-17

33-13. Annunciator Warning System Controls and Indicators (Airplanes 5086 and Up) 33-18

33-14. Annunciator Warning System Schematic Diagram (Airplanes 5001 through 5085) 33-19

33-15. Annunciator Warning System Schematic Diagram (Airplanes 5086 and Up) 33-20

33-16. Aft Compartment Overheat Warning Component Locations (Airplanes 5001 through 5085) . . . 33-21

33-17. Aft Compartment Overheat Warning Schematic Diagram (Airplanes 5001 through 5085) 33-21

33-18. Aft Compartment Overheat Warning Component Locations (Airplanes 5086 and Up) 33-22

33-19. Aft Compartment Overheat Warning Schematic Diagram (Airplanes 5086 and Up) 33-22

33.1.1 DESCRIPTION AND OPERATION.

33.1.1.1 GENERAL. (See figures 33-12 through 33-15.) Certain system failures are immediately indicated on an annunciator panel on the main instrument panel. On airplanes 5001 thru 5085 the annunciator panel contains sixteen modules, and on airplanes 5086 and up the annunciator panel contains twenty-four modules, each indicating a particular fault when the module goes on. When one module goes on, both the pilot's and the copilot's master CAUTION lights go on. The master CAUTION lights can be turned off and reset by depressing the master CAUTION light module, but the annunciator warning module stays on until the fault is corrected or, in some cases, until the equipment concerned is turned off. If a second fault occurs when one module is on, but after the master CAUTION lights have been reset, the master CAUTION lights go on again along with the module corresponding to the second fault. Depressing either master CAUTION module energizes relays in the annunciator panel and/or the controller if a ground circuit (fault signal) is present. The relays disconnect the fault signal from the master CAUTION lights and complete a holding circuit to keep the relay energized as long as the fault is present. An ANNUNCIATOR TEST-BRIGHT-DIM switch has, in addition to the normal position, two momentary positions, "TEST-BRIGHT" and "DIM." With the switch in the "TEST-BRIGHT" position, all the annunciator warning system lights go on brightly, and on subsequent relighting of any of the modules they remain bright until changed. Momentarily placing the switch in the "DIM" position, diminishes the brightness of the lights. If the ENGINE FUEL PANEL & CENTER STAND INSTRUMENT LIGHTS control is in the "OFF" position, the lights cannot be dimmed. The annunciator warning system is supplied 28-volt, essential DC bus power from the ignition bus through a WARNING INTERIOR LIGHTS circuit breaker on the pilot's side circuit breaker panel. The warning light dimming relays and resistors are located under the copilot's floor on the nose wheel well structure. The warning light controller is located on the left-hand forward side of the nose compartment bulkhead.

Note

> On airplanes 5086 and up, the master lights are titled WARNING instead of CAUTION (see figure 33-13).

33.1.1.2 ANNUNCIATOR WARNING SYSTEM SENSING DEVICES. The sensing devices, or sensors, can be grouped into three classes according to their method of operation. The first is composed of sensors that respond to heat. This type of sensor is basically a bimetallic strip that closes a switch at a predetermined temperature. The second is composed of sensors that respond to pressure. These sensors make use of a flexible chamber that moves when pressurized. The cabin pressure sensors belong in this group. The third group consists of switches that are mechanically operated; and, in some

33.1.1.2 (Cont).

cases, a set of contacts on a relay. An example of this group is in the cabin door unlocked warning sensor. The warning light controller consists of numerous modules containing one relay and four diodes each. The warning light controller is used with only three warning systems. These are the generator out, the inverter out, and the low fuel pressure warning system. The controller is required for these three systems since each of the systems have more than one unit which can fail and cause the annunciator and master caution lights to come on. The function of the controller is to allow the master caution light to be reset when any of these three systems illuminate it. After resetting the master caution light, the controller will again illuminate the master caution light if another failure occurs in the same system.

33.1.1.3 AFT EQUIPMENT COMPARTMENT OVERHEAT WARNING LIGHT CIRCUITS.

(See figures 33-16 through 33-19.) An amber-colored AFT EQUIP COMPT OVERHEAT warning light on the pilot's main instrument panel flashes when any of the five thermostats are overheated or when a test switch is depressed. All of the thermostats are in the aft compartment; three at fuselage station 608 and two at fuselage station 672. When depressed, the test switch on the aft compartment electrical equipment rack simulates an overheat condition by applying a ground to the warning circuit. An overheat flasher unit is on the right-hand side of the aft compartment at fuselage station 601. Essential DC bus power at 28 volts is supplied to the system through an AFT COMPTO/HEAT LT circuit breaker on the pilot's upper circuit breaker panel. On airplanes 5086 and up the AFT EQUIP COMPT OVERHEAT warning light is in a module of the annunciator panel and does not flash. The same thermostats are used as the sensors.

33.1.2 OPERATIONAL CHECKOUT.

a. Connect external power to the airplane and engage the WARNING lights circuit breaker on the pilot's side circuit breaker panel.

b. Place the ANNUNCIATOR TEST switch in the TEST-BRIGHT position.
RESULT: All of the annunciator panel and master caution lights should illuminate.

c. Individual warning circuits are checked out during the operational checkouts of the particular systems.

33.1.3 TESTING THE OVERHEAT THERMOSTATS.

The five thermostats (P/N 25720078-01) located in the aft equipment area (see figure 33-16 or 33-18), may be tested for functional accuracy with the aid of the following equipment:

Test Equipment Required

a. Ohmmeter

b. Hypot Tester

c. Calibrated Oil Bath

d. Precision Thermometer (180° to 220°F in 0.1° increments)

Tests

a. With 600V AC (RMS), Hypot each pin (A, B and C) of the MS-3102E-14S-7PX plug wired to each thermostat.

b. Calibration of oil bath. Bring the temperature of the oil bath up to 190° ±2°F and stabilize.

c. Immerse the probe of the thermostat up to the chamfer just below the mounting bracket - allow to soak for five minutes.

d. Place an Ohmmeter across pins "A" and "B".

e. Continuity should not be indicated.

f. Increase the temperature of the oil bath at a rate not to exceed 0.5°F/minute.

g. Record the temperature at which continuity is indicated between pins "A" and "B." This should be 215°F maximum.

33.1.4 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MASTER CAUTION LIGHT FAILS TO GO OUT WHEN RESET.		
Open diode or defective relay in annunciator panel module or controller module.	Check continuity through diode and operation of relay.	Replace annunciator panel or controller.

33.1.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MASTER CAUTION LIGHT FAILS TO ILLUMINATE WHEN FAULT SIGNAL IS RECEIVED IN THE ANNUNCIATOR PANEL.		
Open diode in the annunciator panel module.	Check continuity through diode.	Replace annunciator panel.
ANNUNCIATOR PANEL MODULE FAILS TO ILLUMINATE WHEN TEST SWITCH IS ACTUATED.		
Open diode in the annunciator panel module.	Check continuity through diode.	Replace annunciator panel.
WARNING LIGHTS FAIL TO DIM WHEN DIM SWITCH IS ACTUATED.		
Overhead instrument panel light switch in OFF position.	Check position of switch.	Turn to "ON" position.
Defective dimming relay or resistor.	Check resistors and operation of relays.	Replace resistors or relays.

33.1.5 MASTER WARNING SYSTEM ANNUNCIATOR PANEL REPAIR. Remove four screws and annunciator panel from center instrument panel. (See figures 33-12 and 33-13.) Modules may be repaired as follows:

a. Disassemble the panel.

(1) Remove four bolts clamping modules together.

(2) Remove two screws securing individual module(s) to back plate.

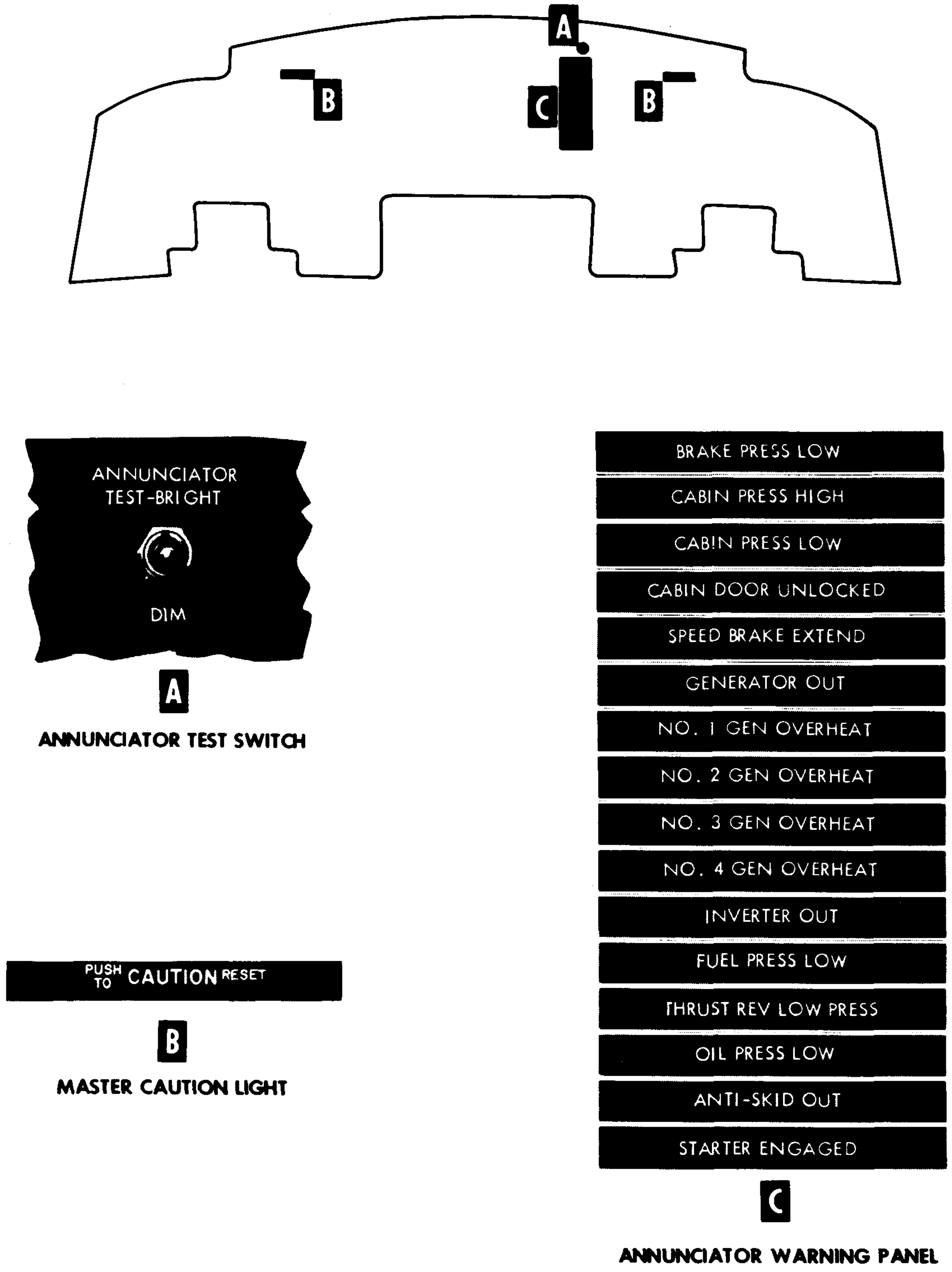
(3) Affected modules can' be separated from other modules for access and parts replacement.

(4) Replace defective part(s). See figure 33-14 or 33-15.

b. Reassemble the panel.

c. Parts list.

Relay - G.E. P/N 3S2791G200C9.
Diodes - IN5061 1A or IN4385 1A
Lamps - #327

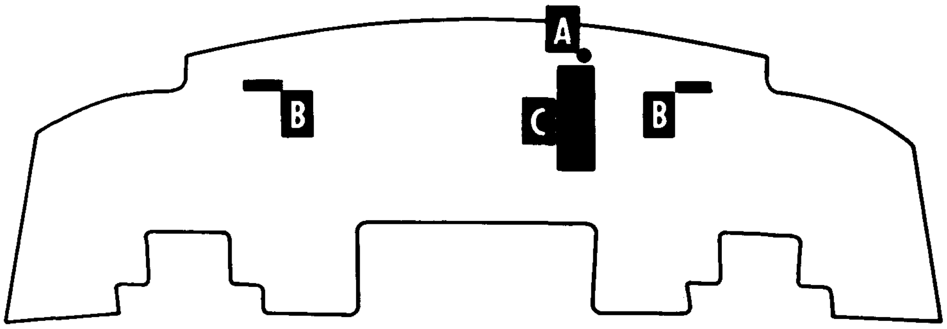


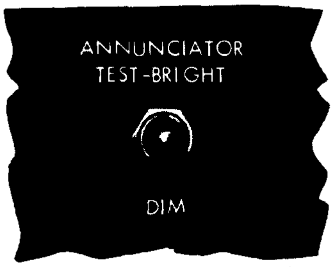
ANNUNCIATOR TEST SWITCH

MASTER CAUTION LIGHT

ANNUNCIATOR WARNING PANEL

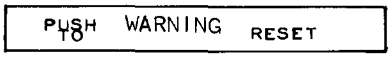
Figure 33-12. Annunciator Warning System Controls and Indicators (Airplanes 5001 thru 5085)





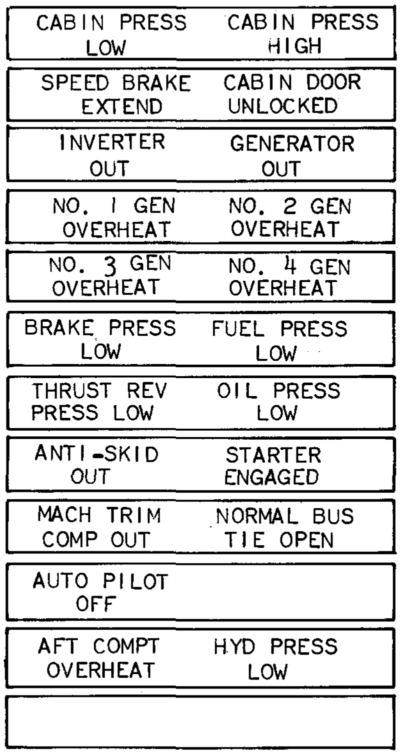
A

ANNUNCIATOR TEST SWITCH



B

MASTER WARNING LIGHT

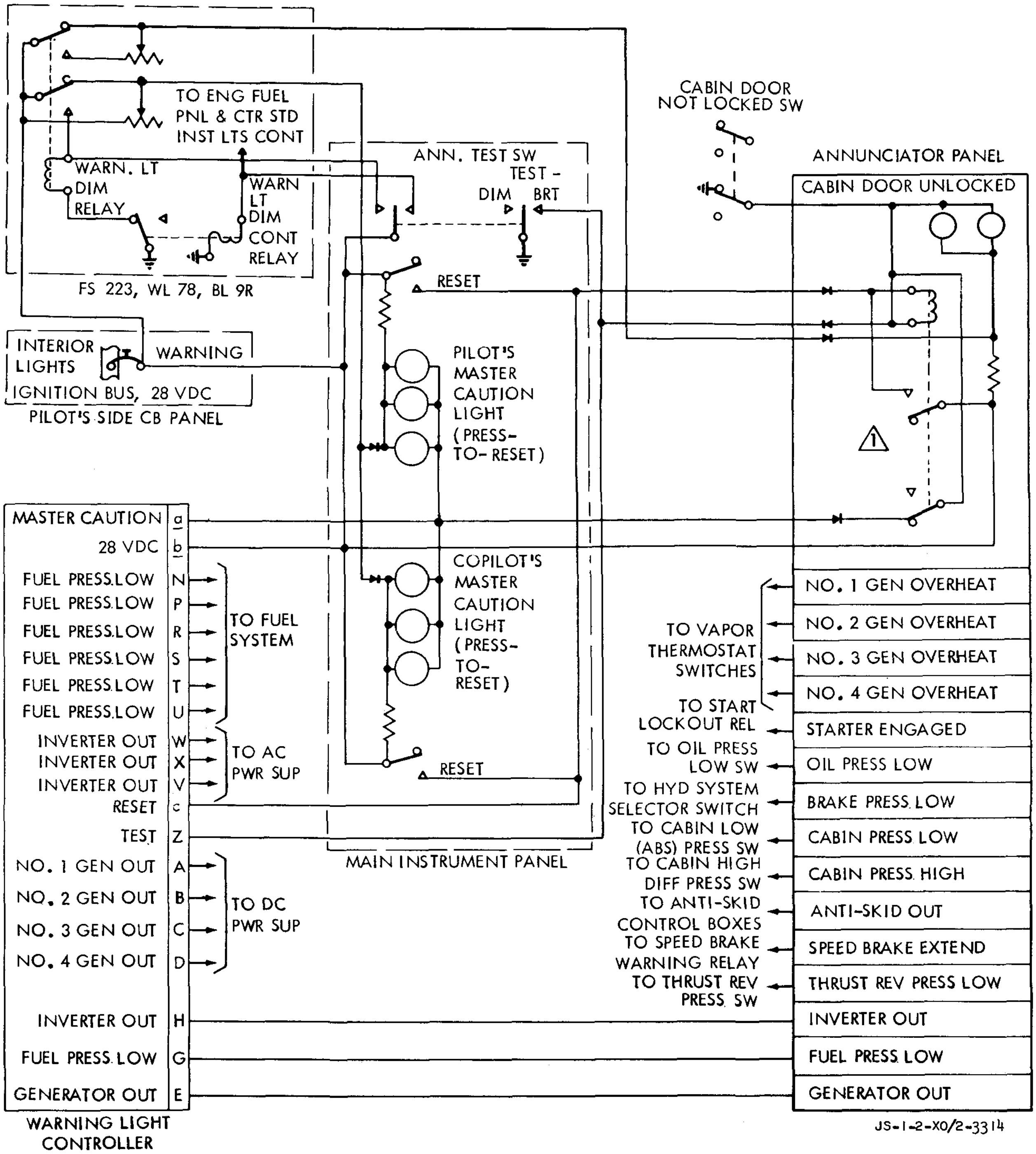


CABIN PRESS LOW	CABIN PRESS HIGH
SPEED BRAKE EXTEND	CABIN DOOR UNLOCKED
INVERTER OUT	GENERATOR OUT
NO. 1 GEN OVERHEAT	NO. 2 GEN OVERHEAT
NO. 3 GEN OVERHEAT	NO. 4 GEN OVERHEAT
BRAKE PRESS LOW	FUEL PRESS LOW
THRUST REV PRESS LOW	OIL PRESS LOW
ANTI-SKID OUT	STARTER ENGAGED
MACH TRIM COMP OUT	NORMAL BUS TIE OPEN
AUTO PILOT OFF	
AFT COMPT OVERHEAT	HYD PRESS LOW

C

ANNUNCIATOR WARNING PANEL

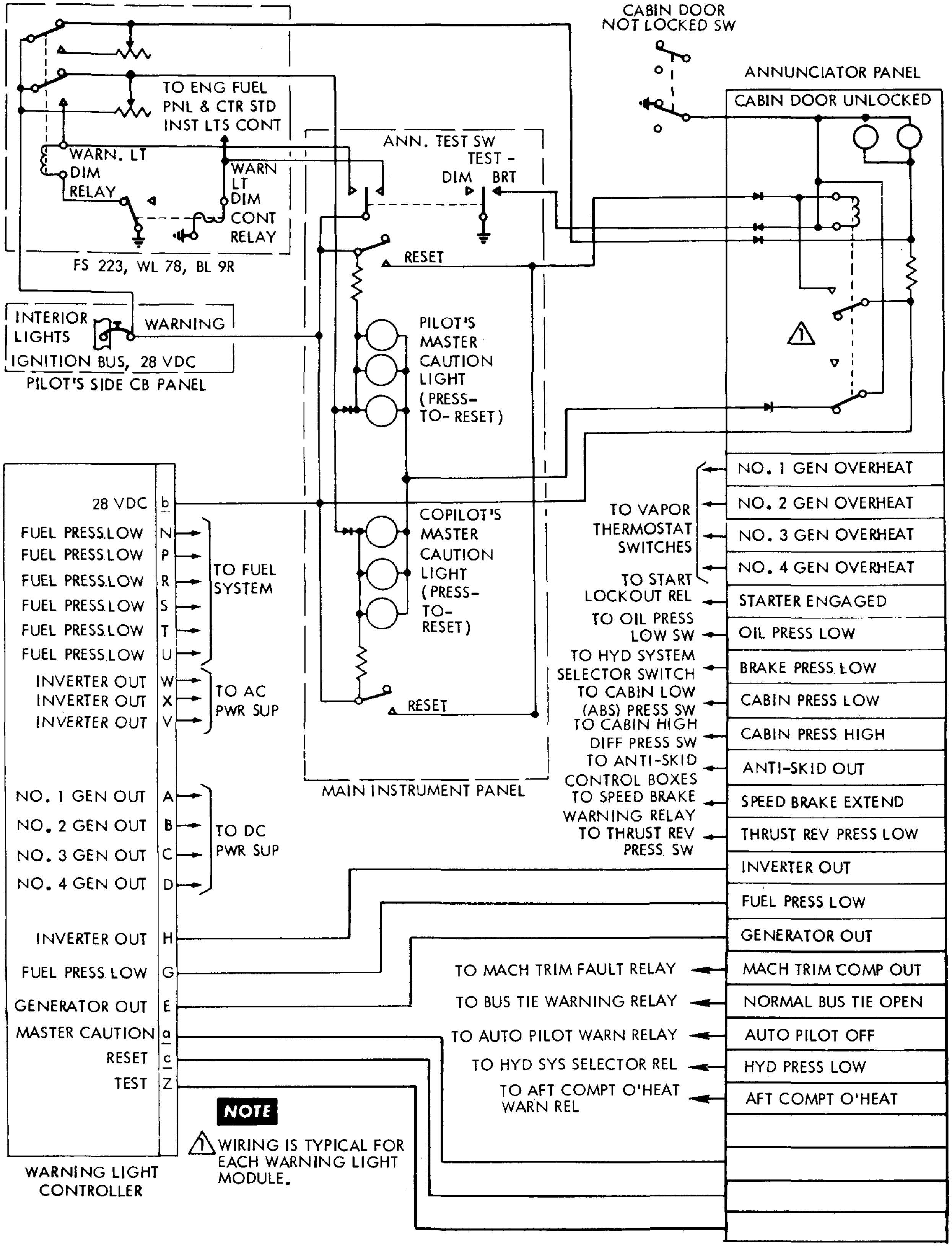
Figure 33-13. Annunciator Warning System Controls and Indicators (Airplanes 5086 and Up)



NOTE

1 WIRING IS TYPICAL FOR EACH WARNING LIGHT MODULE.

Figure 33-14. Annunciator Warning System Schematic Diagram (Airplanes 5001 thru 5085)

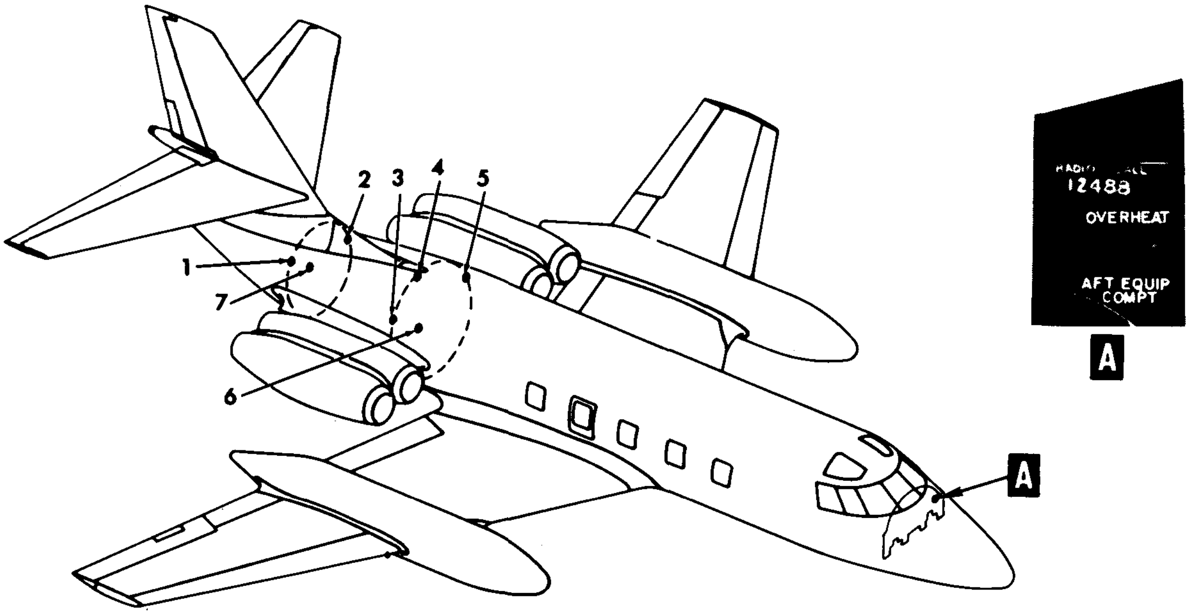


NOTE

1 WIRING IS TYPICAL FOR EACH WARNING LIGHT MODULE.

JS-1-2-X0/2-3315

Figure 33-15. Annunciator Warning System Schematic Diagram (Airplanes 5086 and Up)



1. OVERHEAT THERMOSTAT NO. 5 (FS 680 RH)
2. OVERHEAT THERMOSTAT NO. 4 (FS 680 LH)
3. OVERHEAT THERMOSTAT NO. 1 (FS 608 RH)
4. OVERHEAT THERMOSTAT NO. 2 (FS 608 CTR)
5. OVERHEAT THERMOSTAT NO. 3 (FS 608 LH)
6. AFT COMPT OVERHEAT FLASHER UNIT (FS 601.5 RH)
7. AFT COMPT OVERHEAT TEST SWITCH (FS 670 RH)

Figure 33-16. Aft Compartment Overheat Warning Components Locations (Airplanes 5001 thru 5085)

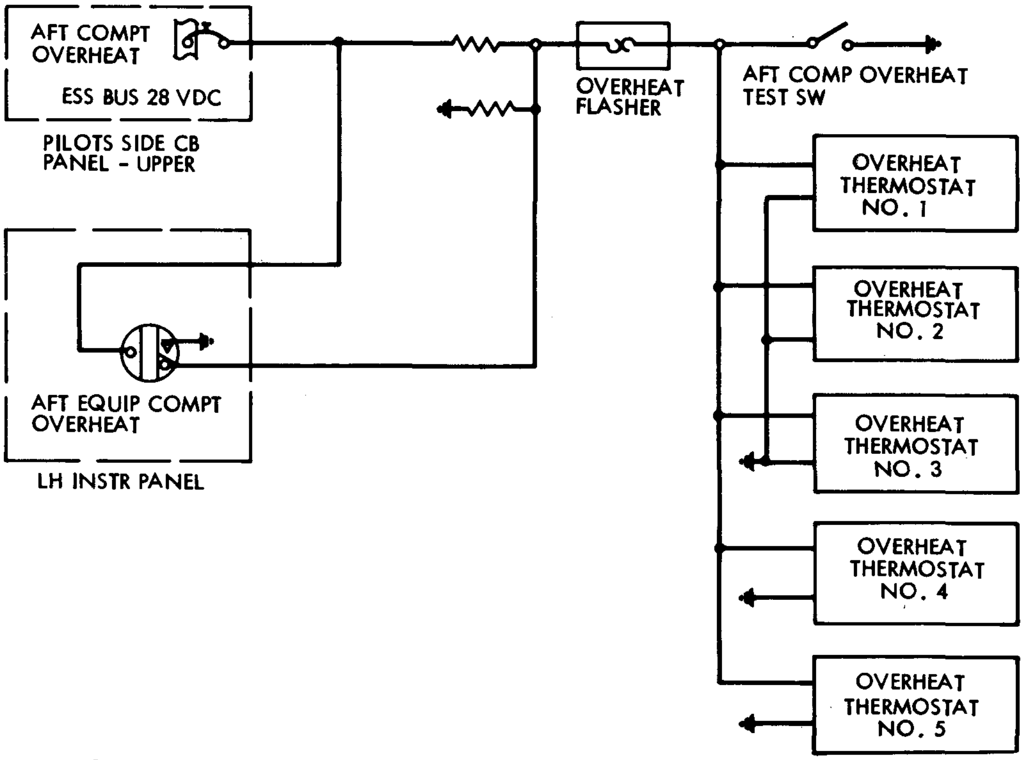
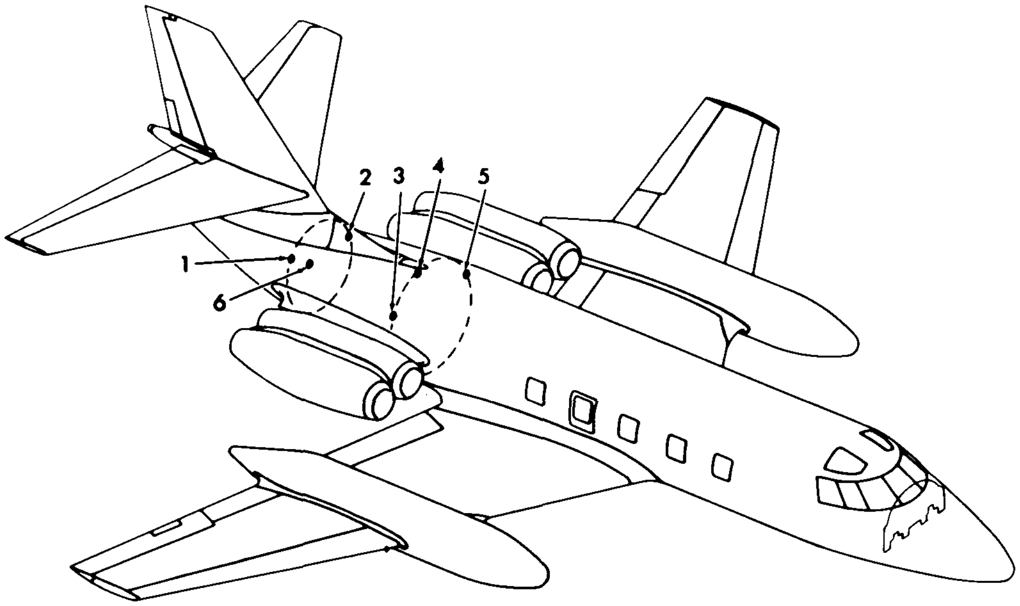

AFT COMPT OVERHEAT
ESS BUS 28 VDC
PILOTS SIDE CB PANEL - UPPER
OVERHEAT FLASHER
AFT COMP OVERHEAT TEST SW
OVERHEAT THERMOSTAT NO. 1
OVERHEAT THERMOSTAT NO. 2
OVERHEAT THERMOSTAT NO. 3
OVERHEAT THERMOSTAT NO. 4
OVERHEAT THERMOSTAT NO. 5
AFT EQUIP COMPT OVERHEAT
LH INSTR PANEL

Figure 33-17. Aft Compartment Overheat Warning Schematic Diagram (Airplanes 5001 thru 5085)



1. OVERHEAT THERMOSTAT NO. 5 (FS 680 RH)
2. OVERHEAT THERMOSTAT NO. 4 (FS 680 LH)
3. OVERHEAT THERMOSTAT NO. 1 (FS 608 RH)
4. OVERHEAT THERMOSTAT NO. 2 (FS 608 CTR)
5. OVERHEAT THERMOSTAT NO. 3 (FS 608 LH)
6. AFT COMPT OVERHEAT TEST SWITCH (FS 670 RH)

Figure 33-18. Aft Compartment Overheat Warning Component Locations (Airplanes 5086 and Up)

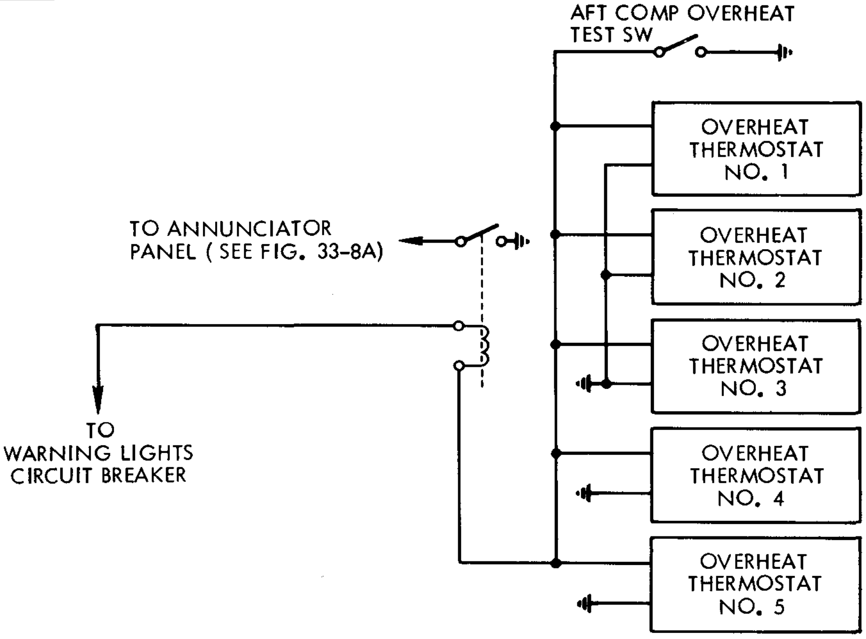

AFT COMP OVERHEAT TEST SW
OVERHEAT THERMOSTAT NO. 1
OVERHEAT THERMOSTAT NO. 2
OVERHEAT THERMOSTAT NO. 3
OVERHEAT THERMOSTAT NO. 4
OVERHEAT THERMOSTAT NO. 5
TO ANNUNCIATOR PANEL (SEE FIG. 33-8A)
TO WARNING LIGHTS CIRCUIT BREAKER

Figure 33-19. Aft Compartment Overheat Warning Schematic Diagram (Airplanes 5086 and Up)

CHAPTER 34
NAVIGATION SYSTEMS

TABLE OF CONTENTS

SECTION PAGE

34.0 GENERAL 34-3
34.1 VHF NAVIGATION SYSTEMS (VOR) 34-11
34.2 FLIGHT DIRECTOR SYSTEMS 34-19
34.3 GLIDESLOPE SYSTEMS 34-33
34.4 MARKER BEACON SYSTEM 34-39
34.5 AUTOMATIC DIRECTION FINDER 34-45
34.6 GYROSYN COMPASS SYSTEMS 34-53
34.7 WEATHER RADAR 34-61
34.8 DISTANCE MEASURING EQUIPMENT SYSTEM 34-71
34.9 AIR TRAFFIC CONTROL TRANSPONDER SYSTEM 34-77
34.10 AIR DATA, ATTITUDE, AND DIRECTION INSTRUMENTATION 34-83
34.11 ALTERNATE ELECTRONIC SYSTEMS 34-98A
34.12 DOPPLER RADAR NAVIGATION AND COMPUTER SYSTEM 34-101

SECTION 34.0
GENERAL

Contents

Contents Page

DESCRIPTION 34-3
  List of Systems 34-3
  Functions of Systems 34-3

INSTALLATION 34-4
  Black Box Components 34-4
  Control Panels and Instruments 34-4
  Equipment Rack 34-4
  Design Weight and CG for Equipment 34-4
  Special Mountings 34-4
  Cooling Provisions 34-4
  Antennas 34-5
  Radome 34-5

MAINTENANCE 34-5
  Preservation of Electronic Equipment 34-5
  Depreservation of Electronic Equipment 34-5

FIGURES
  34-1. Indicator and Control Panel Locations 34-7
  34-2. Nose Electronic Equipment Rack (Airplanes 5001 thru 5049 and 5052, 5056 and 5059) 34-8
  34-2A. Nose Electronic Equipment Rack (Airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up) 34-9
  34-3. External Antenna Locations 34-10

34.0.1 DESCRIPTION.

34.0.1.1 LIST OF SYSTEMS. The JetStar contains electronic navigational systems for all-weather operation. These are:

a. Two VOR-localizer systems
b. Two glideslope receivers
c. Two flight director systems
d. One marker beacon receiver
e. One distance measuring system
f. Two automatic direction finder systems
g. Two gyrosyn compass systems
h. One X-band weather radar system
i. One air traffic control transponder.

Note

> On aircraft 5046 and up, black box components of navigational systems will not be installed when the aircraft is manufactured. However, complete provisions for these boxes are installed by Lockheed including all wiring, mounting provisions, relays, circuit breakers, etc. All antennas except weather radar and ADF loops are also installed.

34.0.1.2 FUNCTION OF SYSTEMS. The principal presentation of navigational information is displayed on the two sets of flight director instruments. Using these instruments, omnirange radial courses and magnetic headings may be flown enroute, and a complete ILS approach may be made utilizing localizer and glideslope inputs. Dual installations of the flight directors, VOR-localizers, glideslopes, and gyrosyn compasses are made to provide independent pilot and copilot navigational facilities. In order to further insure the independence of these dual systems, the pilot's systems (number one) are powered by the essential AC and DC busses. The copilot's systems (number two) are powered by the main busses. The three-light marker beacon receiver includes aural outputs and the automatic direction finder system is provided with remote tuning. A DME system, tuned by the pilot's VOR frequency selector, is included to indicate distance to VOR-TAC or TACAN ground stations. X-band weather radar is provided; features of this system include a daylight indicator visible to both pilots and a fully stabilized 18-inch antenna dish. In addition, an air traffic control transponder is provided to assist ground identification of the aircraft.

34.0.2 INSTALLATION INFORMATION.

34.0.2.1 BLACK BOX COMPONENTS. Practically all electronic boxes are located inside the nose section just aft of the nose radome. Access to this compartment is gained by releasing the latch bolts that secure the nose shell to the aircraft structure and sliding the nose shell forward until stops are reached. This provides complete access to all components except the weather radar and glideslope antennas.

Note

To reach weather radar and glideslope antennas, remove the stops and lift the nose shell off of the aircraft.

34.0.2.2 CONTROL PANELS AND INSTRUMENTS. (See figure 34-1). All control panels for navigation instruments are installed in the center console between the pilot and copilot positions. Instruments which present navigational information are located on the main instrument panel. All navigation instruments on the JetStar are dual (pilot and copilot) instruments except the weather radar scope. This scope is the daylight tube type which can be seen by both pilot and copilot without the use of a hood.

34.0.2.3 EQUIPMENT RACK. (See figure 34-2). The main electronic equipment rack is in the nose section. Positions provided for each component are labeled and provided with index pins to insure that no component is plugged in at the wrong position. Once in place, the box is secured to the rack with mounting studs and thumb nuts. Each of the three shelves is shock-mounted with Lord Company Broad Temperature Range shock-mounts. These mounts are selected to effectively isolate the equipment from shock and vibration loads, however, if the total weight of equipment on any shelf is changed significantly (±25%), shock-mounts with different weight ratings should be installed. Periodic visual checks should be made to insure that nuts, bolts, or other foreign bodies are not wedged beneath any shelf in such a way as to interfere with shelf movement.

34.0.2.3.1 DESIGN WEIGHT AND CG FOR EQUIPMENT.

a. Loads used for design of electronic equipment support beam JF369:
   (1) 21 lbs at forward end, FS110 (radar unit)
   (2) 22 lbs at FS141
   (3) 267 lbs at FS144 (electronic equipment and support structure)
   (4) 69 lbs at FS148 (electronic equipment, etc. on lower shelf
   CG is 379 lbs at FS 142.67

b. Design load for upper shelf is 104 lbs at FS152.

Note

(1) 1.0 bending moment at FS 164 is 8084 inch-pounds

(2) Vertical load factor is ULT 9.0G DOWN, 6.0G UP

(3) Fore and aft load factor is ULT ± 3.0G

(4) Side load factor is ULT 0.70G

Note

The lightweight honeycomb panel between the two vertical beams is designed for stabilizing the beams and is not for supporting equipment items.

34.0.2.4 SPECIAL MOUNTINGS. Several items in the nose require special mounting methods and procedures. See the following sections of this HOMI for these instructions:

Vertical Gyro	-34.2
Autopilot Accelerometer	-22.0

34.0.2.5 COOLING PROVISIONS. Provisions are included in the nose electronic equipment rack for cooling the equipment during flight with cabin exhaust air. The entire nose section is fully pressurized to cabin pressure during flight through an air flow limiter in the bulkhead at fuselage station 164, that will close in case of depressurization of the nose compartment. During normal operation, cooling air passes through the nose section into plenums beneath the boxes, and through an air flow regulator forward of the nose wheel well. Once through the regulator, the air is dumped overboard through the nose wheel well.

a. Ground cooling of electronic equipment is accomplished by a cooling fan that forces air through the plenums beneath the equipment. This fan is part of the flow regulator assembly. On airplanes 5001 thru 5065, the fan is on only when the differential pressure between cabin and outside air is less than 1/2 PSI. Thus, the cooling fan provides cooling air during ground operation and immediately after takeoff until the cabin is pressurized 1/2 PSI above ambient. Then, a pressure switch shuts off the fan and air flow is maintained by cabin pressure. On airplanes 5066 and up the cooling fan is controlled through a switch actuated by weight on the landing gear; thus, the cooling fan provides cooling air only during the time the airplane is on the ground.

b. (Airplanes 5001 thru 5049, 5052, 5056 and 5059). Cooling air (either cabin exhaust or ground cooling fan) passes through the shelf mounted black boxes

34.0.2.5 (Cont).

and enters the plenum through holes located beneath the black boxes (a soft gasket seals between the box and plenum when the box is placed in the rack). These holes are left open or plugged to proportion the air flow through each box. The airplane is delivered with the correct number of holes open under each box. If electronic equipment is added or changed after delivery, it may be necessary to open additional holes if the new box requires additional heat dissipation. Such changes in the number of open holes beneath any box should be made in accordance with the following instructions.

STEP 1. Divide power requirements in watts by 50; result in the total number of open holes required.

STEP 2. No more than 50 nor less than 20 open holes should exist on any of the above airplanes to insure proper air flow.

Note

It should be understood that these instructions are very general and installation of any unusual electronic equipment (high pressure drop of air through box, high heat dissipation in small package etc.) will require a special analysis.

Note

Flight may be made with boxes temporarily removed without altering plug buttons since total air flow regulation is accomplished by the flow regulator in the nose wheel area.

34.0.2.5 (Cont).

c. (Airplanes 5001 thru 5049, 5052, 5056 and 5059). Equipment in the nose area not located on shock-mounted shelves is cooled by convection and circulating air. Some units also contain small integral blowers to assist in internal circulation.

d. (Airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up). Cooling air (either cabin exhaust or ground cooling fan) is forced through the nose section cooling the shelf mounted and non-shelf mounted equipment. The air circulates around the equipment and enters the plenum through fixed holes located in each of the three shelves. Some units also contain small integral blowers to assist in internal circulation.

e. Electronic equipment may be turned on for short periods on the ground without the cooling fan, but operation of several pieces of equipment (especially transmitters) on a hot day or in sunlight without ventilation, with no cooling air flow will shorten the life of the equipment. Generally, JetStar equipment is designed to operate continuously at a maximum of 131°F and for 1/2 hour at a maximum of 180°F.

34.0.2.6 ANTENNAS. (See figure 34-3). All external antennas except the VHF communication, VOR, and marker beacon are flush-mounted. Two antennas, weather radar and glideslope, are located behind the nose radome. All antennas are removable from the exterior of the aircraft except the marker beacon antenna, it is removed from inside the aircraft. For more detailed information of each antenna, refer to the section of this manual describing the applicable electronic system.

34.0.2.7 RADOME. The JetStar nose radome is of honeycomb sandwich construction, designed to allow efficient transmission of X-band frequencies. It is permanently attached to the nose shell by a row of 3/16 inch rivets. If the shell-radome is severely damaged and cannot be repaired, it must be replaced as a unit. The radome is protected against wear due to impingement of rain or other particles by a neoprene rain-erosion coating which may be replaced when necessary. An anti-static coating is also applied to the radome; this prevents the accumulation of an electrical static charge on the radome which could result in severe radio noise. In addition to these surface treatments, on airplanes 5001 thru 5045 four metal foil strips (1/4 inch wide X .013 inch thick), and on airplanes 5046 and up four beryllium copper strips (1/4 inch wide X .0066 inch thick) are embedded beneath the neoprene coating in contact with the actual radome surface. These strips protect the JetStar radome against puncture damage by lightning strikes by bleeding off the large electrical currents into the aircraft framework. The strips actually vaporize during this procedure and must be replaced after a strike has occrred on the nose.

Note

Broken metal foil or beryllium copper strips can cause severe radio noise problems. If such problems exist, the strips should be checked for cracks or breaks.

Note

For airplanes 5001 thru 5045, a kit (LSK 191) is available to install the beryllium copper strips and bonding clips.

34.0.3 MAINTENANCE.

34.0.3.1 PRESERVATION OF ELECTRONIC EQUIPMENT. When the airplane is to be inactive for a prolonged period, take the following precautions to protect the electronic equipment from deterioration and to keep it clean.

a. Cover the electronic equipment with grease-proof paper (Sherman Paper Prod. Cor., "V-29").

b. Enclose all disconnected electrical fittings and connectors in polyethylene bags and tie securely.

Note

Put a small hole in the bottom of the bag to allow the moisture condensate to escape.

c. Cap all disconnected receptacles.

d. If possible, operate all of the electronic equipment for 30 minutes (continuously) once a week to dry out the accumulated moisture.

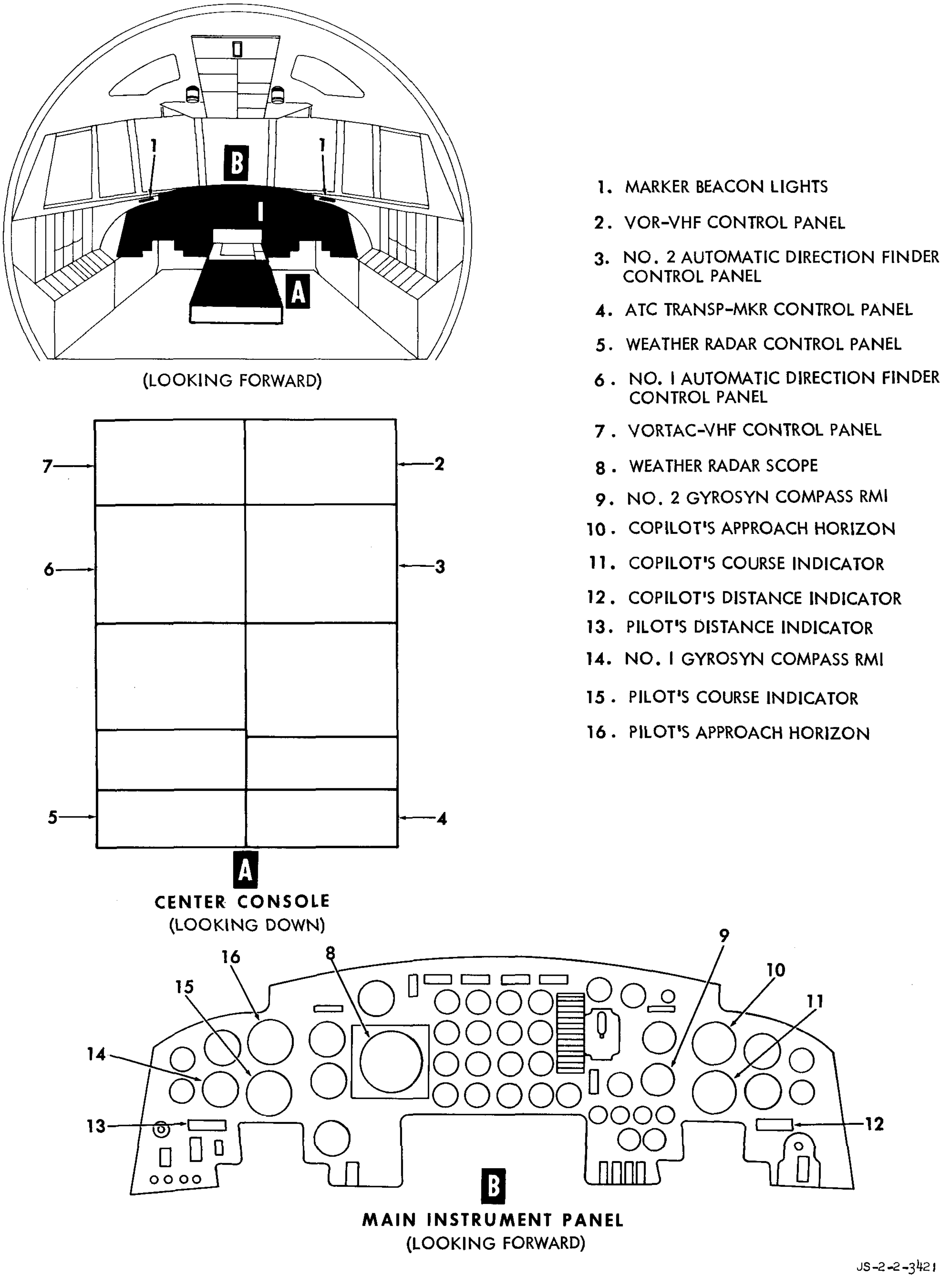
e. If such operation is not possible, isolate the equipment and keep it dry by means of a desicant. Replace the desiccant as necessary.

f. Inspect the equipment once each week. If corrosion is found, remove the corrosion and operate the equipment to thoroughly dry it.

Note

If corrosion is severe, it may be desirable to remove the equipment from the airplane and package it for storage.

34.0.3.2 DEPRESERVATION OF ELECTRONIC EQUIPMENT. Before returning the electronic equipment to service, remove all protective coverings, remove all corrosion, replace desiccants as necessary and reconnect all loose connections.



(LOOKING FORWARD)

CENTER CONSOLE
(LOOKING DOWN)

MAIN INSTRUMENT PANEL
(LOOKING FORWARD)

1. MARKER BEACON LIGHTS
2. VOR-VHF CONTROL PANEL
3. NO. 2 AUTOMATIC DIRECTION FINDER CONTROL PANEL
4. ATC TRANSP-MKR CONTROL PANEL
5. WEATHER RADAR CONTROL PANEL
6. NO. I AUTOMATIC DIRECTION FINDER CONTROL PANEL
7. VORTAC-VHF CONTROL PANEL
8. WEATHER RADAR SCOPE
9. NO. 2 GYROSYN COMPASS RMI
10. COPILOT'S APPROACH HORIZON
11. COPILOT'S COURSE INDICATOR
12. COPILOT'S DISTANCE INDICATOR
13. PILOT'S DISTANCE INDICATOR
14. NO. I GYROSYN COMPASS RMI
15. PILOT'S COURSE INDICATOR
16. PILOT'S APPROACH HORIZON

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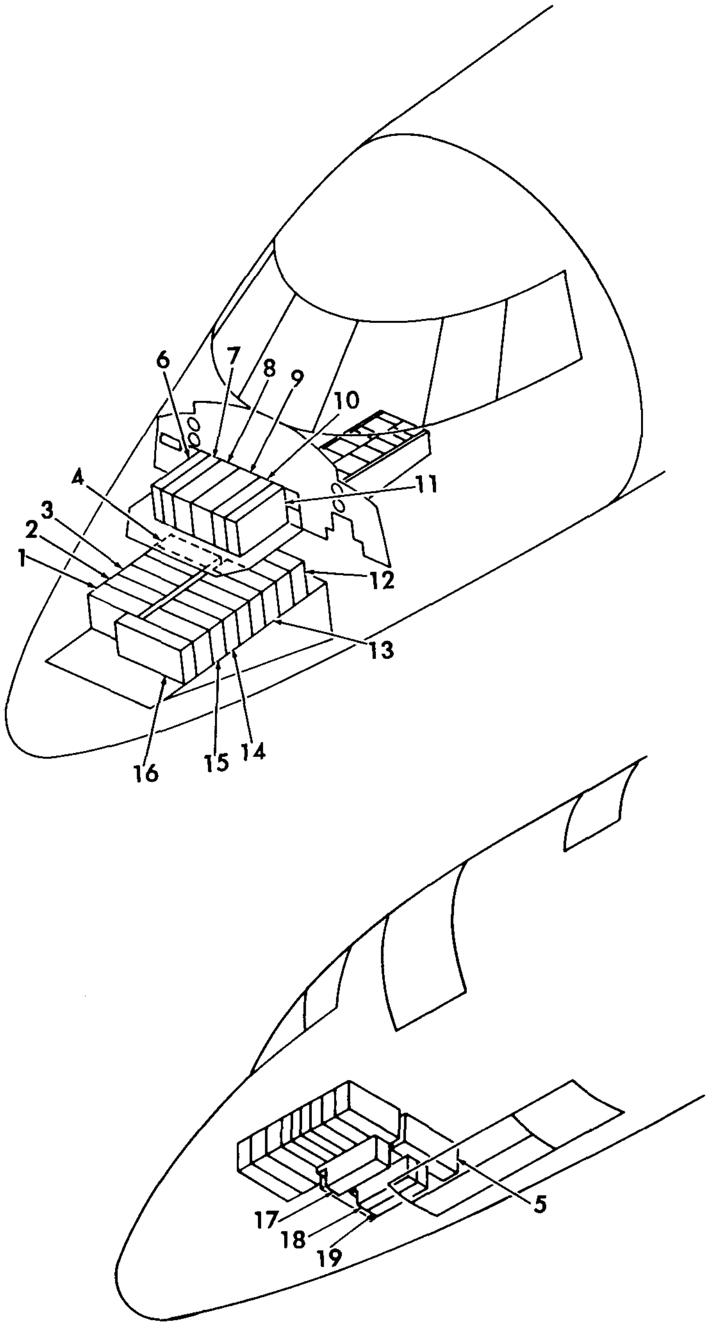
Figure 34-1. Indicator and Control Panel Locations



1. MARKER BEACON RECEIVER
2. NO. 2 SYSTEM COMPASS RACK △1
3. NO. 2 GLIDESLOPE RECEIVER
4. NO. 2 VOR RECEIVER
5. WEATHER RADAR SYNCHRONIZER-AMPLIFIER-POWER SUPPLY
6. NO. 2 FLIGHT DIRECTOR SYSTEM INSTRUMENT AMPLIFIER
7. NO. 2 FLIGHT DIRECTOR SYSTEM STEERING COMPUTER.
8. NO. 2 VOR INSTRUMENTATION UNIT
9. NO. 1 VOR INSTRUMENTATION UNIT
10. NO. 1 FLIGHT DIRECTOR SYSTEM STEERING COMPUTER
11. NO. 1 FLIGHT DIRECTOR SYSTEM INSTRUMENT AMPLIFIER
12. DME RADIO SET
13. NO. 1 VOR RECEIVER
14. NO. 1 GLIDESLOPE RECEIVER
15. NO. 1 SYSTEM COMPASS RACK △1
    NO. 1 & 2 SYSTEM COMPASS RACK
16. WEATHER RADAR TRANSMITTER-RECEIVER
17. ATC TRANSPONDER
18. NO. 2 ADF RECEIVER
19. NO. 1 ADF RECEIVER

△1 AIRPLANES 5001 THRU 5010, 5026, 5028, 5030, AND 5032 ONLY

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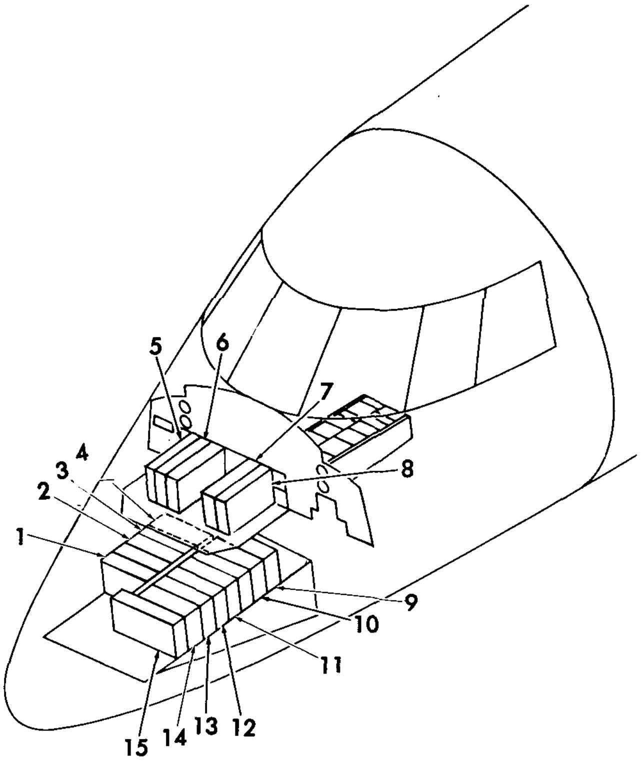
Figure 34-2. Nose Electronic Equipment Rack (Airplanes 5001 thru 5049 and 5052, 5056 and 5059)



1. MARKER BEACON RECEIVER.
2. NO. 2 NAV RECEIVER
3. ATC TRANSPONDER
4. DME RADIO SET
5. NO. 2 FLIGHT DIR INST UNIT
6. NO. 2 FLIGHT DIR STEERING COMPUTER
7. NO. 1 FLIGHT DIR STEERING COMPUTER
8. NO. 1 FLIGHT DIR INST UNIT
9. NO. 2 ADF RECEIVER
10. NO. 1 ADF RECEIVER
11. NO. 1 NAV RECEIVER
12. NO. 1 UHF TRANSCEIVER
13. NO. 1 AND NO. 2 COMPASS RACK
14. RADAR SYNCHRONIZER
15. RADAR REC-TRANSMITTER

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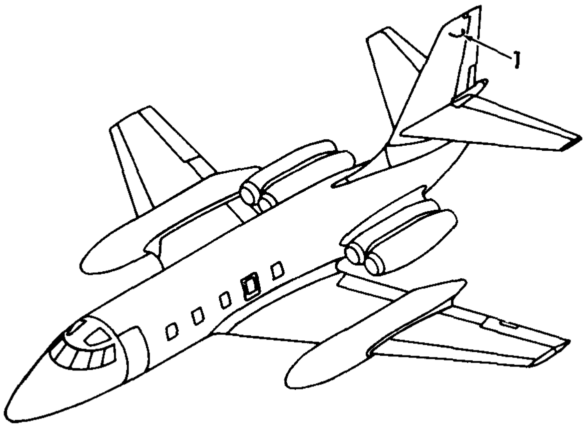
Figure 34-2A. Nose Electronic Equipment Rack (Airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and Up)



1. VOR-LOCALIZER ANTENNA
2. MARKER BEACON ANTENNA
3. NO. 1 ADF SENSE ANTENNA
4. NO. 2 ADF SENSE ANTENNA
5. NO. 2 ADF LOOP ANTENNA
6. NO. 1 ADF LOOP ANTENNA
7. AIR TRAFFIC CONTROL ANTENNA
8. DISTANCE MEASURING EQUIPMENT ANTENNA

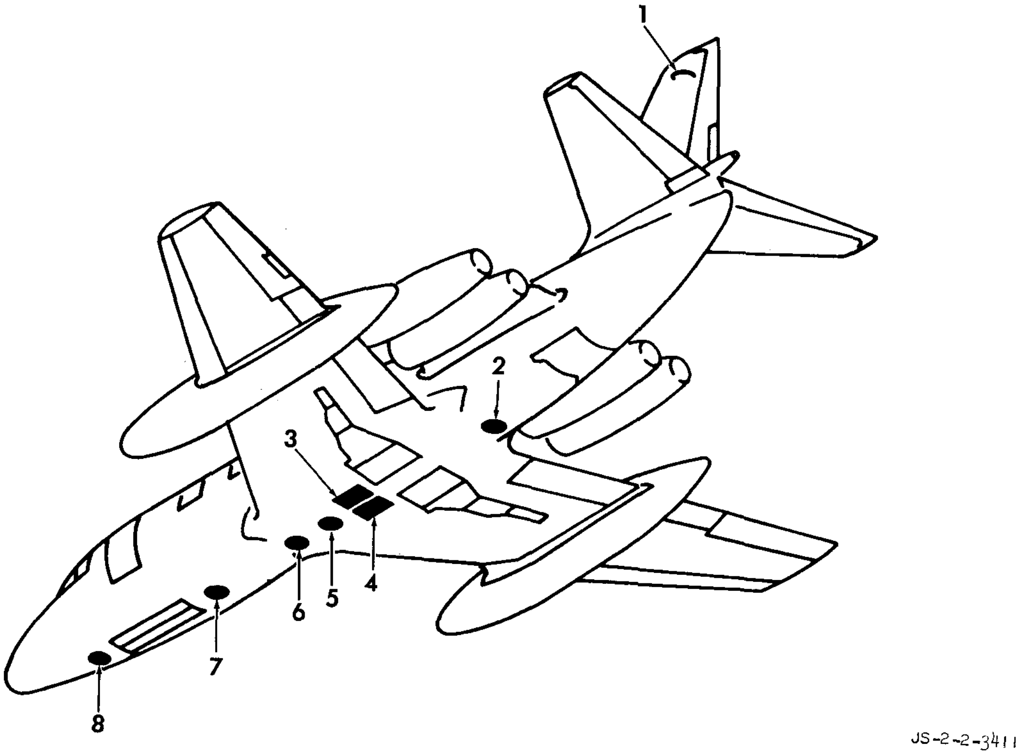


Figure 34-3. External Antenna Locations

SECTION 34.1
VHF NAVIGATION SYSTEMS (VOR)

Contents

	Page
DESCRIPTION AND OPERATION	34-11
General	34-11
List of Components	34-12
Control Panels	34-12
Receivers	34-12
Instrumentation Units	34-12
VOR Antenna	34-12
OPERATIONAL CHECKOUT	34-12
Test Equipment Required	34-12
VOR Systems	34-13
TROUBLESHOOTING	34-13
REMOVAL AND REPLACEMENT	34-14
VOR Antenna	34-14
FIGURES	
34-4. VOR-Localizer Systems Components	34-16
34-5. VOR-Localizer Systems Block Diagram	34-17

34.1.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.1.1.1 GENERAL. The VHF navigation system (VOR) consists of two VOR-localizer systems that give the direction to a VOR or localizer station. The distance measuring system (DME) determines the distance of the airplane from a VOR TAC or a TACAN station. Thus, the distance and direction to a known point determines the position of the airplane. A VOR-localizer system with a corresponding flight director system (FDS) enables the airplane to fly an omnirange course. The VOR-localizer system furnishes information to the corresponding flight director system during an ILS approach. The two VOR-localizer systems are separate, but use a common antenna which is installed on the vertical stabilizer. Each VOR-localizer system contains a receiver, an instrumentation unit, a control panel, and a power relay (see figure 34-4). The VOR-localizer systems provide for selecting and maintaining a fixed course referenced to or from a VOR or localizer ground station by either of the VOR-localizer systems. Both systems operate over a frequency range of 108.0 through 117.9 megacycles with a 100-kilocycle spacing between channels. VOR deviation signals are presented visually on a corresponding flight director system course indicator by means of a course bar, and VOR bearing signals are presented visually on a corresponding gyrosyn compass indicator by radio magnetic indicator (RMI) pointers (refer to Section 34.6). During an instrument landing system (ILS) operation, a computed steering signal is shown on the corresponding approach horizon indicator by a steering pointer. The navigation receiver audio signals can be monitored over the pilots' headsets when routed through the AUDIO SELECTOR control panel, or over the pilots' compartment speaker when selected on the overhead audio control panel. The No. 1 VOR-localizer system receives power from the essential AC and the essential DC buses through a VOR NO. 1 AC and a VOR NO. 1 DC circuit breaker on the pilot's side circuit breaker panel. The No. 2 VOR-localizer system receives power from the main AC and the main DC buses through a VOR NO. 2 AC and a VOR NO. 2 DC circuit breaker on the copilot's wainscot circuit breaker panel.

34.1.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Receiver	2	Collins	51X-2 △1	522 0833 054
Instrumentation Unit	2	Collins	344B-1A △1	522 1852 015
VOR TAC/VHF Control Panel	1	Gables	—	G-906V △2
VOR/VHF Control Panel	1	Gables	—	G-905V △2
VOR TAC/VHF Control Panel	1	Gables	—	G-906VA △1 △3
VOR/VHF Control Panel	1	Gables	—	G-905VA △1 △3
VOR Antenna	1	Dorne-Margolin		DMN4-8

△1 On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

△2 Airplanes 5001 thru 5038.

△3 Airplanes 5039 thru 5049, 5052, 5056 and 5059.

34.1.1.3 CONTROL PANELS. The No. 1 VOR-localizer system is controlled from the VOR TAC-VHF panel on the pilot's side of the center console, and the No. 2 VOR-localizer system from the VOR-VHF panel on the copilot's side. Each of these two panels has a VOR-localizer frequency selector and a VOL control. The frequency dialed with a frequency selector appears on an adjacent selected-frequency indicator. When a VOR-localizer system is turned on with its VOL control knob, a corresponding VOR power relay is energized. This connects both AC and DC power to the system. The audio output signal from a receiver can be increased by turning the corresponding VOL control knob clockwise.

34.1.1.4 RECEIVERS. The 51X-2 receivers are crystal-controlled, double-conversion, superheterodyne receivers which can be tuned over a frequency range of 108.0 through 118.0 megacycles with a 100 kilocycle spacing between channels. The receivers are remotely controlled by the corresponding VOR control panel located on the center console. These receivers are interchangeable with the Collins 51X-2 receivers used in the VHF communication systems (see Chapter 23).

34.1.1.5 INSTRUMENTATION UNITS. Input power at 115-volts AC and 28-volts DC is supplied to either Collins 344B-1A instrumentation unit when its corresponding VOR power relay is energized. The signals from the VOR ground station are routed through the VOR-localizer receiver. The reference-phase output signals from the receiver are fed into the instrumentation unit. Also, a gyrosyn compass signal is introduced into the unit. (See Section 34.6.) Output signals are sent from the instrumentation unit to the corresponding course indicator, steering computer, and approach horizon. Additionally, signals to the autopilot are provided by the No. 2 VOR-localizer, on airplanes 5001 thru 5045, and by the No. 1 VOR-localizer on airplanes 5046 and up.

34.1.1.6 VOR ANTENNA. A Dorne-Margolin Type DMN4-8 antenna is installed on the vertical stabilizer. The antenna consists of two separate half loops, one of which is mounted on the left side of the stabilizer and the other is mounted directly opposite on the right side. These are connected by coaxial cables to a tee-connector inside the vertical stabilizer. The DMN4-8 antenna is horizontally-polarized to receive VOR-localizer signals (see figure 34-4).

34.1.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Collins 479-T2	Signal Generator	Supplies directional signal to receiver.

34.1.2.1 VOR SYSTEMS. The interphone system and gyrosyn compass system must be installed and operational before performing this operation.

a. Connect external power to the airplane (see Chapter 24) and energize the AC buses.

b. Make sure the following circuit breakers are closed: VOR NO. 1 AC, VOR NO. 1 DC, COMPASS NO. 1 26 VAC, COMPASS NO. 1 115 VAC, INPH, and COPILOT ISOLATION AMP.

c. Make sure one of the headsets is connected, and the VOR-1 monitor switch on the corresponding AUDIO SELECTOR panel is turned on.

d. Set up a Collins 479-T2 signal generator approximately 75 feet from the VOR-localizer antenna, with the test antenna connected to the NAV OUTPUT terminal of the signal generator.

e. Set the ADF-VOR knob on the pilot's and the copilot's gyrosyn compass indicators in the VOR No. 1 and No. 2 position. Then the single pointers indicate VOR No. 1 readings, and the double pointers No. 2.

f. Dial 110.1 megacycles with the No. 1 VOR-localizer frequency selector, and turn on the VOL control.

Note

Gentle tapping of the indicator cases during this test is permissible and might be necessary to prevent sticking of the meters. When the No. 1 VOR-localizer frequency selector is dialed to between 108.0 and 117.9 megacycles, and when the No. 1 VOR-localizer VOL ON-OFF control is turned on, the single pointer on the pilot's gyrosyn compass indicator might continue to rotate. This condition is permissible.

g. On the signal generator front panel, set the NAV-GS switch in the "NAV" position, set the frequency selector at 110.1 megacycles, the AUDIO SELECTOR in the "VOR" position, and adjust the MICROVOLTS control for maximum output.
RESULT: A 1000 CPS tone should be heard through the headset.

h. Repeat steps (f) and (g) using the following frequencies: 108.0, 111.0, 114.9, and 117.9 megacycles.

i. Dial 114.9 megacycles with the No. 1 VOR-localizer frequency selector.

j. On the signal generator front panel, set the frequency selector at 114.9 megacycles, and adjust the MICROVOLTS control for 5000-microvolts output.
RESULT: The LOC warning flag on the pilot's approach horizon should disappear.

k. Set the VOR control on the signal generator at 0 degrees.
RESULT: The single pointer on the gyrosyn compass indicators should align with the lubber line within ±3.0 degrees.

l. Set the pilot's approach horizon HDG-ILS knob in the "HDG" position, and set the pilot's course indicator course arrow 10 degrees clockwise from the lubber line.
RESULT: The pilot's course indicator course bar should displace full scale left, and the to-from arrow should point in the same direction as the course arrow.

m. Align the pilot's course indicator course arrow with the lubber line.
RESULT: The pilot's course indicator course bar should align with the course arrow and the lubber line, and the to-from arrow should point in the same direction as the course arrow.

n. Set the pilot's course indicator course arrow 10 degrees counterclockwise from the lubber line.
RESULT: The pilot's course indicator course bar should displace full scale right, and the to-from arrow should point in the same direction as the course arrow.

o. Set the VOR control on the signal generator at successive 45 degree intervals from 0 to 360 degrees.
RESULT: The readings of both the pilot's and the copilot's gyrosyn compass indicator No. 1 pointers should be within ±3.0 degrees of the heading selected on the signal generator.

p. Repeat steps (b) through (o) for the No. 2 VOR-localizer system.

q. Disconnect external power and restore the airplane to normal.

34.1.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

VOR Manuals, Collins Part Numbers 523 0022 006, 523 0022 004, 523 0022 003, 520 5598 00. No manuals are available for the Gables control panels. These should be returned to the manufacturer for repair.

34.1.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
POWER CONTROL SWITCH "ON", BUT NO INDICATION FROM ANY INDICATOR.	
VOR No. 1 or VOR No. 2 circuit breaker open.	Check circuit breakers for correct setting.
Defective VOR No. 1 or VOR No. 2 power relay.	Check relay for proper operation.
Defective receiver.	Substitute a new receiver and check for proper operation.
Defective instrumentation unit.	Substitute a new instrumentation unit and check for proper operation.
Loose or damaged cabling between system components.	Inspect cabling for damaged or loose connectors.
POWER CONTROL SWITCH "ON," BUT ERRONEOUS INDICATION ON ONE OR MORE INDICATORS.	
Defective receiver. Defective indicator. Fault in indication received from gyrosyn compass system.	Observe all navigation indicators. If one indicator gives erroneous indications and the others give normal indications, indicator giving erroneous indications is probably defective. If all indicators give the same erroneous indications, trouble is probably in receiver or gyrosyn compass system operation.
ERRATIC INDICATION ON BOTH PILOT'S AND COPILOT'S INDICATORS.	
Defective coaxial cables.	Check coaxial cable from antenna to coaxial tee connector and from tee connector to both receivers. Defective cable between tee and receiver can cause erratic indication on both systems.

34.1.4 REMOVAL AND REPLACEMENT.

34.1.4.1 VOR ANTENNA.

a. Open the VOR ANTENNA ACCESS door, located on the left side of the vertical stabilizer, just below the VOR antenna.

b. Reach through the access door and disconnect the antenna cable connector.

c. Remove the six screws, at each end of the antenna, which secure the antenna to the left side of the vertical stabilizer.

d. Remove the antenna from the left side of the stabilizer.

e. Remove the six screws, at each end of the antenna, which secure the antenna to the right side of the vertical stabilizer.

f. Remove the antenna from the right side of the stabilizer.

g. To install the VOR antenna, reverse the procedure of steps (a) through (f).

h. Perform an operational checkout of the VOR-localizer system.

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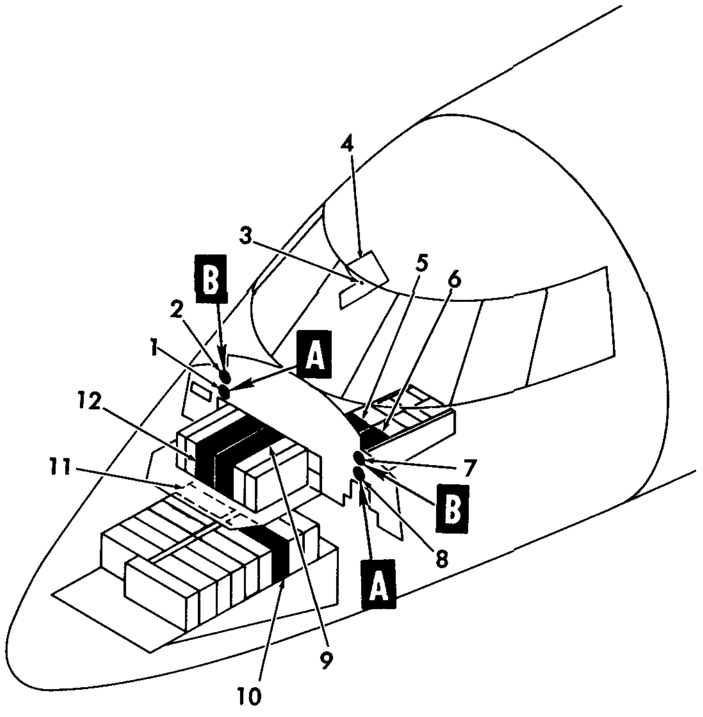
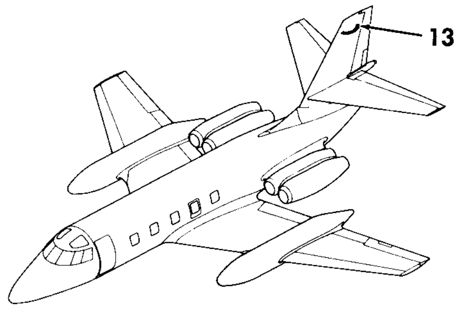
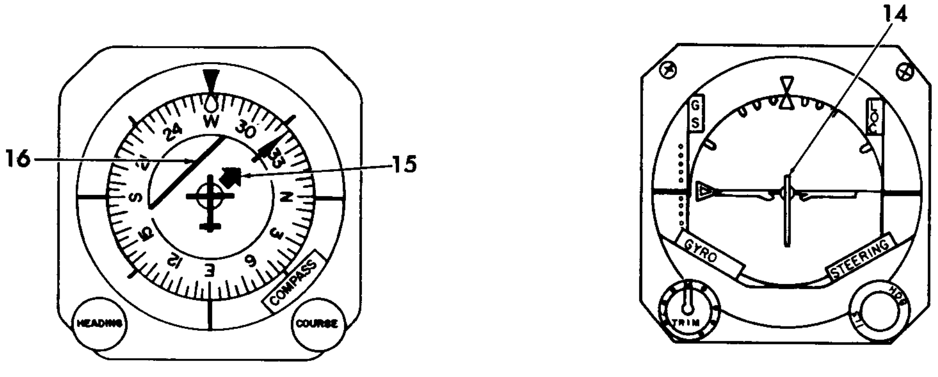




1. COPILOT'S COURSE INDICATOR
2. COPILOT'S APPROACH HORIZON
3. NO. 2 VOR POWER RELAY
4. NO. 1 VOR POWER RELAY
5. VOR-VHF PANEL
6. VORTAC - VHF PANEL
7. PILOT'S APPROACH HORIZON
8. PILOT'S COURSE INDICATOR
9. NO. 1 VOR INSTRUMENTATION UNIT
10. NO. 1 VOR RECEIVER
11. NO. 2 VOR RECEIVER (OPPOSITE NO. 1 RCVR)
12. NO. 2 VOR INSTRUMENTATION UNIT
13. VOR ANTENNA
14. STEERING POINTER
15. TO-FROM ARROW
16. COURSE BAR



A
COURSE INDICATOR
(TYPICAL)
(LOOKING FORWARD)

B
APPROACH HORIZON
(TYPICAL)
(LOOKING FORWARD)

JS-2-2-3412

Figure 34-4. VOR-Localizer Systems Components

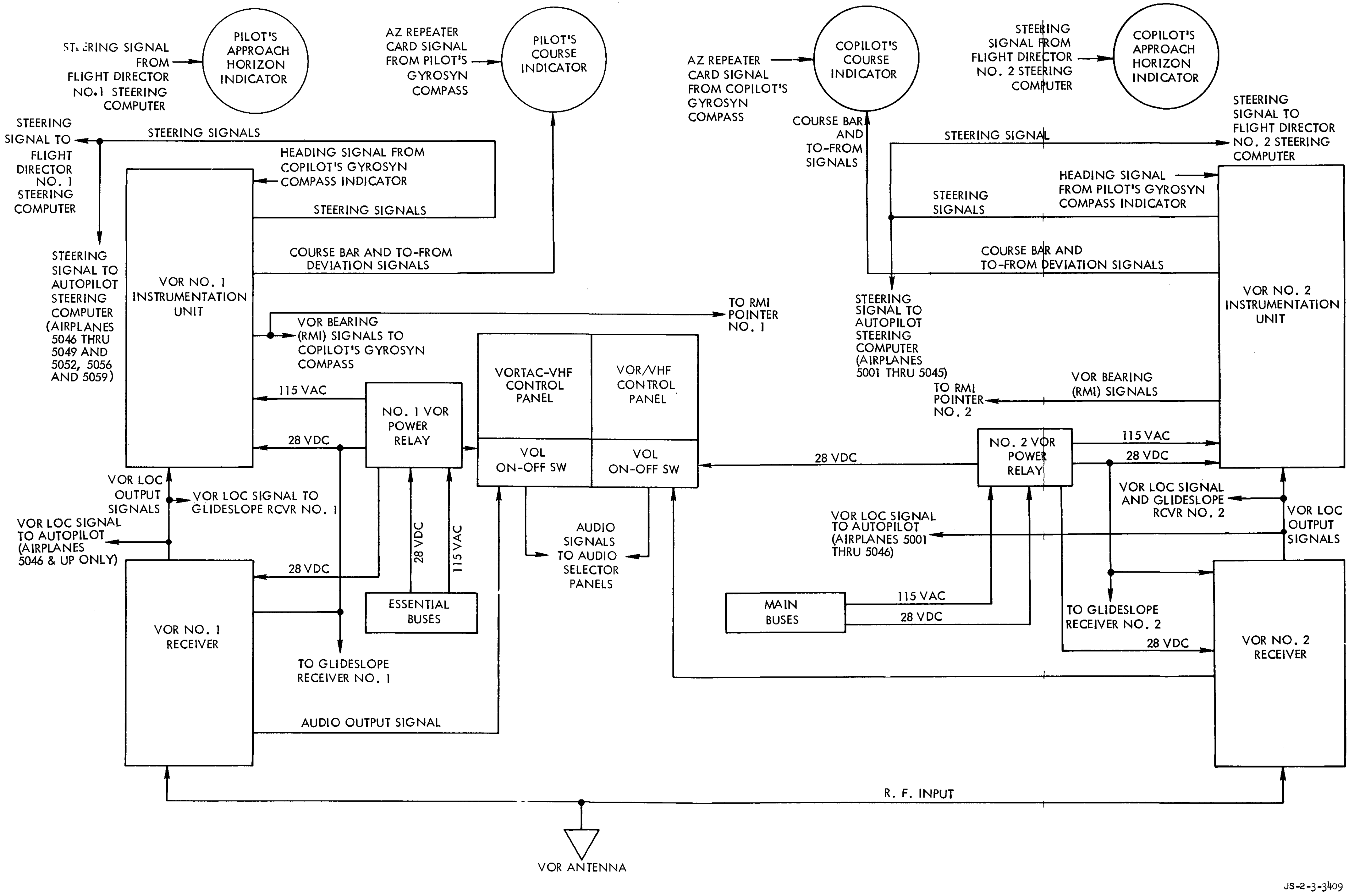

STEERING SIGNAL FROM FLIGHT DIRECTOR NO. 1 STEERING COMPUTER
PILOT'S APPROACH HORIZON INDICATOR
AZ REPEATER CARD SIGNAL FROM PILOT'S GYROSYN COMPASS
PILOT'S COURSE INDICATOR
AZ REPEATER CARD SIGNAL FROM COPILOT'S GYROSYN COMPASS
COPILOT'S COURSE INDICATOR
STEERING SIGNAL FROM FLIGHT DIRECTOR NO. 2 STEERING COMPUTER
COPILOT'S APPROACH HORIZON INDICATOR
STEERING SIGNAL TO FLIGHT DIRECTOR NO. 1 STEERING COMPUTER
STEERING SIGNALS
HEADING SIGNAL FROM COPILOT'S GYROSYN COMPASS INDICATOR
STEERING SIGNALS
COURSE BAR AND TO-FROM DEVIATION SIGNALS
STEERING SIGNAL TO AUTOPILOT STEERING COMPUTER (AIRPLANES 5046 THRU 5049 AND 5052, 5056 AND 5059)
VOR NO. 1 INSTRUMENTATION UNIT
VOR BEARING (RMI) SIGNALS TO COPILOT'S GYROSYN COMPASS
TO RMI POINTER NO. 1
COURSE BAR AND TO-FROM SIGNALS
STEERING SIGNAL
STEERING SIGNAL TO FLIGHT DIRECTOR NO. 2 STEERING COMPUTER
STEERING SIGNALS
HEADING SIGNAL FROM PILOT'S GYROSYN COMPASS INDICATOR
COURSE BAR AND TO-FROM DEVIATION SIGNALS
VOR NO. 2 INSTRUMENTATION UNIT
STEERING SIGNAL TO AUTOPILOT STEERING COMPUTER (AIRPLANES 5001 THRU 5045)
TO RMI POINTER NO. 2
VOR BEARING (RMI) SIGNALS
115 VAC
28 VDC
NO. 1 VOR POWER RELAY
VORTAC-VHF CONTROL PANEL
VOR/VHF CONTROL PANEL
VOL ON-OFF SW
VOL ON-OFF SW
28 VDC
NO. 2 VOR POWER RELAY
115 VAC
28 VDC
VOR LOC OUTPUT SIGNALS
VOR LOC SIGNAL TO GLIDESLOPE RCVR NO. 1
VOR LOC SIGNAL TO AUTOPILOT (AIRPLANES 5046 & UP ONLY)
28 VDC
115 VAC
AUDIO SIGNALS TO AUDIO SELECTOR PANELS
VOR LOC SIGNAL AND GLIDESLOPE RCVR NO. 2
VOR LOC OUTPUT SIGNALS
VOR LOC SIGNAL TO AUTOPILOT (AIRPLANES 5001 THRU 5046)
28 VDC
ESSENTIAL BUSES
MAIN BUSES
115 VAC
28 VDC
VOR NO. 1 RECEIVER
TO GLIDESLOPE RECEIVER NO. 1
TO GLIDESLOPE RECEIVER NO. 2
28 VDC
VOR NO. 2 RECEIVER
AUDIO OUTPUT SIGNAL
R. F. INPUT
VOR ANTENNA
JS-2-3-3409

Figure 34-5. VOR-Localizer Systems Block Diagram

SECTION 34.2
FLIGHT DIRECTOR SYSTEMS

Contents . . . Page

DESCRIPTION AND OPERATION . . . 34-19
  General . . . 34-19
  List of Components . . . 34-20
  Course Indicators . . . 34-20
  Approach Horizon Indicators . . . 34-20
  Instrument Amplifier . . . 34-21
  Steering Computer . . . 34-21
  Vertical Gyros . . . 34-21
  Interlock Relays . . . 34-21
  Phase Adapters . . . 34-21

OPERATIONAL CHECKOUT . . . 34-21
  Test Equipment Required . . . 34-21
  Flight Director Systems . . . 34-21

TROUBLESHOOTING . . . 34-23

REMOVAL AND INSTALLATION . . . 34-25
  Removal of the Vertical Gyros . . . 34-25
  Installation of the Vertical Gyros . . . 34-25

FIGURES
  34-6. Flight Director Systems Components . . . 34-27
  34-7. Flight Director System No. 1 Block Diagram . . . 34-27
  34-8. Flight Director System No. 2 Block Diagram . . . 34-29

34.2.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.2.1.1 GENERAL. Two flight director systems provide visual flight information for pilots' observation for flying VOR omnidirectional range courses, for flying instrument landing system (ILS) approaches, for flying compass headings, and maintaining proper pitch and roll attitudes. Each system includes a course indicator, an approach horizon indicator, an instrument amplifier, a steering computer, a vertical gyro, and interlock relays. AC power at 115 volts AC, 400 CPS is supplied either of the two flight director systems through a FLT DIR HDG and a FLT DIR ATT circuit breaker. Flight director system No. 1 receives power from the essential AC bus through the pilot's side circuit breaker panel, and the flight director system No. 2 receives power from the main AC bus through the copilot's wainscot circuit breaker panel. The course indicators and the approach horizon indicators are located on the pilots' main instrument panel. The instrument amplifiers, the steering computers, and the vertical gyros are in the nose compartment, on the electronic equipment rack upper shelf. The interlock relays and the phase adapters are mounted on the bulkhead in the nose compartment.

34.2.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Course Indicator	2	Collins	331A-4 △1	522 0891 005
Approach Horizon Indicator	2	Collins	329B-4A △1	522 0892 005
Instrument Amplifier	2	Collins	344C-1 △1	522 0893 004
Steering Computer	2	Collins	562A-5 △1	522 0894 004
Vertical Gyro No. 1	1	Sperry	JR1000-35 or JR1000-65 △1	1782356-111 or 2586137-1
Vertical Gyro Interlock Relays				
AC Interlock Relays	2	Hartman		C12-A
DC Interlock Relay	2			AN3311-2
Phase Adapter	2	Sperry	JR1000-9	661102B

△1 On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

34.2.1.3 COURSE INDICATORS. A plan view of the airplane with respect to a selected omnirange radial or to a selected magnetic heading is shown on a course indicator. The position of the airplane is simulated on the indicator by a stationary miniature airplane. A stationary lubber line at the top of the course indicator points to the magnetic heading of the airplane on the azimuth ring of a compass repeater card. A desired magnetic heading can be selected by turning a HEADING knob, on the lower left face of the course indicator, which moves a heading marker to the desired point on the azimuth ring. Once set, the heading marker moves with the azimuth ring as the ring rotates. When the heading marker aligns with the lubber line, the airplane is flying the selected heading. An omnirange radial can be selected by turning a COURSE knob, on the lower right face of the course indicator, which moves a course arrow to the desired point on the azimuth ring. When the airplane is displaced from the selected omnirange or localizer course, a course bar moves parallel to the course arrow, and perpendicular to the horizontal row of dots. A to-from arrow indicates the direction of the station so a distinction can be made as to whether the airplane is flying to or from the station. During an instrument landing system (ILS) operation, the to-from arrow is not visible. A COMPASS flag appears when heading power is lost, when there is no compass monitor voltage, or when the compass synchro error signal exceeds 4.5 degrees.

34.2.1.4 APPROACH HORIZON INDICATORS. A pictorial display of the airplane with respect to the glideslope, of pitch and bank attitude, and of computed lateral guidance is shown on an approach horizon indicator. During an instrument landing system (ILS) approach operation, a glideslope pointer relates the descent of the airplane with respect to the glideslope. If the glideslope pointer is above the center of the indicator, the glideslope is above the airplane. If the glideslope pointer is below the center, the glideslope is below the airplane. When the glideslope signal strength falls below a reliable level, a GS flag appears. A horizon, pivoted at the center, indicates the roll or bank of the airplane, independent of the pitch attitude. The degree of bank is shown by a bank pointer at the top center of the movable horizon. A HDG-ILS knob permits switching to either of two modes of operation. In the HDG mode, a steering pointer is fed heading error and bank information. In the ILS mode, localizer deviation, synthetic heading, and bank information are fed to the steering pointer. When either heading or attitude power to the steering computer is lost, a STEERING flag is visible. A pitch bar moves in a vertical plane above and below the center of the indicator to show either nose up or nose down attitude. Except in an ILS operation when the pitch trim is automatically set, a pitch TRIM control positions the pitch bar at the desired flight attitude. When the localizer signal falls below the required operating leval a LOC flag is visible. A gyro flag appears when any of the following conditions are present:

34.2.1.4 (Cont).

a. The pitch synchro error is greater than 4.5 degrees.

b. The bank synchro error is greater than 4.5 degrees.

c. Inadequate attitude power is present.

d. Vertical gyro malfunction.

34.2.1.5 INSTRUMENT AMPLIFIER. The instrument amplifier is composed of five plug-in modules mounted on a dual chassis. One module is a power supply, one is a warning flag module, and three are servoamplifier modules. The flag module provides control signals to the compass flag on the course indicator, and to the GYRO flag on the approach horizon indicator. A pitch servoamplifier provides output voltage that drives a servomotor which positions the pitch bar of the approach horizon indicator. A bank servoamplifier provides output voltage that drives a servomotor which rotates the horizon of the approach horizon indicator. The third servoamplifier module is a compass servoamplifier that provides output voltage which drives the servomotor that positions the azimuth ring of the course indicator.

34.2.1.6 STEERING COMPUTER. The steering computer supplies computed lateral guidance signals to the steering pointer of the approach horizon indicator. Four information signals are used by the steering computer to obtain the lateral guidance signals. The steering computer will function in one of two modes of operation, HDG or ILS. In the HDG mode, the computer uses the bank signal from the vertical gyro, and the selected heading error signal from the course indicator. In the ILS mode, the bank signal from the vertical gyro, the localizer deviation signal from the VOR receiver, and the course datum signal from the course indicator are used to obtain the steering pointer signal. In the HDG mode, the steering pointer information provides for either maintaining or coming to a selected magnetic heading; in the ILS mode, the steering pointer information provides for either maintaining or approaching an inbound localizer course.

34.2.1.7 VERTICAL GYROS. Two vertical gyros establish stable pitch and roll attitude reference signals. Vertical gyro No. 1 supplies signals to flight director system No. 1 as well as to the weather radar system, and vertical gyro No. 2 supplies signals to flight director system No. 2 as well as to the autopilot system. Power is supplied to each vertical gyro through corresponding VERT GYRO AC and VERT GYRO DC circuit breakers and interlock relays.

CAUTION

Improper handling of the vertical gyro circuit breakers can cause the vertical gyros to erect in an inverted position. After opening the circuit breakers allow five minutes for the gyros to run down before closing the breakers again.

34.2.1.8 INTERLOCK RELAYS. Neither AC nor DC power is supplied to a gyro unless both 115-volt, AC and 28-volt, DC power is available on the proper buses. This is controlled by a pair of interlock relays for each gyro.

34.2.1.9 PHASE ADAPTERS. Single phase AC power for each gyro is routed through contacts of a DC interlock relay and through a phase adapter to furnish the three-phase, wye-connected, AC power required by the vertical gyros.

34.2.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Collins 479-T2	Signal generator	Supplies directional signal to receiver.
Lockheed 11031-TK	Synchro simulator	Simulates output signals of vertical gyros.

34.2.2.1 FLIGHT DIRECTOR SYSTEMS.

a. Connect external power to the airplane (refer to Chapter 24), and energize the AC buses.

b. Make sure the following circuit breakers are closed: FLT DIR NO. 1 HDG and ATT, COMPASS No. 1 26 VAC and 115 VAC, VOR NO. 1 AC and DC, and VERT GYRO NO. 1 DC and AC.

c. Turn on the No. 1 VOR-localizer system by turning the VOR, VOL ON-OFF control, on the pilot's side of the center console, in a clockwise direction. Allow a 5-minute warmup for the receiver system.
RESULT: The GYRO and the STEERING warning flags on the pilot's approach horizon indicator and the COMPASS warning flag on the pilot's course indicator should disappear. The azimuth ring on the pilot's course indicator should read the same as the pilot's gyrosyn compass indicator.

d. Disconnect the No. 1 vertical gyro from the No. 1 flight director system at the vertical gyro. Connect a Lockheed 11031-TK synchro simulator to the No. 1 vertical gyro. All connections to the simulator are made through a single test cable,

34.2.2.1 (Cont).

one end of which mates with a panel connector on the simulator. The other end of the test cable mates with the appropriate No. 1 vertical gyro receptacle.

e. Zero all inputs to the synchro simulator and set the pilot's approach horizon indicator HDG-ILS knob to the "HDG" position.

f. Set the synchro simulator 15 degrees up in pitch.
RESULT: The pitch bar on the pilot's approach horizon indicator should move up 4 dots.

g. Set the synchro simulator 15 degrees down in pitch.
RESULT: The pitch bar on the pilot's approach horizon indicator should go down 4 dots below center.

h. Set the synchro simulator at 0 degrees pitch.
RESULT: The pitch bar on the pilot's approach horizon indicator should center.

i. Check that all inputs to the flight director system are zero.

j. Set the synchro simulator 15 degrees LEFT bank.
RESULT: The horizon on the pilot's approach horizon indicator should rotate to show 15 degrees of left bank, and the steering pointer should deflect to the right.

k. Set the synchro simulator 15 degrees RIGHT bank.
RESULT: The horizon on the pilot's approach horizon should rotate to show 15 degrees of right bank, and the steering pointer should deflect to the left.

l. Set the synchro simulator at 0 degrees bank.
RESULT: The horizon on the pilot's approach horizon should rotate to the level position, and the steering pointer should be vertical.

m. Using the HEADING knob, set the heading marker on the pilot's course indicator 10 degrees to the left of the lubber line.
RESULT: The steering pointer on the pilot's approach horizon indicator should deflect to the left.

n. Adjust the synchro simulator so the steering pointer, on the pilot's approach horizon indicator, is centered.
RESULT: The synchro simulator should indicate 15 (±2) degrees of left bank.

o. Set the heading marker on the pilot's course indicator 90 degrees to the left of the lubber line.
RESULT: The steering pointer on the pilot's approach horizon indicator should deflect to the left.

p. Adjust the synchro simulator so the steering pointer, on the pilot's approach horizon indicator, is centered.
RESULT: The synchro simulator should indicate 30 (±2) degrees of left bank.

q. Repeat steps (m) through (p), substituting right for left.

r. Set the pilot's approach horizon indicator HDG-ILS knob to the "ILS" position and, using the pilot's course indicator COURSE knob, align the course arrow with the lubber line.

s. Set up a 479-T2 signal generator approximately 75 feet from the VOR-localizer antenna. Connect the test antenna to the NAV OUTPUT terminal of the signal generator.

t. On the signal generator front panel, set the NAV-GS switch to the "NAV" position, set the frequency selector at 110.1 megacycles, place the AUDIO SELECTOR in the "VOR" position, and adjust the MICROVOLTS control for maximum output.

u. Adjust the LOC-GS control on the signal generator to produce a course bar deflection of 5 dots to the right on the course indicator.
RESULT: The steering pointer on the pilot's approach horizon indicator should deflect to the right.

v. Adjust the synchro simulator until the steering pointer, on the pilot's approach horizon indicator, is centered.
RESULT: The synchro simulator should indicate 15 (±2) degrees bank.

w. Adjust the LOC-GS control on the signal generator to produce a course bar deflection of 5 dots to the left on the course indicator.
RESULT: The steering pointer on the pilot's approach horizon indicator should deflect to the left.

x. Adjust the synchro simulator until the steering pointer, on the pilot's approach horizon indicator, is centered.
RESULT: The synchro simulator should indicate 15 (±2) degrees bank.

y. Adjust the synchro simulator to 0 degrees bank.

z. Adjust the LOC-GS control on the signal generator to center the steering pointer on the pilot's approach indicator, and turn off the signal generator.

aa. Using the pilot's course indicator COURSE knob, set the course arrow first at 10 degrees to the right of the lubber line, and then 10 degrees to the left of the lubber line.
RESULT: The steering pointer on the pilot's approach horizon indicator should deflect first to the right, and then to the left.

34.2.2.1 (Cont).

ab. Turn off the No. 1 VOR-localizer VOL control, and open the following circuit breakers: FLT DIR NO. 1 HDG, FLT DIR NO. 1 ATT, COMPASS NO. 1 26 VAC, COMPASS NO. 1 115 VAC, VOR NO. 1 AC, VOR NO. 1 DC, VERT GYRO NO. 1 AC, and VERT GYRO NO. 1 DC. Disconnect the synchro simulator test cable from the No. 1 vertical gyro and reconnect the No. 1 vertical gyro to the No. 1 flight director system.

ac. Make sure the following circuit breakers are closed: FLT DIR NO. 2 HDG, FLT DIR NO. 2 ATT, COMPASS NO. 2 26 VAC, COMPASS NO. 2 115 VAC, VOR NO. 2 AC, VOR NO. 2 DC, VERT GYRO NO. 2 AC, and VERT GYRO NO. 2 DC.

ad. Turn on the No. 2 VOR-localizer system with its VOL control knob. Allow a 5-minute warmup.
RESULT: The GYRO and the STEERING warning flags on the copilot's approach horizon indicator and the COMPASS warning flag on the copilot's course indicator should disappear. The azimuth ring on the copilot's course indicator should read the same as the copilot's gyrosyn compass indicator.

ae. Disconnect the No. 2 vertical gyro from the No. 2 flight director system at the vertical gyro. Connect a Lockheed 11031-TK synchro simulator to the No. 2 vertical gyro. All connections to the simulator are made through a single test cable, one end of which mates with a panel connector on the simulator. The other end of the test cable mates with the appropriate No. 2 vertical gyro receptacle.

af. Repeat steps (e) through (aa) for the No. 2 flight director system, substituting the word "copilot's" for "pilot's" in all instances.

ag. Open all circuit breakers, turn off the No. 2 VOL-localizer system, and disconnect external power. Disconnect the synchro simulator test cable from the No. 2 vertical gyro, and reconnect the No. 2 vertical gyro to the No. 2 flight director system.

34.2.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

Flight Director Manuals, Collins Part Numbers 523 0019 006, 523 0019 004, 523 0019 003.

PROBABLE CAUSE	ISOLATION PROCEDURE
HORIZON ON APPROACH HORIZON INDICATOR DOES NOT GIVE CORRECT INDICATION ON GROUND WHEN AIRPLANE IS LEVEL.	
Defective instrument amplifier.	Substitute new instrument amplifier and check horizon indication.
HORIZON ON APPROACH HORIZON INDICATOR DOES NOT DEFLECT WHEN AIRPLANE IS BANKED OR VERTICAL GYRO IS TILTED.	
Defective vertical gyro.	Substitute a test synchro or a new vertical gyro and check for proper operation.
Defective instrument amplifier.	Substitute a new instrument amplifier and check for proper operation.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
Loss of AC or DC power caused the vertical gyro to topple.	Check VERTICAL GYRO circuit breakers. (NOTE: Vertical gyro should be erect after 3 minutes of operation.)
PITCH BAR ON APPROACH HORIZON INDICATOR DOES NOT DEFLECT WHEN AIRPLANE IS PITCHING OR VERTICAL GYRO IS TILTED ON PITCH AXIS.	
Defective vertical gyro.	Substitute a new vertical gyro and check for proper operation.
Defective instrument amplifier.	Substitute a new instrument amplifier and check for proper operation.

34.2.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
PITCH BAR ON APPROACH HORIZON INDICATOR DOES NOT DEFLECT WHEN AIRPLANE IS PITCHING OR VERTICAL GYRO IS TILTED ON PITCH AXIS (Cont).	
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
Loss of AC or DC power caused the vertical gyro to topple.	Check VERTICAL GYRO circuit breakers. (NOTE: Vertical gyro should be erect after 3 minutes of operation.)
NO INDICATION ON APPROACH HORIZON INDICATOR GLIDESLOPE POINTER.	
Defective glideslope receiver.	Check if GS flag on approach horizon indicator is in view.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
GYRO FLAG ON APPROACH HORIZON INDICATOR REMAINS IN VIEW AFTER 3-MINUTE WARMUP.	
Defective vertical gyro.	Substitute a new vertical gyro and check for proper operation.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
STEERING POINTER ON APPROACH HORIZON INDICATOR DOES NOT RESPOND TO SIGNALS.	
Open circuit breaker.	Check FLIGHT DIRECTOR circuit breakers.
Defective steering computer.	Substitute a new steering computer and check for proper operation.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
STEERING POINTER DOES NOT CENTER WITH ZERO BANK, COURSE DEVIATION, AND HEADING ERROR SIGNALS.	
Defective steering computer.	Substitute a new steering computer and check for proper operation.
STEERING POINTER FLUCTUATES DURING FLIGHT.	
Defective steering computer.	Substitute a new steering computer and check for proper operation.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.
STEERING POINTER DOES NOT DEFLECT WHEN HEADING SELECTOR KNOB IS ROTATED.	
Defective steering computer.	Substitute a new steering computer and check for proper operation.
Defective approach horizon indicator.	Substitute a new approach horizon indicator and check for proper operation.

34.2.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
HORIZON INVERTED ON APPROACH HORIZON INDICATOR.	
Gyro erected in inverted position.	This condition can occur on newly installed gyros and is caused by improper handling prior to installation. Switch off gyro by pulling gyro circuit breakers. Wait at least five minutes and switch on gyro. Condition should correct itself. If not, replace.

34.2.4 REMOVAL AND INSTALLATION.

34.2.4.1 REMOVAL OF THE VERTICAL GYROS.

a. Open the nose cone, as described in 34.0.3.1, sufficiently to expose the vertical gyros.

b. Disconnect the electrical cable, at the connector adjacent to the vertical gyro.

c. Remove the three securing screws, taking particular note of the number and arrangement of any shimming washers and adapters. Remove the vertical gyro.

CAUTION

Take great care not to jar the vertical gyro, also do not rapidly invert the gyro, as this may cause inverted erection when the gyro is reinstalled.

34.2.4.2 INSTALLATION OF THE VERTICAL GYROS.

a. Carefully position the gyro and secure with the existing screws. Ensure that the existing shimming washers are installed in their original positions.

b. On airplanes 5122 and up, use part number JR259-1 wedge assembly.

c. Reconnect the electrical connector.

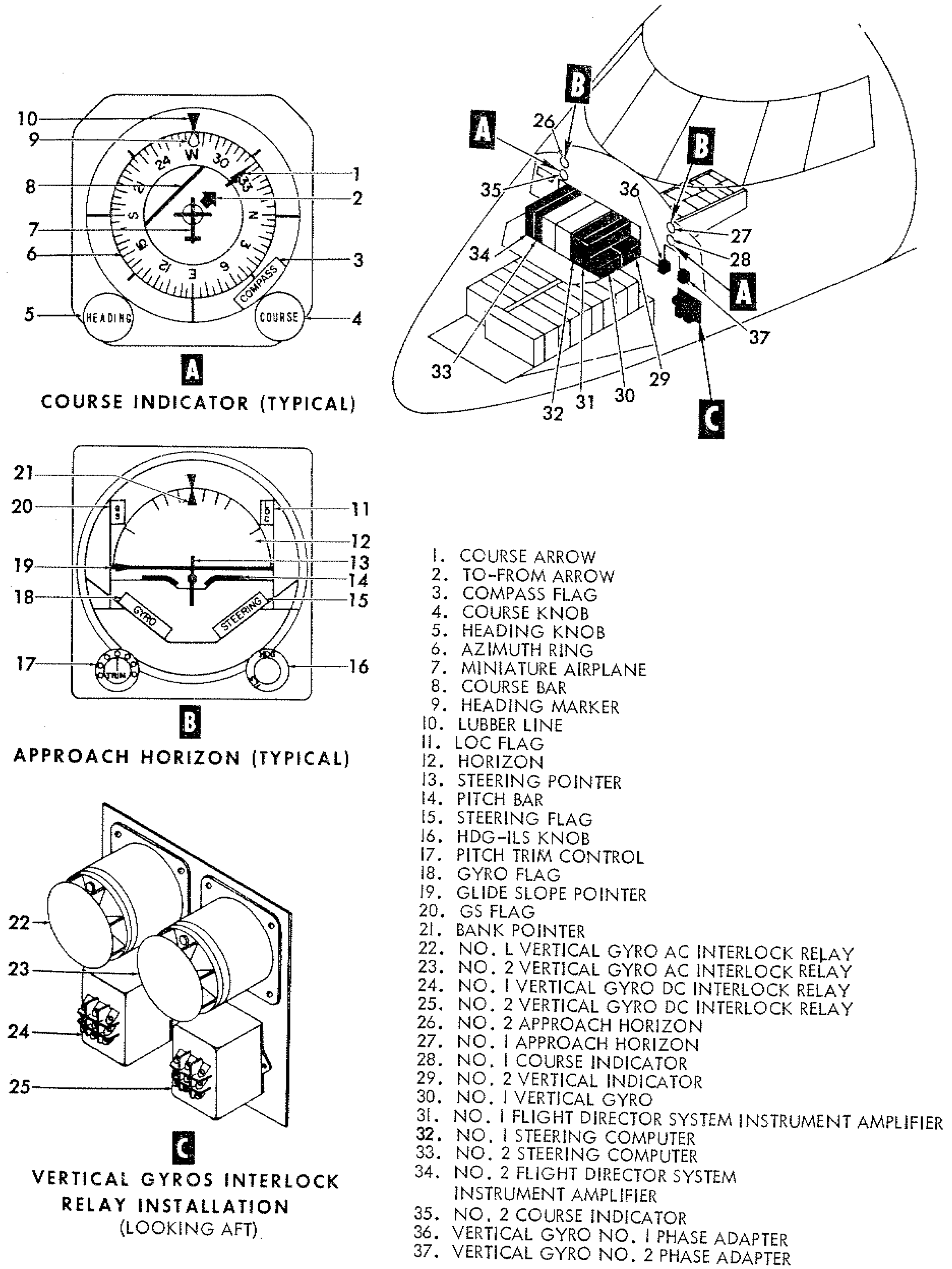
d. If the position of the shimming washers was not noted, it will be necessary to level the airplane and add shimming washers AN960-D-8 until the Approach Horizon indicates level within ± 1/2° of roll. With the airplane level, the Approach Horizon will indicate level if the vertical gyro is mounted directly to the avionics shelf, and 3° nose down if the vertical gyro is mounted to a wedge bracket.

e. On airplanes 5001 thru 5043: to replace the JR1000-35 gyro with a JR1000-65 gyro, it will be necessary to install a single JR1000-65 gyro in the forward position and use an adapter Part Number JR451-2. The adapter is installed on the gyro and the assembly is then mounted in the aircraft, using leveling washers as required.

f. A second JR1000-65 gyro can then be added, if required at anytime, in the aft position without any further modification, and does not require an adapter plate.

g. Functionally check system.

h. Close the nose cone.



COURSE INDICATOR (TYPICAL)

APPROACH HORIZON (TYPICAL)

VERTICAL GYROS INTERLOCK RELAY INSTALLATION (LOOKING AFT)

1. COURSE ARROW
2. TO-FROM ARROW
3. COMPASS FLAG
4. COURSE KNOB
5. HEADING KNOB
6. AZIMUTH RING
7. MINIATURE AIRPLANE
8. COURSE BAR
9. HEADING MARKER
10. LUBBER LINE
11. LOC FLAG
12. HORIZON
13. STEERING POINTER
14. PITCH BAR
15. STEERING FLAG
16. HDG-ILS KNOB
17. PITCH TRIM CONTROL
18. GYRO FLAG
19. GLIDE SLOPE POINTER
20. GS FLAG
21. BANK POINTER
22. NO. L VERTICAL GYRO AC INTERLOCK RELAY
23. NO. 2 VERTICAL GYRO AC INTERLOCK RELAY
24. NO. 1 VERTICAL GYRO DC INTERLOCK RELAY
25. NO. 2 VERTICAL GYRO DC INTERLOCK RELAY
26. NO. 2 APPROACH HORIZON
27. NO. 1 APPROACH HORIZON
28. NO. 1 COURSE INDICATOR
29. NO. 2 VERTICAL INDICATOR
30. NO. 1 VERTICAL GYRO
31. NO. 1 FLIGHT DIRECTOR SYSTEM INSTRUMENT AMPLIFIER
32. NO. 1 STEERING COMPUTER
33. NO. 2 STEERING COMPUTER
34. NO. 2 FLIGHT DIRECTOR SYSTEM INSTRUMENT AMPLIFIER
35. NO. 2 COURSE INDICATOR
36. VERTICAL GYRO NO. 1 PHASE ADAPTER
37. VERTICAL GYRO NO. 2 PHASE ADAPTER

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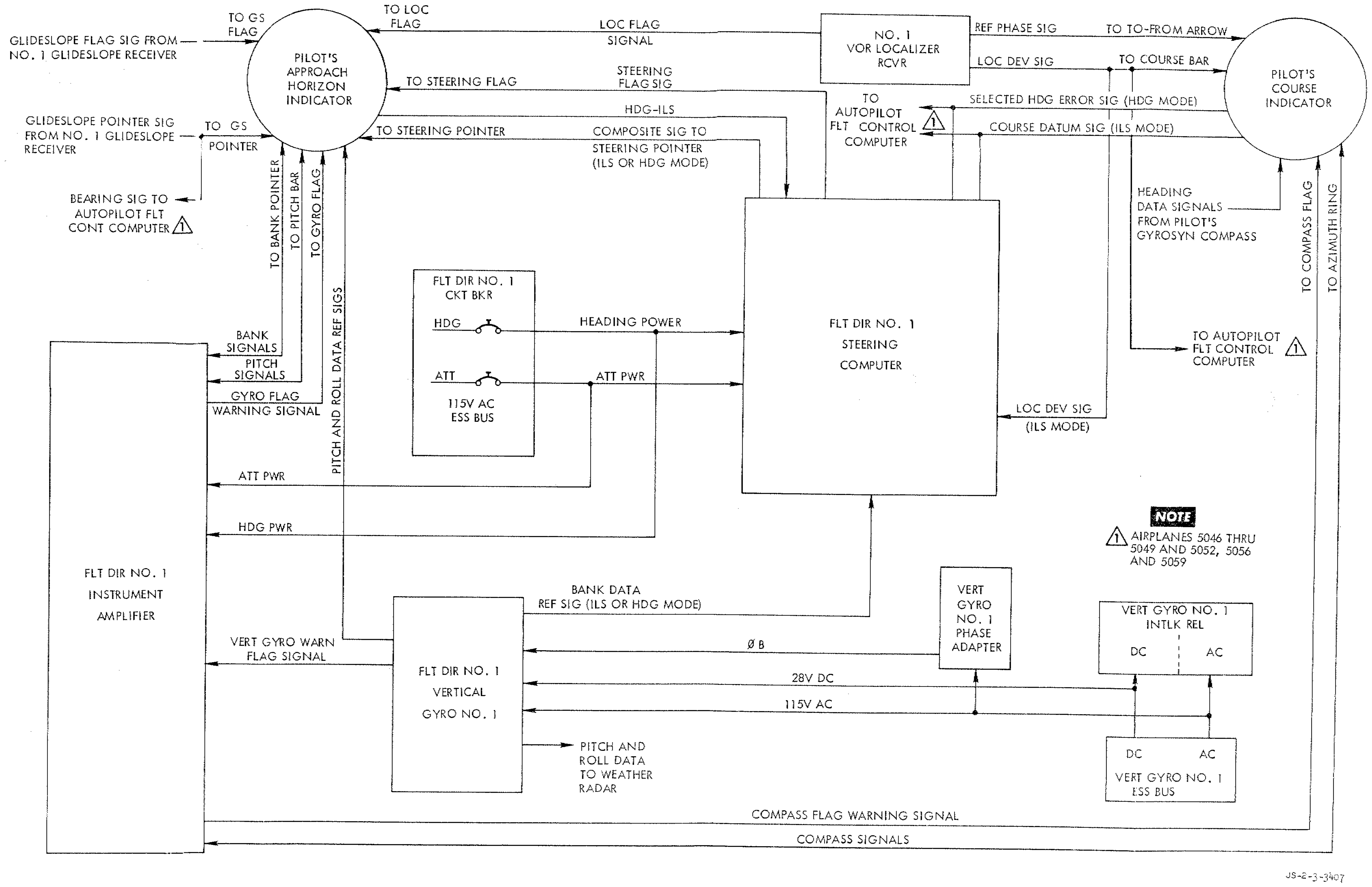
Figure 34-6. Flight Director Systems Components

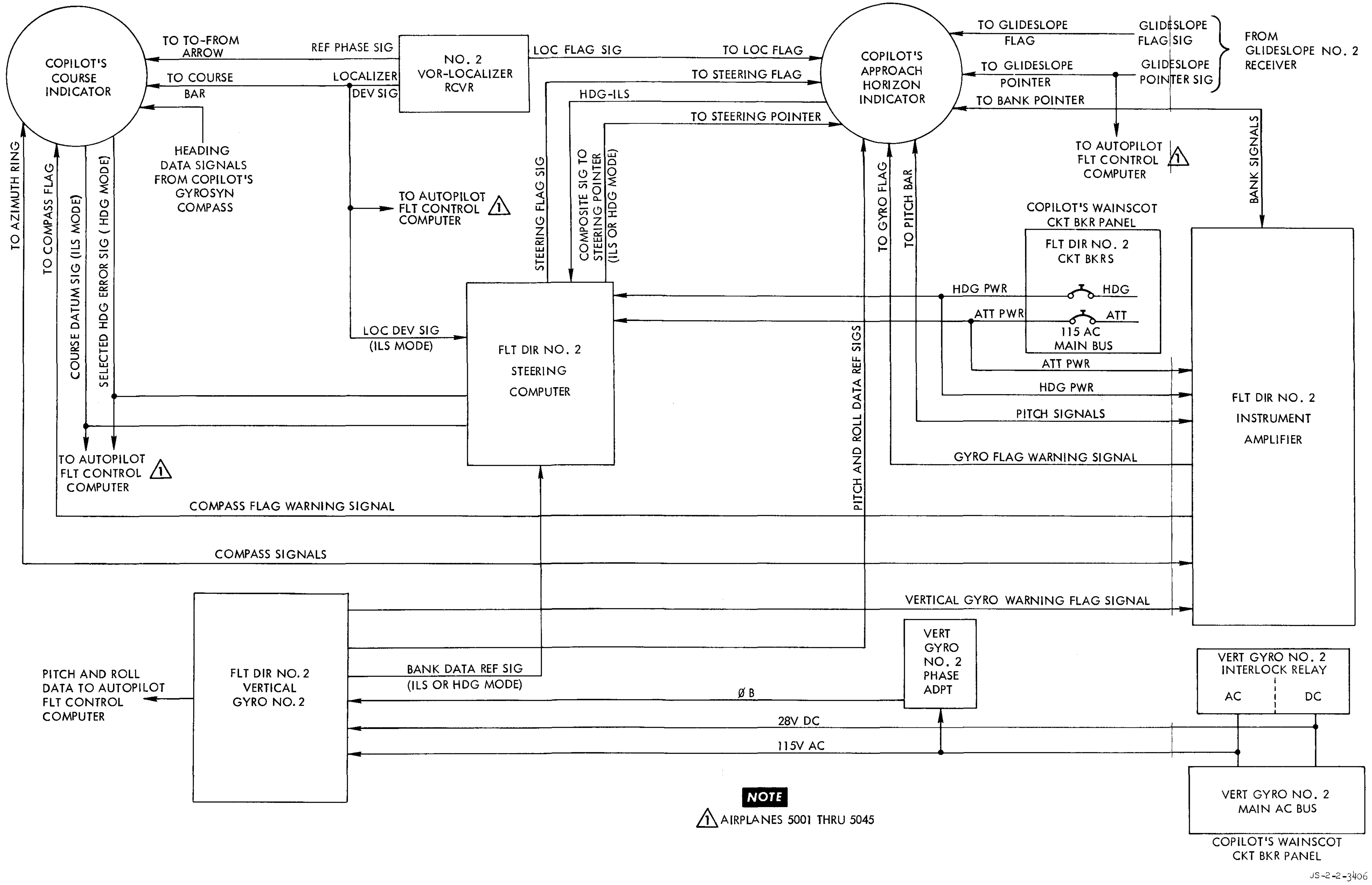

GLIDESLOPE FLAG SIG FROM NO. 1 GLIDESLOPE RECEIVER
TO GS FLAG
TO LOC FLAG
LOC FLAG SIGNAL
NO. 1 VOR LOCALIZER RCVR
REF PHASE SIG
TO TO-FROM ARROW
LOC DEV SIG
TO COURSE BAR
PILOT'S COURSE INDICATOR
PILOT'S APPROACH HORIZON INDICATOR
TO STEERING FLAG
STEERING FLAG SIG
HDG-ILS
TO AUTOPILOT FLT CONTROL COMPUTER 1
SELECTED HDG ERROR SIG (HDG MODE)
COURSE DATUM SIG (ILS MODE)
GLIDESLOPE POINTER SIG FROM NO. 1 GLIDESLOPE RECEIVER
TO GS POINTER
TO STEERING POINTER
COMPOSITE SIG TO STEERING POINTER (ILS OR HDG MODE)
BEARING SIG TO AUTOPILOT FLT CONT COMPUTER 1
TO BANK POINTER
TO PITCH BAR
TO GYRO FLAG
HEADING DATA SIGNALS FROM PILOT'S GYROSYN COMPASS
TO COMPASS FLAG
TO AZIMUTH RING
PITCH AND ROLL DATA REF SIGS
FLT DIR NO. 1 CKT BKR
HDG
ATT
115V AC ESS BUS
HEADING POWER
ATT PWR
FLT DIR NO. 1 STEERING COMPUTER
TO AUTOPILOT FLT CONTROL COMPUTER 1
BANK SIGNALS
PITCH SIGNALS
GYRO FLAG WARNING SIGNAL
LOC DEV SIG (ILS MODE)
ATT PWR
HDG PWR
FLT DIR NO. 1 INSTRUMENT AMPLIFIER
BANK DATA REF SIG (ILS OR HDG MODE)
VERT GYRO NO. 1 PHASE ADAPTER
VERT GYRO NO. 1 INTLK REL
DC
AC
VERT GYRO WARN FLAG SIGNAL
FLT DIR NO. 1 VERTICAL GYRO NO. 1
ØB
28V DC
115V AC
PITCH AND ROLL DATA TO WEATHER RADAR
DC
AC
VERT GYRO NO. 1 ESS BUS
COMPASS FLAG WARNING SIGNAL
COMPASS SIGNALS

NOTE

1 AIRPLANES 5046 THRU 5049 AND 5052, 5056 AND 5059

JS-2-3-3407

Figure 34-7. Flight Director System No. 1 Block Diagram


COPILOT'S COURSE INDICATOR
TO TO-FROM ARROW
REF PHASE SIG
TO COURSE BAR
LOCALIZER DEV SIG
NO. 2 VOR-LOCALIZER RCVR
LOC FLAG SIG
TO LOC FLAG
TO STEERING FLAG
HDG-ILS
TO STEERING POINTER
COPILOT'S APPROACH HORIZON INDICATOR
TO GLIDESLOPE FLAG
GLIDESLOPE FLAG SIG
TO GLIDESLOPE POINTER
GLIDESLOPE POINTER SIG
FROM GLIDESLOPE NO. 2 RECEIVER
TO BANK POINTER
TO AUTOPILOT FLT CONTROL COMPUTER 1
BANK SIGNALS
HEADING DATA SIGNALS FROM COPILOT'S GYROSYN COMPASS
TO AUTOPILOT FLT CONTROL COMPUTER 1
STEERING FLAG SIG
COMPOSITE SIG TO STEERING POINTER (ILS OR HDG MODE)
TO GYRO FLAG
TO PITCH BAR
COPILOT'S WAINSCOT CKT BKR PANEL
FLT DIR NO. 2 CKT BKRS
HDG PWR
HDG
ATT PWR
ATT
115 AC MAIN BUS
TO AZIMUTH RING
TO COMPASS FLAG
COURSE DATUM SIG (ILS MODE)
SELECTED HDG ERROR SIG (HDG MODE)
LOC DEV SIG (ILS MODE)
FLT DIR NO. 2 STEERING COMPUTER
PITCH AND ROLL DATA REF SIGS
ATT PWR
HDG PWR
PITCH SIGNALS
GYRO FLAG WARNING SIGNAL
FLT DIR NO. 2 INSTRUMENT AMPLIFIER
TO AUTOPILOT FLT CONTROL COMPUTER 1
COMPASS FLAG WARNING SIGNAL
COMPASS SIGNALS
VERTICAL GYRO WARNING FLAG SIGNAL
VERT GYRO NO. 2 PHASE ADPT
VERT GYRO NO. 2 INTERLOCK RELAY
AC
DC
PITCH AND ROLL DATA TO AUTOPILOT FLT CONTROL COMPUTER
FLT DIR NO. 2 VERTICAL GYRO NO. 2
BANK DATA REF SIG (ILS OR HDG MODE)
Ø B
28V DC
115V AC
VERT GYRO NO. 2 MAIN AC BUS
COPILOT'S WAINSCOT CKT BKR PANEL
JS-2-2-3406

NOTE

1 AIRPLANES 5001 THRU 5045

Figure 34-8. Flight Director System No. 2 Block Diagram

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SECTION 34.3
GLIDESLOPE SYSTEMS

Contents Page

DESCRIPTION AND OPERATION 34-33
General 34-33
List of Components 34-34
Receivers 34-34
Glideslope Antenna 34-34

OPERATIONAL CHECKOUT 34-34
Test Equipment Required 34-34
Glideslope System 34-34

TROUBLESHOOTING 34-35

REMOVAL AND REPLACEMENT 34-36
Glideslope Antenna 34-36

FIGURES
34-9. Glideslope Receiver System Components 34-37
34-10. Glideslope Receiver System Block Diagram 34-37

34.3.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.3.1.1 GENERAL. (See figure 34-9.) Two Collins 51V-3 Glideslope Receiver Systems are used with two VOR-localizer systems. During an instrument landing system (ILS) approach, the proper descent can be made with the use of the glideslope receiver system. When either of the two VOR-localizer systems is energized, a corresponding glideslope receiver is armed. Arming of the No. 1 glideslope receiver is controlled by the VOR, ON-OFF VOL control knob, on the VOR TAC-VHF control panel on the pilot's side of the center console. The No. 2 glideslope is controlled by the VOR, ON-OFF VOL control knob, on the VOR/VHF control panel, on the copilot's side of the center console. When the corresponding VOL control knob is turned on, the VOR NO. 1 or No. 2 power relay is energized, and 27.5 volts DC is fed through the closed contacts of the power relay from the essential DC bus to the No. 1 glideslope receiver, and from the main DC bus to the No. 2 glideslope receiver. GLIDESLOPE NO. 1 and NO. 2 circuit breakers are on the pilot's side and copilot's wainscot circuit breaker panels, respectively. The GLIDESLOPE receivers are switched "ON" only when a localizer/glideslope paired frequency is tuned in by the VOR/LOC receiver. Glideslope deviation signals are directly presented on the corresponding approach horizon indicator by the glideslope pointer. If the glideslope pointer sets above the horizontal centerline of the approach horizon indicator, the airplane is below the glideslope. If the pointer sets below the centerline, the airplane is above the glideslope. In case the glideslope signal is weak or falls below a predetermined safe level, a GS warning flag appears in the upper left quarter of the face of the approach horizon indicator. Also, deviation signals to the autopilot flight control computer are supplied by the No. 2 Glideslope system on airplanes 5001 thru 5045, and by the No. 1 Glideslope system on airplanes 5046 thru 5049, 5052, 5056 and 5059. A single glideslope antenna receives signals for both the glideslope receivers.

34.3.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Receiver	2	Collins	51V-3 △3	522 0671 334 △1 522 0671 294 △2
Glideslope Antenna	1	Collins	37P-4	522 0688 013

△1 Airplanes 5001, 5003, 5039 and 5046 thru 5049, 5052, 5056 and 5059.
△2 Airplanes 5002, 5004 thru 5038, 5040 thru 5045
△3 On Airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured

34.3.1.3 RECEIVERS. Glide Slope Receiver 51V-3 covers the navigation band of 329.3 to 335.0 megacycles on a 20- or 10-channel basis. The 522 0671 334 Receiver (see paragraph 34.3.1.2) covers 20 channels and the 522 0671 294 receiver covers 10 channels. The No. 1 glideslope receiver is turned on and its frequency is selected by the No. 1 VOR-localizer controls on the VOR-TAC/VHF control panel. The No. 2 glideslope receiver is controlled by the No. 2 VOR-localizer controls on the VOR-VHF panel. When a localizer station is dialed on a VOR-localizer frequency selector, the corresponding glideslope receiver frequency is also dialed.

34.3.1.4 GLIDESLOPE ANTENNA. (See figure 34-3). The glideslope systems use a Collins type 37P-4 horizontally-polarized antenna. The antenna is mounted on the forward end of the nose electronic equipment rack at fuselage station 116, and receives signals for both glideslope receivers.

34.3.2 OPERATIONAL CHECKOUT

Test Equipment Required

Nomenclature	Description	Use and Application
Collins 479-T2	Signal Generator.	Supplies directional signal to receiver.

34.3.2.1 GLIDESLOPE SYSTEM.

a. The No. 1 and No. 2 VOR-localizer and glideslope receiver systems must be installed and operational before performing this operation.

b. Connect external power to the airplane (refer to Chapter 22), and energize the AC buses.

c. Make sure the following circuit breakers are closed: VOR NO. 1 AC and DC, GLIDESLOPE NO. 1, FLT DIR NO. 1 AC and DC, and VERT GYRO NO. 1 AC and DC.

d. Set the pilot's approach horizon indicator HDG-ILS control knob to the "ILS" position.

e. Set up a Collins 479-T2 signal generator approximately 50 feet directly in front of the airplane with the test antenna connected to the GS OUTPUT terminal of the signal generator. The signal generator antenna should be positioned at right angles to the longitudinal axis of the airplane.

f. On the signal generator front panel, set the NAV-GS switch to the "GS" position, set the frequency selector at 332.0 megacycles, place the AUDIO SELECTOR in the "GLIDESLOPE" position, and adjust the MICROVOLTS control for maximum output.

g. Dial 109.3 megacycles with the No. 1 VOR-localizer frequency selector, on the VOR-TAC/VHF control panel, on the pilot's side of the center console. Turn on the VOR-localizer receiver system by turning the VOR, VOL ON-OFF control knob in a clockwise direction, to mid-travel position. Allow a 5-minute warmup for the receiver system.

h. Turn the signal generator LOC-GS control, in a clockwise direction, to point to the shaded area on the right.
RESULT: The GS flag on the pilot's approach indicator should disappear, and the glideslope pointer should set below the horizontal centerline, thus giving a "fly down" indication.

i. Turn the signal generator LOC-GS control, in a counterclockwise direction, to point to the shaded area on the left.
RESULT: The GS flag on the pilot's approach horizon indicator should remain hidden, and the

34.3.2.1 (Cont).

glideslope pointer should set above the horizontal centerline, thus giving a "fly up" indication.

j. Repeat steps (g) through (i) for the following frequencies:

VOR-localizer frequency selector	Signal generator frequency selector
110.1	334.4
110.3	335.0
110.5	329.6
110.7	320.2
110.9	330.8
109.1	331.4
109.5	332.6
109.7	333.2
109.9	333.8

For 20-channel receivers, also check the following frequencies:

108.1	334.7
108.3	334.1
108.5	329.9
108.7	330.5
108.9	329.8
111.1	331.7
111.3	332.3
111.5	332.9
111.7	333.5
111.9	331.1

k. Make sure the following circuit breakers are closed: VOR NO. 2 AC and DC, GLIDESLOPE NO. 2, FLT DIR NO. 2 AC and DC, and VERT GYRO NO. 2 AC and DC.

l. Set the copilot's approach horizon indicator HDG-ILS knob to the "ILS" position.

m. Repeat steps (f) through (j) for the No. 2 glideslope receiver system, substituting the words No. 2 VOR-localizer for No. 1 VOR-localizer, VOR/VHF in place of VOR-TAC/VHF control panel, and copilot's for pilot's in all instances.

n. Disconnect external power, return all switches and circuit breakers to the de-energized position, and restore all other equipment to normal.

34.3.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

Glideslope Manuals, Collins part numbers 523 0736 006, 523 0736 004, 523 0736 003.

PROBABLE CAUSE	ISOLATION PROCEDURE
VOR NAVIGATION RECEIVER SYSTEM OPERATING PROPERLY, AND SET TO LOCALIZER FREQUENCY, BUT NO VISUAL INDICATION FROM GLIDESLOPE RECEIVER.	
Glideslope circuit breaker open.	Check circuit breakers for correct setting.
Defective crystal.	Select other channels and check for normal operation.
Defective receiver.	Substitute a new receiver and check for proper operation.
Loose or damaged coaxial cable.	Check coaxial cable for loose connectors or damage.
Defective antenna.	Substitute a new antenna and check for proper operation.
BOTH GLIDESLOPE AND VOR INOPERATIVE.	
Defective VOR power relay.	Check relay for proper operation.
Defective VOR-TAC/VHF or VOR/VHF control panel	Substitute a new control panel and check for proper operation.

34.3.4 REMOVAL AND REPLACEMENT.

34.3.4.1 GLIDESLOPE ANTENNA. (See figure 34-3.)

a. Open the nose cone (refer to paragraph 34.0.2.1).

b. Disconnect the antenna cabling connector from the glideslope receiver antenna.

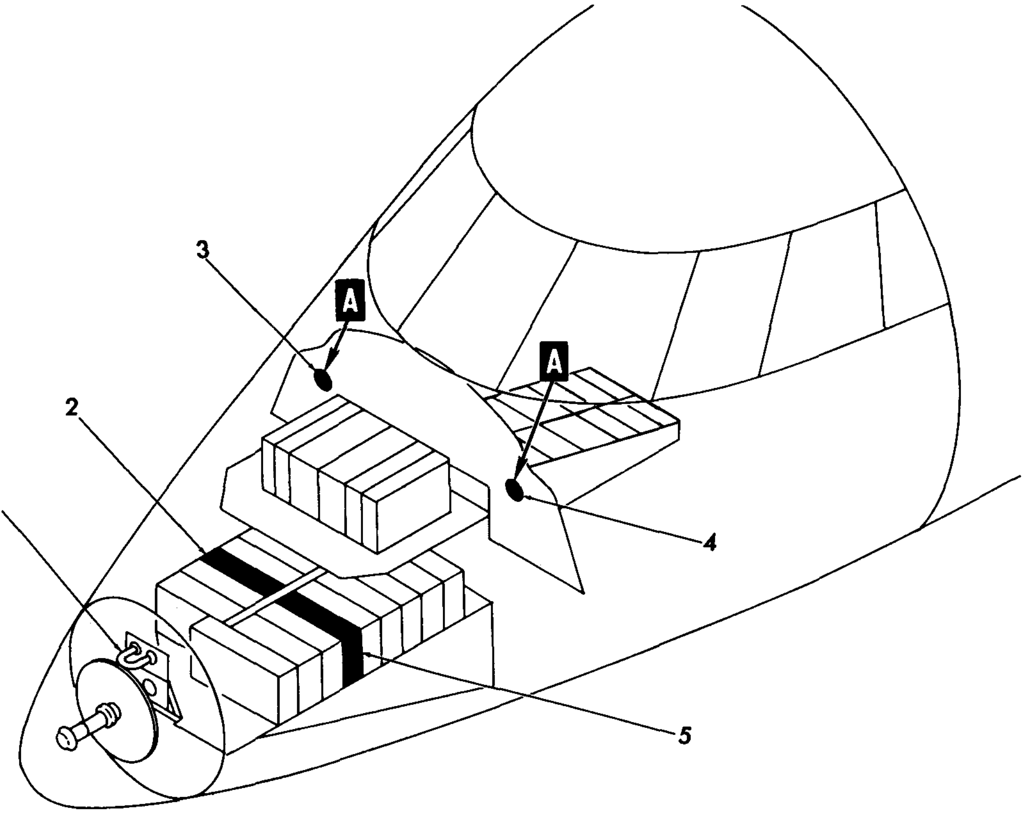
c. At fuselage station 116, remove the six nuts from the six screws which secure the glideslope antenna to the antenna mounting installation.

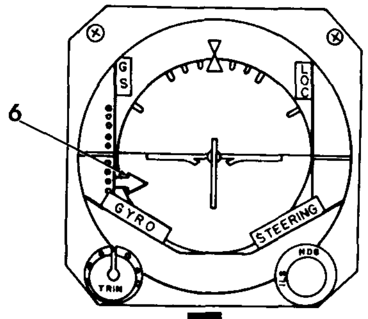
d. Remove the six screws from the mounting flange of the glideslope antenna mounting installation, and remove the glideslope antenna.

e. To install the glideslope antenna, reverse the procedures of steps (a) through (d).

f. Perform an operational checkout of the glideslope receiver system.

g. Close the nose cone.





A

APPROACH HORIZON (TYPICAL)
(LOOKING FORWARD)

1. GLIDESLOPE ANTENNA
2. NO. 2 GLIDESLOPE RECEIVER
3. COPILOT'S APPROACH HORIZON
4. PILOT'S APPROACH HORIZON
5. NO. 1 GLIDESLOPE RECEIVER
6. GLIDESLOPE POINTER

JS-2-2-3410

Figure 34-9. Glideslope Receiver System Components

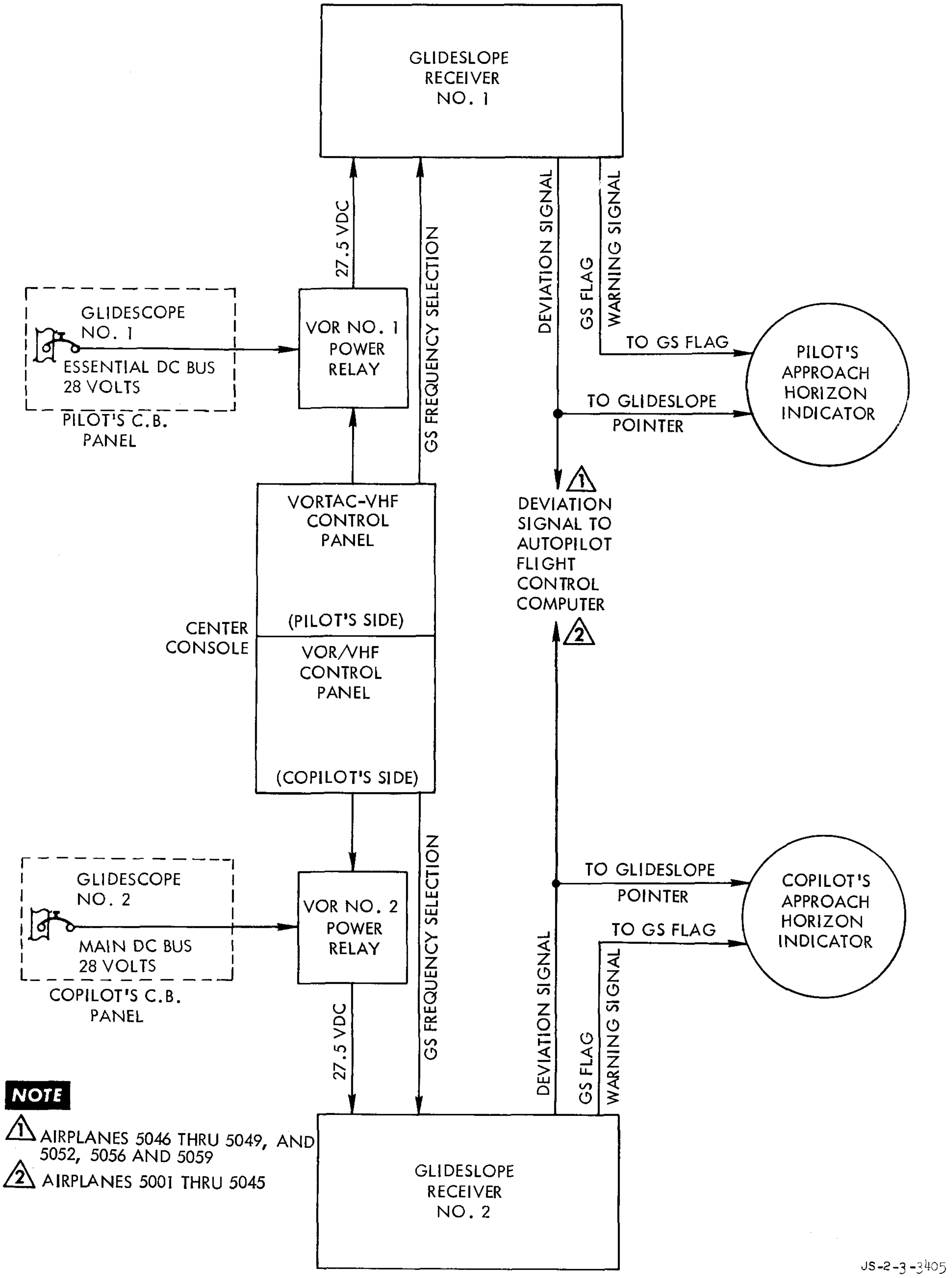


Figure 34-10. Glideslope Receiver System Block Diagram

SECTION 34.4
MARKER BEACON SYSTEM

Contents — Page

DESCRIPTION AND OPERATION 34-39
- General 34-39
- List of Components 34-40
- Receiver 34-40
- Control Panel 34-40
- Antenna 34-40

OPERATIONAL CHECKOUT 34-40
- Test Equipment Required 34-40
- Marker Beacon System 34-40

TROUBLESHOOTING 34-41

REMOVAL AND REPLACEMENT 34-42
- Marker Beacon Antenna 34-42

FIGURES
- 34-11 Marker Beacon Receiver System Components 34-43
- 34-12 Marker Beacon Receiver System Block Diagram 34-44

34.4.1 DESCRIPTION AND OPERATION. (See figures 34-11 and 34-12.)

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.4.1.1 GENERAL. A Collins 51Z-2, or 51Z-3 Marker Beacon Receiver System is used to verify the position of the airplane relative to the end of the runway when an instrument landing system (ILS) approach operation is performed. Approaching on course, the airplane passes over a series of three marker beacon transmitters which direct vertical signals only. All three transmitters operate on a frequency of 75 megacycles with amplitude-modulated signals at 400, 1,300, and 3,000 cycles per second, to distinguish between the three stations. When the airplane is almost directly above the beacon, the signal is received by the marker beacon receiver. When one of the marker beacon signals is received, a corresponding marker beacon light atop each side of the main instrument panel goes on, and a tone can be heard over the headsets. A blue marker light corresponds to the outer marker beacon (400 cps), an amber marker light corresponds to the inner marker beacon (1,300 cps), and a white marker light corresponds to the airway marker beacon (3,000 cps). The marker beacon receiver system includes a control panel, a receiver, an antenna, and two sets of marker beacon lights. Main DC bus power at 28 volts is supplied to the system through a MARKER BEACON circuit breaker on the copilot's wainscot circuit breaker panel.

34.4.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Receiver	1	Collins	51Z-3	522 1744 00 △1 △3
Receiver	1	Collins	51Z-2	522 0592 074 △2
ATC Transp - MKR Control Panel	1	Gables		G 907 △3
Marker Beacon Antenna	1	Collins	37X-2	522 0854 003
LAMP (BLUE)	2	-	-	VM911-10A
LAMP (AMBER)	2	-	-	MS25331-4
LAMP (WHITE)	2	-	-	MS25331-1

△1 Airplanes 5003, 5046 thru 5049, 5056 and 5059.
△2 Airplanes 5001, 5002, and 5004 thru 5045.
△3 On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

34.4.1.3 RECEIVER. The marker beacon receiver system uses a Collins 51Z-2 or 51Z-3 type receiver (see para. 34.4.1.2) for the reception of marker beacon signals. The 51Z-2 receiver chassis contains a DC power supply module, an RF and IF module, and a three-lamp module. The three-lamp module provides for visual and/or aural identification for each characteristic tone modulating frequency. The 51Z-3 is a completely transistorized miniature 75 mc marker receiver and is completely interchangeable electrically and mechanically with the 51Z-2 receiver.

34.4.1.4 CONTROL PANEL. The marker beacon receiver system is controlled from an ATC TRANSP-MKR control panel. A three-position ("OFF," "LO," "HI") MKR switch turns on the equipment and permits the selection of either high or low sensitivity; a VOL control enables the adjusting of the aural output of the receiver.

34.4.1.5 ANTENNA. The marker beacon receiver system uses a Collins Type 37X-2 horizontally-polarized antenna. The antenna is mounted on the airplane underside at fuselage station 545. and receives signals transmitted from the marker beacon transmitters.

34.4.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Dare MBG-1 or equal	Signal Generator	Supplies 75 megacycle frequency signals to receiver.

34.4.2.1 MARKER BEACON SYSTEM.

a. Connect external power to the airplane (see Chapter 24).

b. Make sure the following circuit breakers are closed; MARKER BEACON, INPH, and COPILOT ISOLATION AMP.

c. Make sure one of the headsets is connected, and the MKR monitor switch on the corresponding AUDIO SELECTOR panel is turned on.

d. Place the ATC TRANSP-MKR control panel MKR switch in the "LO" position, and turn the VOL control to the mid-position. Allow a 3-minute warmup.

34.4.2.1 (Cont).

e. Set up a MBG-1 signal generator approximately 8 feet from the airplane marker beacon antenna. Extend the signal generator antenna completely, and place it parallel to the fore-and-aft axis of the airplane.

f. Turn on the signal generator according to the instructions on the case lid. Using a carrier frequency of 75 megacycles, place the MODULATION switch at 400 CPS.
RESULT: The blue (outer) marker light should go on, and a 400-CPS tone should be heard over the headset.

g. Place the signal generator MODULATION switch at 1,300 CPS.
RESULT: The amber (inner) marker light should go on, and a 1,300-CPS tone should be heard over the headset.

h. Place the signal generator MODULATION switch at 3,000-CPS.
RESULT: The white (airway) marker light should go on, and a 3,000-CPS tone should be heard over the headset.

i. Move the signal generator to approximately 15 feet from the airplane marker beacon antenna. Leave the signal generator antenna parallel to the fore-and-aft axis of the airplane.

j. Place the ATC TRANSP-MKR control panel MKR switch in the "HI" position.
RESULT: The white (airway) marker light should remain on, and a 3,000-CPS tone should still be heard over the headset.

k. Place the signal generator MODULATION switch at 1,300 CPS.
RESULT: The amber (inner) marker light should go on, and a 1300-CPS tone should be heard over the headset.

l. Place the signal generator MODULATION switch at 400 CPS.
RESULT: The blue (outer) marker light should go on, and a 400-CPS tone should be heard over the headset.

m. Turn off all the equipment, and disconnect external power from the airplane.

34.4.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

Marker Beacon Manuals, Collins part number (51Z-2 Receiver) 520 5543 006

Marker Beacon Manuals, Collins part number (51Z-3 Receiver) 523 0092 006, 523 0092 004, 523 0092 003

No manuals are available for the Gables control panels. These should be returned to the manufacturer for repair.

PROBABLE CAUSE	ISOLATION PROCEDURE
NO AUDIO OR BACKGROUND NOISE IN HEADSET.	
Receiver volume control set too low.	Attempt adjustment of volume control screw on receiver.
Defective interphone system.	Perform an operational checkout on the interphone system.
INDICATOR LIGHTS DO NOT GO ON WHEN OVER MARKER BEACON, WHEN USING SIGNAL GENERATOR OR WHEN LIGHTS ARE PRESSED-TO-TEST.	
MARKER BEACON circuit breaker open.	Check circuit breakers for correct setting.
Defective indicator light.	Substitute a new indicator light bulb, and check operation.
NO AUDIO OR BACKGROUND NOISE IN HEADSET, INDICATOR LIGHTS DO NOT GO ON WHEN OVER MARKER BEACON OR USING SIGNAL GENERATOR, MARKER BEACON LIGHTS GO ON WHEN PRESSED-TO-TEST.	
Defective receiver.	Perform operation checkout of the marker beacon receiver system.

34.4.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
NO AUDIO OR BACKGROUND NOISE IN HEADSET, INDICATOR LIGHTS DO NOT GO ON WHEN OVER MARKER BEACON OR USING SIGNAL GENERATOR, MARKER BEACON LIGHTS GO ON WHEN PRESSED-TO-TEST (Cont).	
Defective antenna.	Substitute a new antenna and check for proper operation.
LOW VOLUME LEVEL.	
Defective HI-LO switch.	Switch HI-LO sensitivity switch back and forth several times while listening to audio tone.

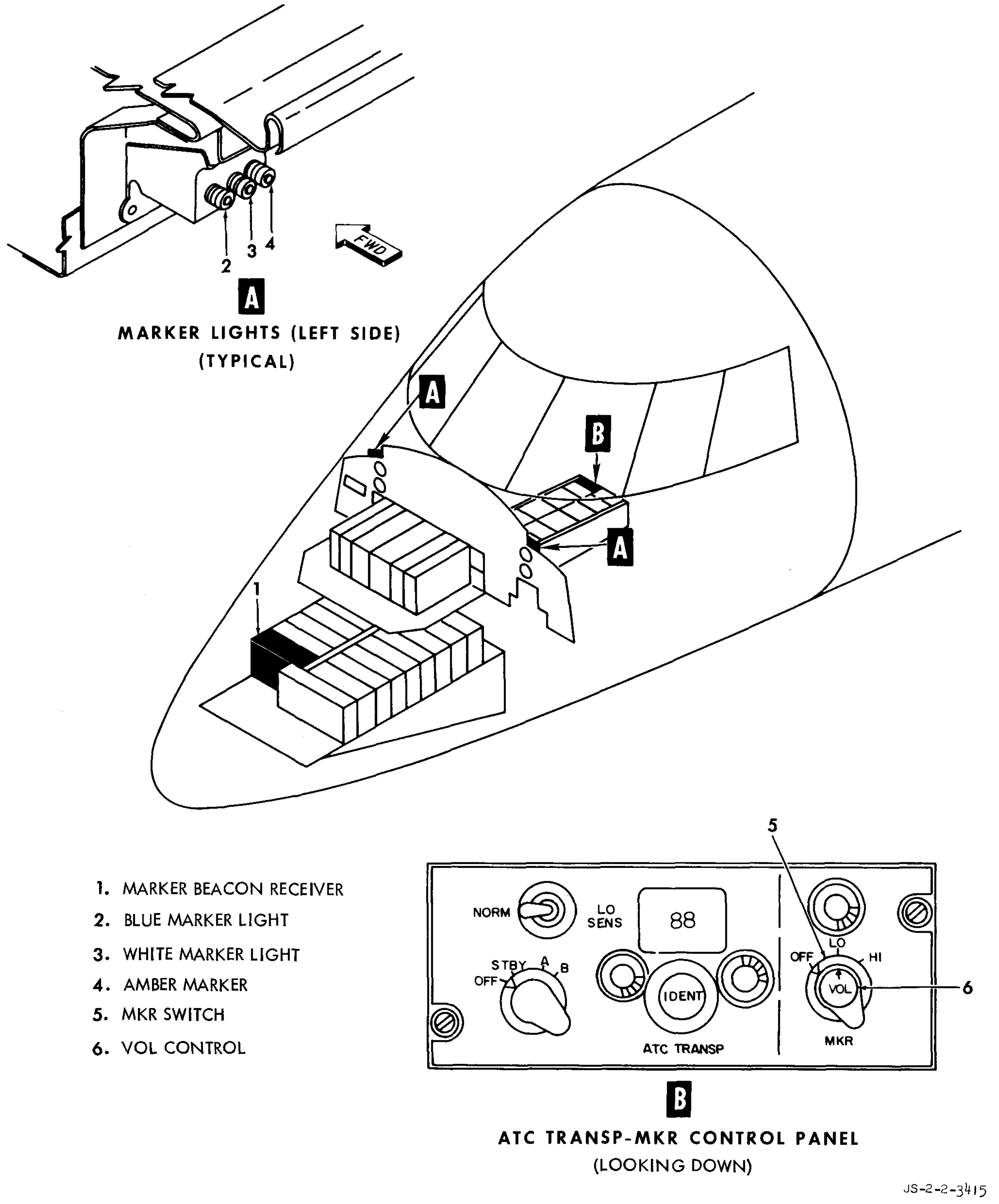
34.4.4 REMOVAL AND REPLACEMENT.

34.4.4.1 MARKER BEACON ANTENNA.

a. At fuselage station 545 remove the eight screws and washers from around the base flange of the marker beacon antenna.

b. Lower the antenna far enough from the fuselage to reach the antenna cabling connector. Disconnect the antenna cabling connector, and remove the antenna from the airplane.

c. To replace the marker beacon antenna, reverse the procedures of steps (a) and (b) and perform an operational checkout of the system.



1. MARKER BEACON RECEIVER
2. BLUE MARKER LIGHT
3. WHITE MARKER LIGHT
4. AMBER MARKER
5. MKR SWITCH
6. VOL CONTROL

Figure 34-11. Marker Beacon Receiver System Components

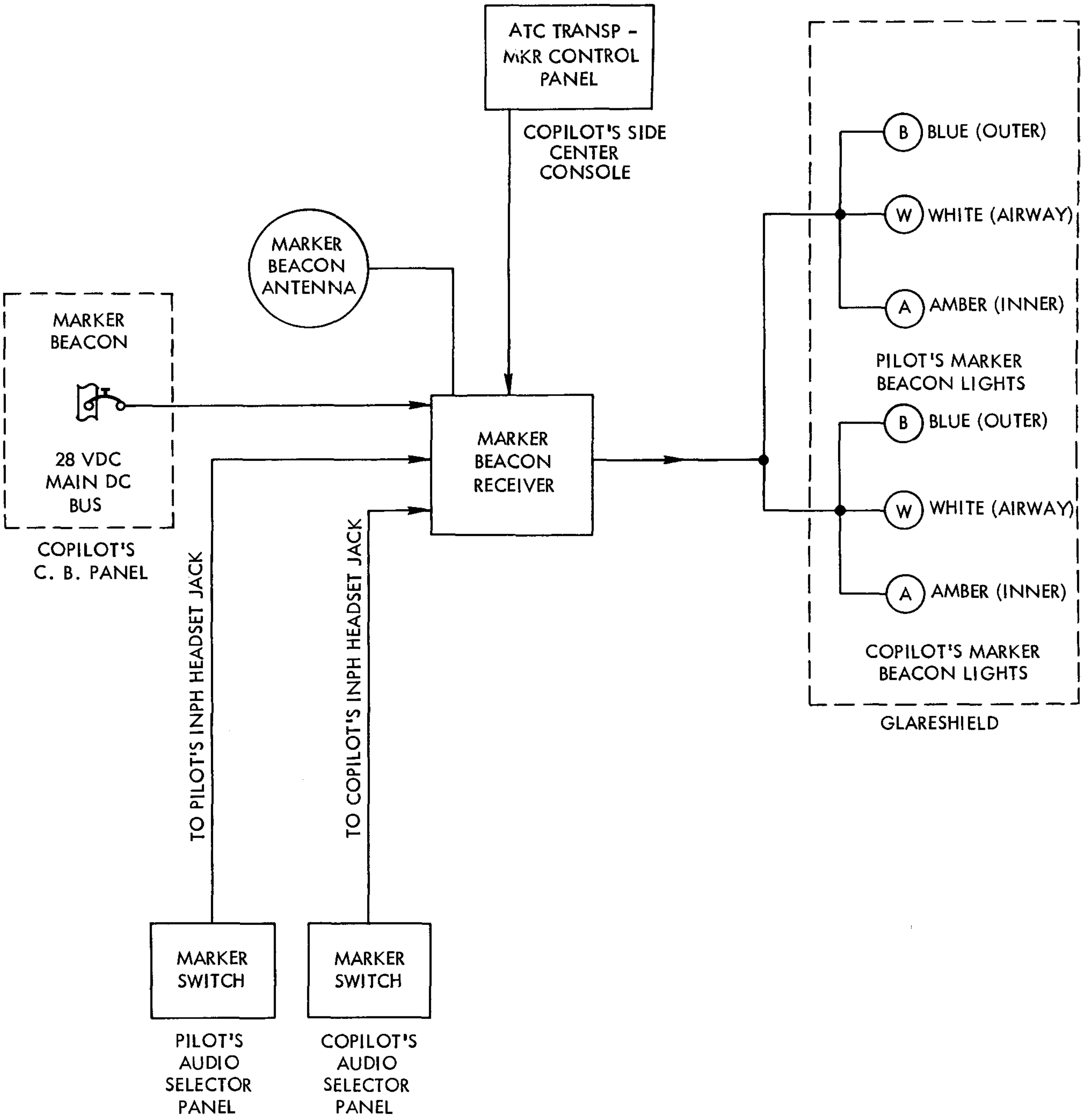

ATC TRANSP - MKR CONTROL PANEL
COPILOT'S SIDE CENTER CONSOLE
MARKER BEACON ANTENNA
MARKER BEACON
28 VDC MAIN DC BUS
COPILOT'S C. B. PANEL
MARKER BEACON RECEIVER
B BLUE (OUTER)
W WHITE (AIRWAY)
A AMBER (INNER)
PILOT'S MARKER BEACON LIGHTS
B BLUE (OUTER)
W WHITE (AIRWAY)
A AMBER (INNER)
COPILOT'S MARKER BEACON LIGHTS
GLARESHIELD
TO PILOT'S INPH HEADSET JACK
TO COPILOT'S INPH HEADSET JACK
MARKER SWITCH
PILOT'S AUDIO SELECTOR PANEL
MARKER SWITCH
COPILOT'S AUDIO SELECTOR PANEL
JS-2-2-3408

Figure 34-12. Marker Beacon Receiver System Block Diagram

SECTION 34.5
AUTOMATIC DIRECTION FINDER

Contents Page

DESCRIPTION AND OPERATION 34-45
- General 34-45
- List of Components 34-46
- Control Panels 34-46
- Receivers 34-47
- Range Filters 34-47
- Antennas 34-47
- Sense Antenna Coupler 34-47
- Line Simulator 34-47
- Quadrantal Error Correctors 34-47

OPERATIONAL CHECKOUT 34-47

TROUBLESHOOTING 34-48

REMOVAL AND REPLACEMENT 34-49
- Sense Antenna 34-49
- Loop Antenna 34-49

FIGURES
- 34-13. Automatic Direction Finder Systems Components 34-51
- 34-14. Automatic Direction Finder Systems Block Diagram 34-52

34.5.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.5.1.1 GENERAL. (See figures 34-13 and 34-14.) Operating in the frequency range of 90 to 1,800 kilocycles, two Collins DF-202 automatic direction finder systems primarily use broadcast and range station signals for navigational purposes. Each of the two independent systems includes: a Collins 51Y-3 receiver, a range filter, a control panel, a loop antenna, and a sense antenna. AC power at 115 volts and at 26 volts, and DC power at 28 volts are supplied to both systems. Circuit breakers for both the No. 1 and the No. 2 systems are on the copilot's wainscot circuit breaker panel. Each system is controlled by a separate automatic direction finder control panel on the center console. An automatic direction finder system can be used to determine automatically the bearing to a station, and to show this information on a gyrosyn compass indicator (see Section 34.6). If desired, the location of the station can be found by manually positioning the loop antenna. When flying on course on a range station, a steady tone is routed to the AUDIO SELECTOR panels. Deviating off course interrupts the signal. This interruption will identify to which side the airplane is deviating. If desired, the ADF audio can be heard over the cabin speakers.

34.5.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Receiver	2	Collins	51Y-3 △1	522 0769 024
ADF Control Panel	2	Collins	614L-6 △1	522 0928 034
Range Filter	2	Bendix		MR16D
Antenna Coupler	2	Collins	179J-1A △1	522 1970 004
Loop Antenna	2	Collins	137A-2 △1	522 0771 005
Sense Antenna (Right Side)	1	Lockheed		JR303-2R
Sense Antenna (Left Side)	1	Lockheed		JR303-2L
Line Simulator	2	Collins	316A-25 △1	522 1040 005
Loop Cable	2	Collins		426 1718 00 △2 522 3224 003 △3
Quadrantal Error Corrector (Large)	2	Collins		542 0119 003
Quadrantal Error Corrector (Medium)	2	Collins		542 4871 003

△1 On aircraft 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

△2 Aircraft 5001 thru 5045.

△3 Aircraft 5046 thru 5049, 5052, 5056 and 5059, and preferred spare.

34.5.1.3 CONTROL PANELS. An automatic direction finder control panel on the left side of the center console controls the No. 1 system, and an identical panel on the right side controls the No. 2 system. A four-position ("ADF," "ANT," "LOOP," "OFF") function selector turns on and off the corresponding automatic direction finder system and controls its mode of operation. With the function selector in the "ADF" position, the system operates automatically to find the null of the received signal. This in turn gives the station bearing presented on a gyrosyn compass indicator. If a strong enough signal is not available, the loop antenna continues to rotate. In such cases, manual tuning may provide a bearing. With the function selector in the "LOOP" position, and with a SELECT switch in the "SHARP" position, the null is found by manually rotating the loop antenna with a LOOP control until no signal is heard through the AUDIO SELECTOR panels. The "ANT" position of the function selector is used for reception of broadcasts and for flying radio ranges. The two-position ("SHARP," "BROAD") SELECT switch controls the selectivity of the receiver. If adjacent station interference or static precipitation is not a problem, the SELECT switch is placed in the "BROAD" position for maximum intelligibility. When interference occurs, it can be minimized by placing the SELECT switch in the "SHARP" position. A two-position ("ON," "OFF") BFO switch controls the beat frequency oscillator, which converts the signal from a CW station to an audible tone. This permits a more sensitive selection of frequency and the tuning of stations at distances at which the broadcasts are too weak to be otherwise useful. Frequency is dialed with three knobs and is shown on an adjacent selected-frequency indicator. As a station is tuned, a tuning meter has a maximum clockwise deflection at the station carrier frequency range. When the function selector is in the "ANT" or "LOOP" position, a GAIN control regulates the sensitivity of the receiver. With the function selector in the "ADF" position, the GAIN control regulates the audio output.

34.5.1.4 RECEIVERS. Each automatic direction finder system has a Collins Type 51Y-3 receiver, of modular construction. Capable of receiving range and broadcast stations in the range of 90 to 1,800 kilocycles, the receiver is tuned remotely by frequency selector knobs on the automatic direction finder control panel.

34.5.1.5 RANGE FILTERS. Voice signals or 1,020-CPS tone signals can be eliminated from the receiver output through a range filter. With the ADF RANGE switch on an AUDIO SELECTOR panel turned on, only 1,020-CPS signals are heard (see Chapter 23). With the ADF VOICE switch in the up position, all frequencies are passed except 1,020-CPS. If both the ADF VOICE and the ADF RANGE switches are turned on, all signals are passed.

34.5.1.6 ANTENNAS. (See figure 34-3.) Radio signals received by a Collins 137A-2 loop antenna are compared to those received by a sense antenna when the function selector is in the "ADF" position. Any error signal is translated into power to the motor which rotates the loop antenna until a null position is reached. Here the error signal is zero, and loop rotation stops. When the function selector is in the "LOOP" position, the null is found by manually controlling the rotation of the loop antenna while listening for the signal to cease. The loop antenna is not used at all when the function selector is in the "ANT" position. Then the sense antenna is used simply for radio signal reception for audio purposes.

34.5.1.7 SENSE ANTENNA COUPLER. Sense antenna coupler 179J-1A is a broad band transformer unit that transforms the capacitance of a 195-mmf antenna to 1,750mmf to complete a high-capacity sense antenna circuit.

34.5.1.8 LINE SIMULATOR. Line simulator 316A-25 contains a series of lumped constant circuits that are equivalent to definite lengths of RG-11/U coaxial cable. The simulator is designed to simulate 25 feet of equivalent cable length, so that the total high-capacity system sense antenna cable plus line simulator length can be adjusted to 60 ±2.5 feet.

34.5.1.9 QUADRANTAL ERROR CORRECTORS. Two types of quadrantal error correctors are used to produce the desired quadrantal error compensation for the loop antenna. The large corrector, Collins Part No. 542-0119-003, and small corrector, Collins Part No. 542-4871-003 consist of crescent-shaped ferrite plates that are encased in a foam potting material. Metallic bushings are moulded into the corrector assembly for mounting the correctors to the loop antenna housing. The No. 1 ADF loop antenna is compensated for 16 degrees error by placing two large ferrite corrector bars at three scale divisions fore and aft, and the No. 2 ADF loop antenna is compensated for 9 degrees error by placing two medium ferrite bars at 13.5 scale divisions fore and aft.

34.5.2 OPERATIONAL CHECKOUT.

a. Connect external power to the airplane (see Chapter 24), and energize the AC buses.

b. Make sure the following circuit breakers are closed: ADF NO. 1 DC, ADF NO. 1 26 VAC, ADF NO. 1 115 VAC, INPH, and COPILOT ISOLATION AMP.

c. Set the ADF-VOR selector knob on the pilot's and the copilot's gyrosyn compass indicators in the ADF No. 1 and No. 2 position. Then the No. 1 pointers indicate No. 1 ADF readings, and the No. 2 pointers indicate No. 2 ADF readings.

d. Make sure the pilot's and the copilot's headsets are connected.

e. Place the No. 1 automatic direction finder function selector in the "ANT" position, and allow a 5-minute warmup.

f. Place the No. 1 automatic direction finder SELECT switch in the "BROAD" position, and the BFO switch in the "OFF" position.

g. Tune in at least five broadcasts or range stations dispersed along the frequency range of 90 to 1,800 kilocycles.

RESULT: Satisfactory reception should be heard through the headsets. Selected frequencies indicated on the No. 1 automatic direction finder control panel should be within ±3.5 kilocycles of the received station frequencies in the 90 to 900 kilocycle range, and within ±6.5 kilocycles in the 900 to 1,800 kilocycle range.

h. Place the No. 1 automatic direction finder SELECT switch in the "SHARP" position.

RESULT: The receiver should have a noticeable decrease in high frequency response.

i. Place the No. 1 automatic direction finder function selector in the "LOOP" position, and the LOOP switch counterclockwise to the "L" position.

RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should rotate counterclockwise at approximately 4 RPM, thus indicating the corresponding rotation of the No. 1 automatic direction finder loop antenna.

j. Turn the No. 1 automatic direction finder LOOP switch one step clockwise.

RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should rotate counterclockwise at approximately 2 RPM, indicating the corresponding rotation of the No. 1 loop antenna.

k. Turn the No. 1 automatic direction finder LOOP switch one step clockwise to the vertical or neutral position.

34.5.2 (Cont).

RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should remain stationary, indicating that the No. 1 loop antenna is also stationary.

l. Turn the No. 1 automatic direction finder LOOP switch one step clockwise.
RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should rotate clockwise at approximately 2 RPM, indicating the corresponding rotation of the No. 1 loop antenna.

m. Turn the No. 1 automatic direction finder LOOP switch one step clockwise to the "R" position.
RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should rotate clockwise at approximately 4 RPM, indicating the corresponding rotation of the No. 1 loop antenna.

n. Place the No. 1 automatic direction finder function selector in the "ADF" position, and again tune in five broadcast or range stations dispersed along the frequency range of 90 to 1,800 kilocycles.
RESULT: The No. 1 pointers on the pilot's and copilot's gyrosyn compass indicators should assume a definite bearing for each of the stations received.

o. With a station tuned in, set the No. 1 automatic direction finder BFO switch in the "ON" position.
RESULT: A 1,020-cycle tone should be heard through the headsets.

p. Turn the No. 1 automatic direction finder GAIN control fully clockwise, then counterclockwise.
RESULT: The receiver signal should increase, then decrease.

q. Place the No. 1 automatic direction finder function selector in the "OFF" position.

r. Repeat steps (b) through (q) for the No. 2 automatic direction finder system.

s. Disconnect external power and restore the airplane to normal.

34.5.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

ADF system manuals, Collins Part Numbers
523 0020 006, 523 0020 004, 523 0020 003.

PROBABLE CAUSE	ISOLATION PROCEDURE
RECEIVER DOES NOT OPERATE IN ANY FUNCTION.	
27.5V supply not getting to receiver.	Check circuit breaker and wiring.
Defective K701 coil in receiver.	Substitute a new receiver, and check for proper operation.
Defective switch in control panel.	Substitute a new control panel, and check for proper operation.
LOOP CONTROL HAS NO EFFECT IN ANY FUNCTION.	
115 V AC power not getting to receiver.	Check circuit breaker and wiring.
Faulty circuit in control panel.	Substitute a new control panel, and check for proper operation.
Open loop drive motor.	Substitute a new loop antenna, and check for proper operation.
INDICATED BEARING IS 180 DEGREES IN ERROR ON ALL BANDS.	
Loop to receiver r-f connections reversed.	Check r-f cable connections against wiring diagram.
Internal wiring reversed in loop or receiver.	Substitute a new loop antenna or receiver, and check for proper operation.
Loop mounted backward.	Check mounting of loop.

34.5.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
INDICATED BEARING IS 180 DEGREES IN ERROR ON ONE BAND.	
Connections reversed on loop or sense coils.	Substitute a new receiver, and check for proper operation.
INDICATED BEARING CHANGE IS IN WRONG DIRECTION AS CORRECT BEARING IS CHANGED.	
Loop to indicator synchro X and Y connections are reversed.	Check connections against wiring diagram.
INDICATOR POINTER MOVES IN WRONG DIRECTION WITH MANUAL LOOP ROTATION.	
Loop to indicator synchro X and Y connections are reversed.	Check connections against wiring diagram.
Receiver to control unit pins 6 and 8 improperly interconnected.	Check interconnections against wiring diagram.
INDICATED BEARING ERROR THE SAME ON ALL BANDS.	
Loop assembly not aligned on aircraft.	Check alignment of loop assembly with marks on fuselage.
Quadrantal error correctors broken or improperly installed.	Check for broken or improperly installed error correctors. (See paragraph 34.5.1.9)
INDICATED BEARING 180 DEGREES OUT OF ONE INDICATOR.	
Indicator excitation reversed or no excitation.	Check wiring against wiring diagram.

34.5.4 REMOVAL AND REPLACEMENT.

34.5.4.1 ADF SENSE ANTENNAS.

a. Remove the ADF sense antennas located on the underside of the aircraft at fuselage station 443 by removing 78 screws.

b. Disconnect antenna feeder cable from antenna before completely lowering antenna.

c. Remove antenna from aircraft.

d. To install ADF sense antenna, reverse above procedure.

34.5.4.2 ADF LOOP ANTENNAS.

a. Remove JF440 ADF loop fairing located on bottom centerline of aircraft at fuselage station 383 by removing 43 screws.

b. Remove the four screws which attach the loop rotor assembly to the loop frame.

c. Remove loop rotor assembly from aircraft.

Note

If complete loop assembly is to be removed, proceed with the following steps.

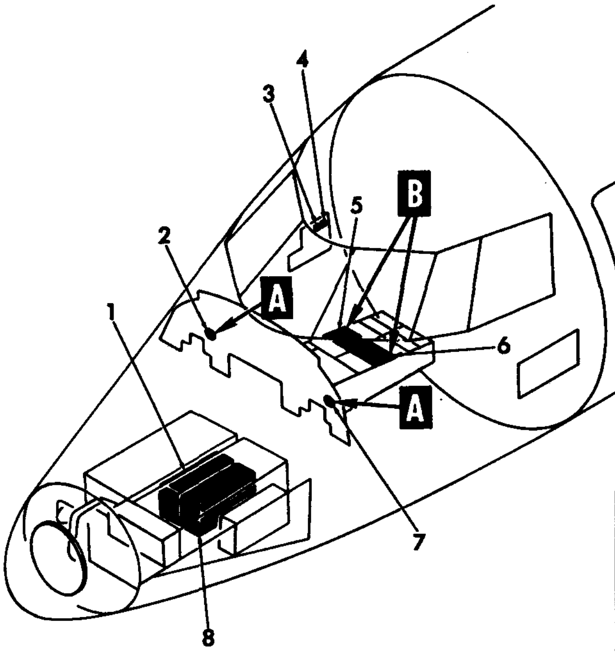
d. Remove floorboards from over loop antennas.

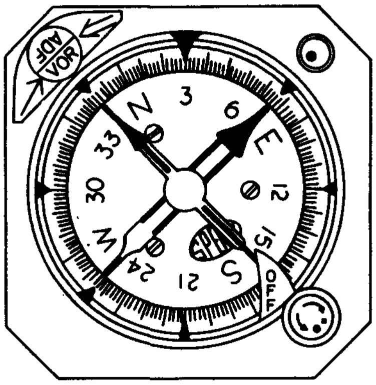
e. Disconnect electrical cables from antenna.

f. Remove 8 screws holding antenna frame in position.

g. Remove frame assembly from aircraft.

h. To install loop antennas, reverse above procedure. If complete antenna assembly is to be installed, it must be fillet sealed around cutout with Minnesota Mining and Manufacturing Co. EC-1329 Class A4 sealing compound (or equal).

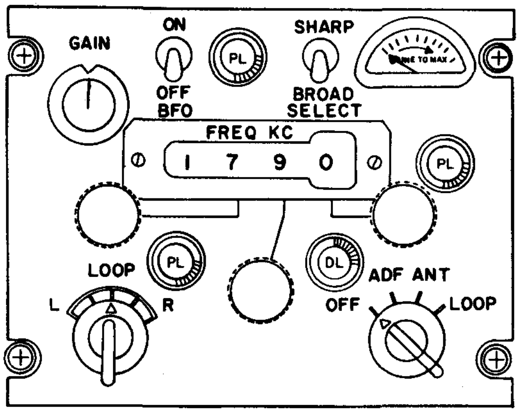




A

GYROSYN COMPASS INDICATOR (TYPICAL)

(LOOKING FORWARD)



B

AUTOMATIC DIRECTION FINDER CONTROL PANEL (TYPICAL)

1. NO. 1 AUTOMATIC DIRECTION FINDER RECEIVER
2. COPILOT'S GYROSYN COMPASS INDICATOR
3. NO. 1 RANGE FILTER
4. NO. 2 RANGE FILTER
5. NO. 2 AUTOMATIC DIRECTION FINDER CONTROL PANEL
6. NO. 1 AUTOMATIC DIRECTION FINDER CONTROL PANEL
7. PILOT'S GYROSYN COMPASS INDICATOR
8. NO. 2 AUTOMATIC DIRECTION FINDER RECEIVER

JS-2-2-3414

Figure 34-13. Automatic Direction Finder Systems Components

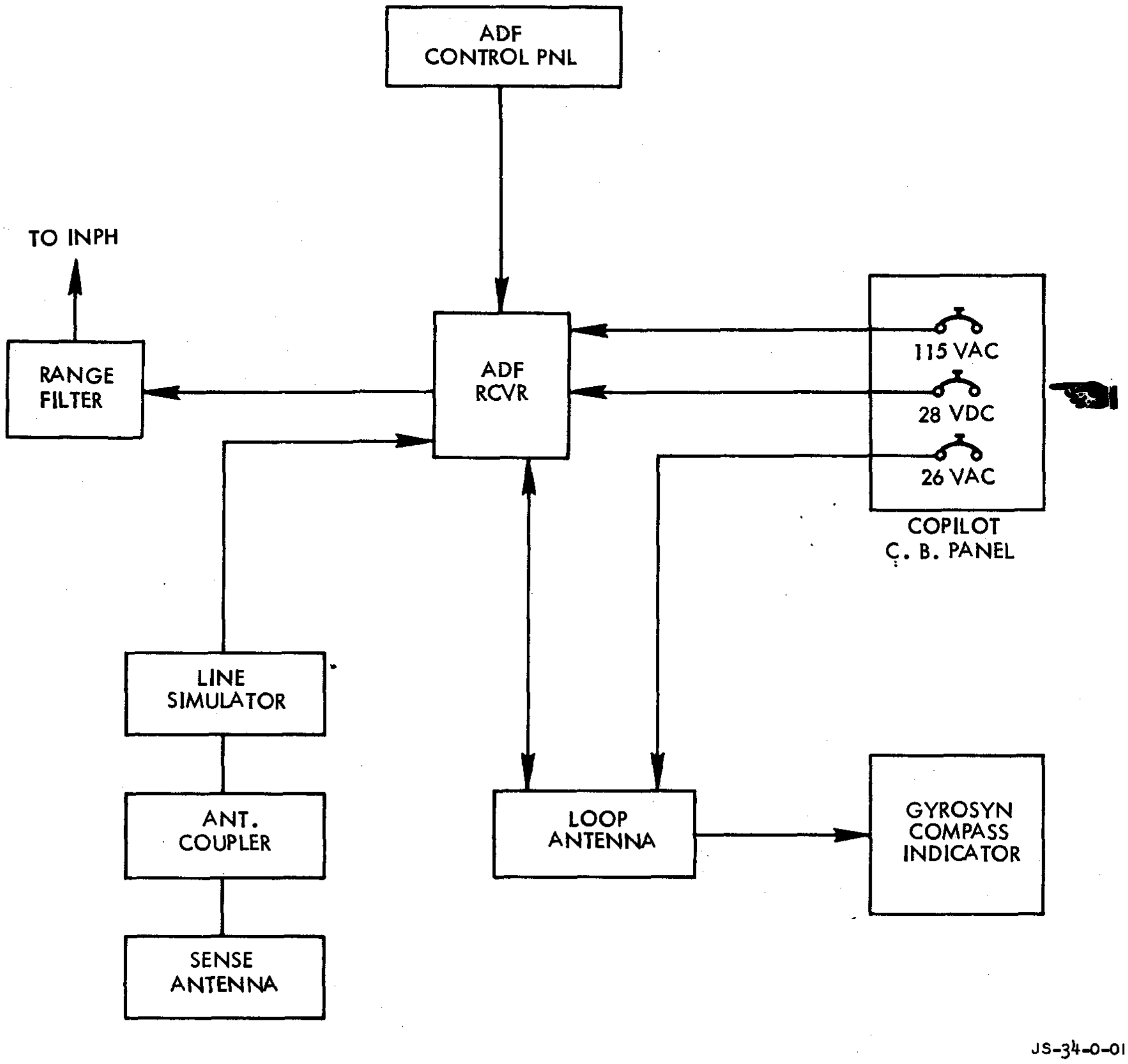


Figure 34-14. Automatic Direction Finder Systems Block Diagram

SECTION 34.6
GYROSYN COMPASS SYSTEMS

Contents Page

DESCRIPTION AND OPERATION 34-53
- General 34-53
- List of Components 34-54
- Flux Valves 34-54
- Directional Gyros 34-54
- Servoamplifiers 34-54
- Slaving Amplifiers 34-55
- Slave Compass-DG Switches 34-55
- Gyrosyn Compass Indicators 34-55

OPERATIONAL CHECKOUT 34-55

TROUBLESHOOTING 34-56

ADJUSTMENT 34-57
- Compass Swinging and Compensation 34-57
- Standby Compass Reswing 34-58

FIGURES
- 34-15. Gyrosyn Compass Systems Components 34-59
- 34-16. Gyrosyn Compass Systems Block Diagram 34-60

34.6.1 DESCRIPTION AND OPERATION.

Note

On airplanes 5046 and up, the gyrosyn compass system components are not installed by Lockheed when the airplane is manufactured. Complete space provisions for the components and wiring provisions are installed. See paragraph 34.6.1.2 for components required to make systems operative.

34.6.1.1 GENERAL. A No. 1 gyrosyn compass system is provided for the pilot, and a No. 2 system for the copilot. Each compass system consists of a flux valve, a directional gyro, a servoamplifier, a slaving amplifier, a SLAVE COMPASS-DG switch, and a gyrosyn compass indicator. Both of the amplifiers are in an equipment box called a compass rack. Either of two modes of operation can be selected with the SLAVE COMPASS-DG switch. The directional gyro can be slaved to the flux valve through a slaving torque motor, or the gyro can be run freely to hold a constant heading. Power is disconnected from both the flux valve and the slaving amplifier when the directional gyro is run freely. The pilot's gyrosyn compass indicator transmits a synchro signal to position the azimuth card of the pilot's course indicator, and the copilot's gyrosyn compass indicator is connected to the copilot's course indicator (see Section 34.2). A COMPASS NO. 1 26 VAC and a COMPASS NO. 1 115 VAC circuit breaker on the pilot's side circuit breaker panel, and a COMPASS NO. 2 26 VAC and a COMPASS NO. 2 115 VAC circuit breaker on the copilot's wainscot circuit breaker panel route power to the two systems.

34.6.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Compensator	2	Sperry		620783
Flux Valve	2	Sperry		620359
Directional Gyro	2	Sperry		613260-2 △3
Servo Amplifier	2	Sperry		618379 △3
Slaving Amplifier	2	Sperry		621107 △3
Gyrosyn Compass Indicator (red lighted)	2	Sperry	C-6F	1782993-623 △3 △4
Gyrosyn Compass Indicator (white lighted)	2	Sperry	C-6F	1782993-653 △3 △5
Slave Compass-DG Switch	2			MS24523-23
Compass System Rack	2	Sperry		614937-10 △2
Compass System Rack	1	Sperry		614937-12 △1 △3

△1 Airplanes 5003, 5011 and up.

△2 Airplanes 5001, 5002 and 5004 thru 5010.

△3 On airplanes 5046 and up, this equipment is not installed by Lockheed when the airplane is manufactured.

△4 Airplanes 5001 thru 5056 and 5058 thru 5065.

△5 Airplanes 5057, 5066 and up.

34.6.1.3 FLUX VALVES. During slaved operation, a flux valve in the left-hand wing tip provides information to the pilot's gyrosyn compass indicator as to the position of the airplane with respect to the earth's magnetic field. Similarly, the copilot's gyrosyn compass indicator is linked with the flux valve in the right-hand wing tip. Each flux valve has three-phase, wye-connected detector synchro acted on by a combination of the earth's magnetic field and the field due to 400 cycle, 23-volt, AC in an excitation winding. Dependent upon the direction of the magnetic field, the different voltages induced in each of the synchro windings constitute the position data transmitted to the gyrosyn indicator. A compensator is fitted atop each flux valve to offset the effects of the magnetic fields due to the airplane itself.

34.6.1.4 DIRECTIONAL GYROS. When one of the COMPASS 115 VAC circuit breakers is closed and power is available on the bus, a corresponding directional gyro will be running. Since the power input to the directional gyro is single-phase, an internal phase adapter is necessary for conversion to three-phase. The position of the gyro is reflected in the three-wire data transmitted by a heading data synchro to a follow-up synchro in a corresponding gyrosyn compass indicator. For the No. 2 directional gyro, another heading data synchro transmits position information to the autopilot flight control computer (see Chapter 22). During slaved operation, a slaving torque motor slaves the gyro to hold a fixed reference with respect to the direction of the earth's magnetic field as detected by the corresponding flux valve. Thus drift is overcome before any significant error occurs. During a free directional gyro operation, however, the rate of random drift is approximately (±) 2.5 degrees per hour.

Note

Extreme care must be used when installing or removing the flux valves. All tools and attaching hardware must be nonmagnetic. Care must be taken not to jar or otherwise damage the flux valves. (See Chapter 28 for requirements on nonmagnetic attaching hardware.)

34.6.1.5 SERVOAMPLIFIERS. In the case of either gyrosyn compass system, any error voltage induced in a heading data control transformer in an indicator is routed to a corresponding servoamplifier. The output of the servoamplifier then drives an azimuth card servomotor in that indicator to effectively eliminate the error voltage.

34.6.1.6 SLAVING AMPLIFIERS. Any error voltage induced in a slaving control transformer in a gyrosyn compass indicator is routed to a corresponding slaving amplifier. The output of the slaving amplifier drives the slaving torque motor in the corresponding directional gyro. Simultaneously, the polarity of the output is indicated on an annunciator of the corresponding gyrosyn compass indicator.

34.6.1.7 SLAVE COMPASS-DG SWITCHES. The mode of operation of either of the two gyrosyn compass systems is dependent upon the position of a corresponding two-position SLAVE COMPASS-DG switch. When the switch is in the "SLAVE COMPASS" position, 115-volt, AC power is connected to the slaving amplifier. The amplifier in turn furnishes 23-volt, AC excitation to the flux valve. This power is disconnected when the switch is in the "DG" position.

34.6.1.8 GYROSYN COMPASS INDICATORS. An indicator for the No. 1 gyrosyn compass system is on the pilot's side of the main instrument panel, and an indicator for the No. 2 system is on the copilot's side. Both indicators are Type C-6F gyrosyn compass indicators. On the front of each is an azimuth card, an annunciator, a synchronizer knob, an ADF/VOR selector knob, and a No. 1 and a No. 2 pointer. The azimuth card is positioned by a servomotor within the indicator according to the output signal of a related servoamplifier. Rotating along with the azimuth card are a heading data control transformer, a slaving control transformer, and a course indicator azimuth repeater synchro, all directly fixed to the same shaft. During a slaved operation, the slaving control transformer seeks proper alignment with a slaving synchro which receives position data from the corresponding flux valve. During any operation, the heading data control transformer seeks proper alignment with a follow-up synchro which receives position data from the corresponding directional gyro. When a compass system is first put into slaved operation, the gyro follow-up synchro should be adjusted to obtain a reliable compass reading without delay. This can be done by turning the synchronizer knob in the direction shown on the annunciator until the direction shown reverses. From then on the servo loop quickly makes any needed corrections. The annunciator indication, either a dot or a cross, shows the polarity of the output signal of the corresponding slaving amplifier. The ADF/VOR selector knob permits the choice of any of four combinations of radio bearing information on the two pointers. Selection is made by positioning the ADF/VOR selector knob so the single-line and the double-line arrows point to "ADF" or "VOR" on the knob as desired. The single-line arrow corresponds to pointer No. 1, and the double-line arrow to pointer No. 2. The No. 1 pointer can indicate radio bearings for either the VOR-localizer system No. 1 or the automatic direction finder system No. 1; likewise, the No. 2 pointer presents bearing information for the No. 2 systems. A warning flag appears in the lower right-hand corner of the indicator when power is lost to the corresponding servoamplifier.

34.6.2 OPERATIONAL CHECKOUT.

a. Connect external power to the airplane (see Chapter 24).

b. Be sure the COMPASS NO. 1 26 VAC and the COMPASS NO. 1 115 VAC circuit breakers are closed.

c. Place the No. 1 SLAVE COMPASS-DG switch in the "SLAVE COMPASS" position.

d. Synchronize the pilot's gyrosyn compass system, using the synchronizer knob while observing the annunciator.

Note

If the cross appears in the annunciator window, turn the synchronizer knob counterclockwise to synchronize the system. For a dot, turn clockwise.

DESIRED RESULT: The azimuth card should rotate quickly (at least 40 degrees per second) to the correct magnetic heading. Neither a cross nor a dot should be visible in the annunciator window when the correct heading is indicated.

e. Drive the compass system 5 degrees out of synchronization, using the synchronizer knob.
DESIRED RESULT: Within 5 minutes, the system should synchronize as indicated by neither the cross nor the dot appearing in the annunciator window.

f. Place the No. 1 SLAVE COMPASS-DG switch in the "DG" position; then drive the compass system approximately 25 degrees out of synchronization, using the synchronization knob.
DESIRED RESULT: After 10 minutes, the compass indication should not have drifted by an amount greater than that given by the following equation,

$$D = 0.75 + 2.5 \text{ SIN } \phi$$

where D is the number of degrees of compass drift indicated, and ϕ the degree of latitude at which the check is made. For example, at 30 degrees of latitude the indicated compass drift allowable should be computed as follows:

$$D = 0.75 + 2.5 \text{ SIN } 30^\circ$$

$$D = 2.0 \text{ degrees.}$$

Note

If a compass drift greater than the computed allowable is observed, replace the gyro. Note the difficulty in accurately reading such

34.6.2 (Cont).

small amounts of drift angle, so be sure that there is a significantly excessive drift before removing a gyro.

g. Repeat steps (b) through (f) for the No. 2 gyrosyn compass system.

h. Disconnect external power, and return the airplane to normal.

34.6.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component.

PROBABLE CAUSE	ISOLATION PROCEDURE
AZIMUTH CARD INDICATES CORRECT HEADING WITH A DOT OR CROSS VISIBLE IN ANNUNCIATOR WINDOW.	
Defective indicator	Substitute a new indicator, and check for proper operation.
NEITHER DOT NOR CROSS VISIBLE IN ANNUNCIATOR WINDOW REGARDLESS OF AZIMUTH CARD INDICATION.	
Open circuit breakers.	Check setting of gyrosyn compass circuit breakers.
Defective Slave Compass - DG switch.	Check switch for proper operation.
Defective wiring.	Check continuity of wiring from 115 VAC circuit breaker to the slaving amplifier.
	Check continuity of wiring from gyrosyn compass indicator to the slaving amplifier.
	Check continuity of wiring from the slaving amplifier to the flux valve.
Defective indicator.	Substitute a new indicator, and check for proper operation.
Defective compass rack.	Substitute a new compass rack, and check for proper operation.
AZIMUTH CARD WILL NOT ROTATE.	
Defective wiring.	Check continuity of wiring from the compass indicator to the servo amplifier.
	Check continuity of wiring from the 115 VAC circuit breakers to the directional gyro.
AZIMUTH CARD WILL NOT ROTATE OR ROTATES AT LESS THAN 40 DEGREES PER SECOND.	
Defective wiring.	Check continuity of wiring from the compass indicator to the directional gyro.
Defective compass indicator.	Substitute a new compass indicator, and check for proper operation.
Defective servo amplifier.	Substitute a new servo amplifier, and check for proper operation.
Defective directional gyro.	Substitute a new directional gyro, and check for proper operation.

34.6.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE
INDICATED MAGNETIC HEADING IS INCORRECT.	
Compass out of adjustment.	Swing the compass (see paragraph 34.6.4.1).
SYSTEM REQUIRES LONGER THAN 5 MINUTES TO SYNCHRONIZE ON 5° ERROR.	
Defective slaving amplifier.	Substitute a new slaving amplifier, and check for proper operation.
Defective directional gyro.	Substitute a new directional gyro, and check for proper operation.
Defective compass indicator.	Substitute a new compass indicator, and check for proper operation.

34.6.4 ADJUSTMENT.

34.6.4.1 COMPASS SWINGING AND COMPENSATION. On aircraft with such electronic systems as Doppler Navigation installed, it becomes particularly important to minimize compass system error. For best accuracy, compass swinging and compensation should be performed by experienced and qualified personnel in an area known to be free of local variations in the magnetic field. (Unacceptable variations are defined as differences of more than .2° in the horizontal component of the earth's magnetic field within a space 2 to 6 feet above ground surface and extending over an area 250 feet in radius.) Swinging and compensation may be accomplished either by means of an electrical tester such as the MC-1 (Sperry Phoenix) or a compass rose. When using the MC-1 (or equivalent), the manufacturer's instructions should be followed. The procedure outlined below may be followed when not using a compass rose.

a. Connect external power to the airplane (refer to Chapter 24).

b. Perform a satisfactory operational checkout of the compass systems.

Note

Personnel engaged in compensating the compasses must be sure all magnetic tools and maintenance material are removed from their persons and from the airplane. A non-magnetic screwdriver must be used for compensating.

c. Drive both azimuth cards through 360 degrees, using the synchronizer knobs.

d. Set the compensator magnets of both flux valves at the neutral positions by aligning the dots on the N-S and E-W shafts with the dots on the covers.

e. Make the first compass swing, taking readings for both gyrosyn compass systems and for the standby compass (see Section 34.10). This first swing and each subsequent one should be done according to the procedures in steps (f) through (k).

f. Align the airplane within 1.0 degree of each specified heading. Determine the actual heading of the airplane within 0.5 degree, using a transit with the airplane centerline as reference. See Chapter 51 for method of determining aircraft centerline.

g. When taking readings on the gyrosyn compass indicators, be sure neither the crosses nor the dots appear in the annunciator windows.

h. When taking readings on the standby compass, energize the following systems:

Battery
Windshield Inverter
Autopilot
VHF Communication
Flight Director Systems
Glideslope Systems
Windshield De-Icing
ATC Transponder System
Main Inverter(s)
Boost Pumps
All the Pilots' Compartment Lights
VOR-Localizer Systems
Weather Radar
Automatic Direction Finder System

34.6.4.1 (Cont).

i. Be sure all the components of the airplane which could influence the compass systems are installed properly; otherwise, errors might be introduced after compensation.

j. Record the compass reading deviations at 0 (±1), 90 (±1), 180 (±1), and 270 (±1) degrees.

k. Algebraically calculate the coefficients A, B, and C from the following formulas. Note here that N, E, S and W are deviations from the actual heading, and not the indicated headings. Deviations are considered positive when the indicated readings are less than the actual heading, and negative when greater.

$$A = 1/4\,(N+E+S+W)$$
$$B = 1/2\,(E-W)$$
$$C = 1/2\,(N-S)$$

l. If the coefficient A for either gyrosyn compass system exceeds 0.5 degree, loosen the corresponding flux valve mounting screws, and rotate the transmitters as necessary. The peripheries of the flux valve flanges should be held firmly to the guides on the surfaces of the mounting brackets while adjustments are being made.

m. If the coefficient A for the standby compass exceeds 1 degree, realign it with respect to the main instrument panel by means of washers and spacers.

n. Using the computed values of B and C, perform the compensations in steps (p) and (q).

o. While at 0 (±1) or 180 (±1) degrees, make an N-S compensation on both flux valves and on the standby compass, according to the magnitude and direction of coefficient C.

p. While at 90 (±1) or 270 (±1) degrees, make an E-W compensation on both flux valves and on the standby compass, according to the magnitude and direction of coefficient B.

q. While at 270 (±1) degrees, adjust the flux valves and the standby compass so as to change the readings according to the magnitude and direction of coefficient A.

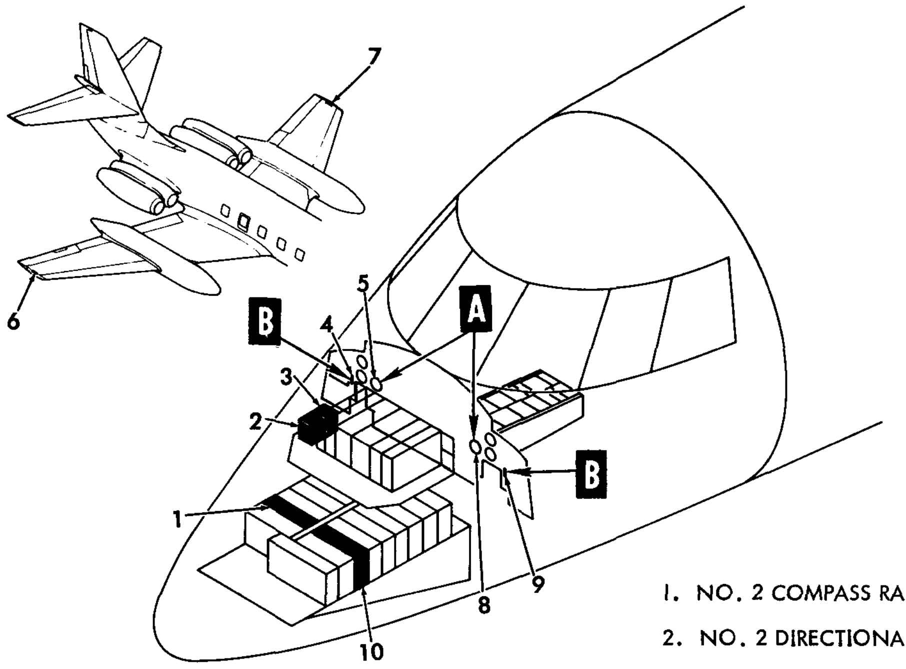
r. Make another compass swing of all cardinal headings to ensure that the compensation is satisfactory. If it is not, repeat steps (p), (q), and (r), using new values of B and C, until all of the compass systems check out.

s. Turn the airplane to eight different magnetic headings, 45 degrees apart, beginning at 0 degree. At each heading, perform steps (u) and (v).

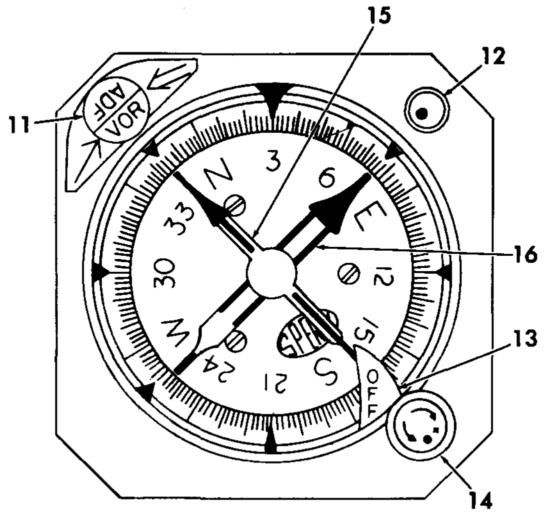
t. Make the required entries on the standby compass correction card.

u. For the gyrosyn compass indicators, be sure the indicated heading is within 1 degree of the actual heading.

34.6.4.2 STANDBY COMPASS RESWING. After the original compass swinging and compensation, the standby compass (see Section 34.10) can be compensated to the gyrosyn compass systems. This check can be done on or off a compass rose, if there are no metal objects within a 250-foot radius.

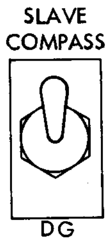


1. NO. 2 COMPASS RACK
2. NO. 2 DIRECTIONAL GYRO
3. NO. 1 DIRECTIONAL GYRO
4. NO. 2 SLAVE COMPASS-DG SWITCH
5. COPILOT'S GYROSYN COMPASS INDICATOR
6. RH FLUX VALVE (NO. 2)
7. LH FLUX VALVE (NO. 1)
8. PILOT'S GYROSYN COMPASS INDICATOR
9. NO. 1 SLAVE COMPASS-DG SWITCH
10. NO. 1 COMPASS RACK
11. ADF/VOR SELECTOR KNOB
12. ANNUNCIATOR
13. WARNING FLAG
14. SYNCHRONIZER KNOB
15. NO. 1 POINTER
16. NO. 2 POINTER



A

GYROSYN COMPASS INDICATOR (TYPICAL)
(LOOKING FORWARD)



B

SLAVE COMPASS-DG SWITCH (TYPICAL)
(LOOKING FORWARD)

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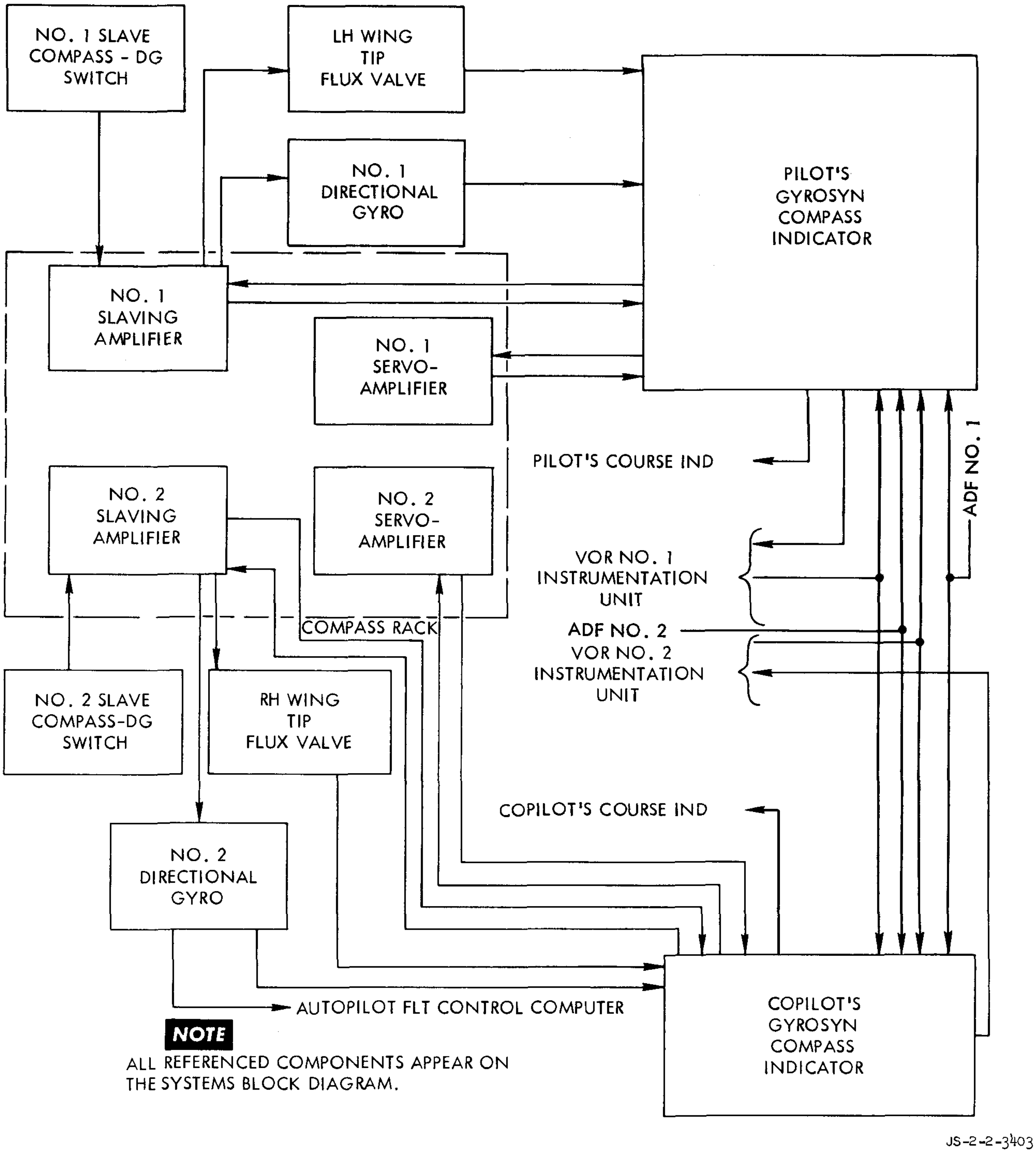
Figure 34-15. Gyrosyn Compass Systems Components



NOTE

ALL REFERENCED COMPONENTS APPEAR ON THE SYSTEMS BLOCK DIAGRAM.

Figure 34-16. Gyrosyn Compass Systems Block Diagram

SECTION 34.7
WEATHER RADAR

Contents Page

DESCRIPTION AND OPERATION 34-61
General 34-61
List of Components 34-62
Weather Radar Control Panel 34-62
Transmitter-Receiver 34-62
Radar Synchronizer 34-62
Weather Radar Scope 34-62
Antenna 34-63
Contour Test Relay 34-63

OPERATIONAL CHECKOUT 34-63

TROUBLESHOOTING 34-65

REMOVAL AND REPLACEMENT 34-66
Weather Radar Antenna 34-66

FIGURES
34-17. Weather Radar System Components 34-68
34-18. Weather Radar System Block Diagram 34-70

34.7.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.7.1.1 GENERAL. (See figure 34-17.) Designed primarily to provide continuous inflight weather information, a Bendix RDR-1D radar system can also be used to avoid terrain. A portion of the short, intense pulses of microwave energy emitted by the system reflect back to the airplane from objects within range of the set. These reflected signals are translated into visible indications of the size, shape, range, and azimuth bearing of the objects. These indications appear as luminous forms on the radar scope. The maximum range of the system is 150 nautical miles, and the maximum usable azimuth sweep is 120 degrees. A weather RADAR CONTROL panel, a transmitter-receiver, a synchronizer-amplifier-power supply, a radar scope, an antenna, and a contour test relay constitute the main components of the system. Power is supplied through three circuit breakers on the copilot's wainscot circuit breaker panel: 115-volt, essential AC through a WEATHER RADAR REF AC circuit breaker, 115-volt, main AC through a WEATHER RADAR POWER AC circuit breaker, and 28-volt, main DC through a WEATHER RADAR DC circuit breaker. Also, for the system to operate properly, the No. 1 vertical gyro must be energized. It is supplied 400-volt, essential AC through a VERT GYRO NO. 1 AC circuit breaker and 28-volt, essential DC through a VERT GYRO NO. 1 DC circuit breaker, both on the pilot's side circuit breaker panel.

34.7.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Transmitter-Receiver	1	Bendix	RDR-1D	JR 1015-7 [1]
Radar Synchronizer	1	Bendix	SYN-1B	JR 1015-3 [1]
Radar Scope	1	Bendix	PPI-1E	JR 1015-11 [1]
Control Panel	1	Gables		G 908 [1]
Antenna	1	Bendix	ANT-1G	JR 1015-1 [1]
Contour Test Relay	1			AN 3311-2

[1] On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

34.7.1.3 WEATHER RADAR CONTROL PANEL. Operation of the system is controlled primarily from a weather RADAR CONTROL panel on the center console. This panel has a five-position ("OFF," "STBY," "NORM," "C'TOUR," "C'TOUR TEST") function selector switch, a GAIN control, and an ANT TILT switch. With the function selector switch in the "STBY" position, the set is warmed up for 3 to 5 minutes before use. In the "NORM" position, target pictures appear on the radar scope as essentially solid forms. However, in the "C'TOUR" position, the target pictures appear on the radar scope differentiated by the darkening of certain center portions to indicate areas of high rainfall rate, also areas of greatest turbulence. The GAIN control regulates the amplification of signals in the IF section of the receiver. The angle between the radar beam and the horizontal plane of the airplane can be varied 15 degrees either up or down with the ANT TILT switch.

CAUTION

When first turning on the system, leave the function selector switch in the "STBY" position for three minutes before putting the switch in the "NORM" or "C'TOUR" position. This precludes the possibility of serious damage to the system.

34.7.1.4 TRANSMITTER-RECEIVER. Operating in the range of 9,345 to 9,405 megacycles, a BENDIX RDR-1D radar transmitter-receiver emits 2.5 microsecond pulses of microwave energy at a rate synchronous with the airplane's main AC power supply. Reflected signals are received at the transmitted frequency, and appropriate signals are in turn routed to the radar scope. A current meter, a meter selector switch, and a directional coupler output test receptacle are on the front panel of the transmitter-receiver.

34.7.1.5 RADAR SYNCHRONIZER. A Bendix SYN-1B radar synchronizer provides the following: three simultaneous trigger pulses; plate, bias, reference, and filament voltages; and synchronization of the sweep drive and the antenna oscillation. A meter, a meter selector switch, and a trigger test jack are on the front panel of the unit.

34.7.1.6 WEATHER RADAR SCOPE. The PPI-1E is equipped with a BRIGHT DISPLAY storage tube which is designed to permit direct viewing under all ambient cockpit lighting conditions. The storage tube presents a clear, high-contrast target picture which retains its brilliancy for the time of a complete antenna revolution. The longer viewing time permits a pilot to make a more accurate evaluation of weather ahead, and the high contrast of the target picture permits him to view the indicator without the need of a special viewing hood in daylight, or under other high-ambient lighting conditions. Controls on the front panel of the radar scope include a DIMMER control, a filter adjusting control, a RANGE & MARKS selector switch, a TRACE ADJUST control, an ERASE RATE control, and a fast erase pushbutton switch. The DIMMER control regulates the brightness of the radar scope front panel. The brightness of the radar scope itself is determined by the position of the filter adjusting control which rotates a polaroid lens. A RANGE & MARKS selector switch has three switch positions: "20" and "5," "50" and "10," and "150" and "25." The first number represents nautical miles of range, and the second

34.7.1.6 (Cont).

number is the distance in nautical miles between range marks on the radar scope. Normal control over the trace cutoff level is provided by a TRACE ADJUST control. An ERASE RATE control enables the setting of the erase time, and a pushbutton switch permits a fast erasure of the target picture.

CAUTION

The weather radarscope must not be stored near any item containing a strong magnet, such as a magnetron used in radar transmitters. If the scope should inadvertently become magnetized, it must be degaussed before it will function correctly.

34.7.1.7 ANTENNA. A Bendix type ANT-1(G) antenna is used for both transmitting and receiving. The antenna assembly consists of a reflector, an RF feed system, azimuth and elevation drive systems, azimuth and elevation synchros, and mounting provisions. Pitch and roll data from vertical gyro No. 1 are furnished to maintain effectively the same antenna sweep (see Section 34.2). For ground test purposes, a two-position ("AZ," "OFF") AZ switch and a two-position ("EL," "OFF") EL switch are provided; both of these switches are normally held in the on position with a cotter pin. For ground checkout and alignment purposes, an azimuth index and a tilt index are included in the antenna assembly to indicate the position of the reflector.

34.7.1.8 CONTOUR TEST RELAY. When the weather radar function selector switch is placed in the "C'TOUR TEST" position, a contour test relay is energized. This applies 115-volt, AC power to a contour test circuit in the transmitter-receiver. The test results are satisfactory if a curved band of illumination appears on the radar scope.

CAUTION

Do not use high intensity illumination unless both scan and sweep are present. Otherwise, permanent damage to the radarscope indicator tube might result.

34.7.2 OPERATIONAL CHECKOUT.

CAUTION

If at any point during the following procedure the desired results are not realized, discontinue the check until the malfunction is corrected.

a. On the weather radar control panel, set the function selector switch in the "OFF" position, the ANT TILT switch at zero degrees, and the GAIN control fully counterclockwise.

b. On the radar scope, set both the DIMMER control and the TRACE ADJUST control fully counterclockwise, and the ERASE RATE control fully clockwise. The RANGE & MARKS selector switch can be in any position.

c. On the transmitter-receiver front panel, set the meter selector switch in the "OFF" position.

d. On the synchronizer-amplifier-power supply front panel, set the meter selector switch at "-250V."

e. On the antenna, place both the AZ switch and the EL switch in the "OFF" position.

f. Remove the HV fuse from the transmitter-receiver front panel.

g. Connect external power to the airplane (see Chapter 24), and energize the AC buses.

h. Make sure the following circuit breakers are closed: WEATHER RADAR REF AC, WEATHER RADAR POWER AC, and WEATHER RADAR DC.

CAUTION

Do not perform tests involving the radiation of RF energy by the antenna if it is directed at nearby large metal objects such as hangars or doors.

i. Place the function selector switch on the weather RADAR CONTROL panel in the "STBY" position, and allow a 5-minute warmup.
RESULT: The synchronizer-amplifier-power supply front panel meter should indicate .4 to .6.

j. Place the function selector switch in the "NORM" position, and set the synchronizer-amplifier-power supply front panel meter selector switch at "+140V," "+260V," "+300V," and "-250V" consecutively.
RESULT: In each case the synchronizer-amplifier-power supply front panel meter should indicate .4 to .6.

k. Place the synchronizer-amplifier-power supply meter selector switch in the "OFF" position.

l. Make sure the VERT GYRO NO. 1 DC and the VERT GYRO NO. 1 AC circuit breakers are open.

m. Place the antenna AZ switch in the "AZ" position.
RESULT: The antenna reflector assembly should oscillate through approximately 120 degrees.

n. Place the antenna AZ switch in the "OFF" position.
RESULT: The antenna reflector assembly should cease to oscillate.

o. Manually align the antenna reflector with the zero degree azimuth index line. Then secure the reflector with a strip of masking tape, yet not

34.7.2 (Cont).

restricting the reflector's free movement on its tilt axis.

p. Place the antenna EL switch in the "EL" position.
RESULT: With the ANT TILT switch at zero degrees, the reflector should assume a zero degree tilt.

CAUTION

The tilt position of the antenna can be checked by noting the tilt index. If incorrect, do not attempt to adjust the alignment of the tilt index lines at this point, for their misalignment is probably due to some other fault in the system.

q. Set the ANT TILT switch first at 15 degrees "UP," then at 15 degrees "DOWN."
RESULT: The antenna reflector should tilt up 15 degrees, then down 15 degrees.

r. Set the ANT TILT switch at zero degrees.
RESULT: The antenna reflector should assume a zero degree tilt.

s. Close the VERT GYRO NO. 1 DC and the VERT GYRO NO. 1 AC circuit breakers.
RESULT: If the pitch axis of the airplane is horizontal, the antenna should still have a zero degree tilt.

Note

If the pitch axis of the airplane is not horizontal, note the angle of difference and whether up or down. This angle of difference should be reflected at the tilt index on the antenna.

t. Note the exact reading of the tilt index. Manually tilt the antenna reflector to the extreme up position and release without forcing it against a mechanical stop.
RESULT: The antenna reflector should return to the position noted at the first part of this step.

u. Note the exact reading of the tilt index. Manually tilt the antenna reflector to the extreme down position and release without forcing it against a mechanical stop.
RESULT: The antenna reflector should return to the position noted at the first part of this step.

v. Place the AZ switch on the antenna in the "OFF" position, and align the antenna reflector with the azimuth mark.

w. Remove the No. 1 vertical gyro mounting screws, and lift up the front of the gyro.
RESULT: The antenna reflector should move down.

CAUTION

Whenever a gyro is moved, it should be returned exactly to its previous position when resecured. Thus, the number of washers for each mounting screw should be noted so that all can be replaced properly. If any of the washers are lost, or if the number of washers for each screw is not noted, the airplane must be leveled prior to leveling the gyro.

x. Move the antenna reflector to the left to about the 45-or 55-degree azimuth mark.

y. Lift up the outboard side of the No. 1 vertical gyro.
RESULT: The antenna reflector should move down.

z. Lift up the inboard side of the No. 1 vertical gyro.
RESULT: The antenna reflector should move up.

aa. Replace the No. 1 vertical gyro mounting screws.

ab. Align the antenna reflector with the zero azimuth mark and secure with masking tape. Set the AZ switch in the "AZ" position and replace the cotter pin.

ac. Pull out the VERT GYRO NO. 1 DC and the VERT GYRO NO. 1 AC circuit breakers.

ad. Turn the TRACE ADJUST control clockwise until the range marks appear on the radar scope.
RESULT: The sweep trace should be vertical, in line with the zero-degree azimuth on the indicator dial.

ae. Turn the antenna reflector to first one side, then the other, in 30-degree increments.
RESULT: The sweep trace on the radar scope should deflect the same number of the degrees in the same direction as the antenna reflector is moved.

af. Set the TRACE ADJUST control on the indicator to provide a sweep trace of low intensity, but distinguishable over the entire radar scope.
RESULT: The sweep trace should extend from the apex of the radar scope to or beyond the outer perimeter at all points, and should not be curved or intermittent. The rate of the sweep trace should coincide with the movement of the antenna.

ag. Turn the GAIN control on the control panel clockwise.
RESULT: Noise should appear on the radar scope.

34.7.2 (Cont).

ah. Turn the GAIN control fully counterclockwise.
RESULT: The noise on the radar scope should disappear.

ai. Set the RANGE & MARKS selector switch on the indicator at "20" miles range and "5" miles distance between range marks.
RESULT: Four equally spaced range marks should appear on the radar scope, the outer mark being about 1/8 inch from the outer perimeter.

aj. Set the RANGE & MARKS selector switch at "50" miles range and "10" miles distance between range marks.
RESULT: Five equally spaced range marks should appear on the radar scope, the outer mark being about 1/8 inch from the outer perimeter.

ak. Set the RANGE & MARKS selector switch at "150" miles range and "25" miles distance between range marks.
RESULT: Six equally spaced range marks should appear on the radar scope, the outer mark being about 1/8 inch from the outer perimeter.

al. Turn the TRACE ADJUST fully counterclockwise.
RESULT: The radar scope display should be completely extinguished.

am. Turn the DIMMER control clockwise.
RESULT: The entire radar scope overlay should go on and increase in intensity as the DIMMER control is turned clockwise.

an. Place the function selector switch in the "OFF" position.

ao. Place the switches and controls in the positions called out in steps a and b.

ap. Remove the masking tape used to secure the antenna reflector.

aq. Replace the HV fuse in the transmitter-receiver front panel.

ar. Disconnect external power from the airplane.

34.7.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

Weather Radar Manuals, Bendix Radio part numbers I.B. 732, I.B. 739, I.B. 740-1, I.B. 765.

No manuals are available for the Gables control panels. These should be returned to the manufacturer for repair.

PROBABLE CAUSE	ISOLATION PROCEDURE
TRANSMITTER-RECEIVER DOES NOT OPERATE IN ANY FUNCTION.	
28V supply not getting to receiver.	Check circuit breaker setting. Check wiring from circuit breaker to control panel to transmitter-receiver.
Defective transmitter receiver.	Substitute a new R-T unit, and check for proper operation.
Defective function selector switch in control panel.	Substitute a new control panel, and check for proper operation.
NO INDICATION OR INCORRECT INDICATION ON PANEL METER OF SYNCHRONIZER - AMPLIFIER - POWER SUPPLY.	
Defective synchronizer-amplifier-power supply unit.	Substitute a new unit, and check for proper operation.
PPI-1E INDICATOR INOPERATIVE OR GIVES FAULTY INDICATION.	
DEFECTIVE INDICATOR.	Substitute a new indicator, and check for proper operation.
PPI-1E INDICATOR WILL NOT RESPOND TO CONTROLS.	
Defective controls in indicator.	Substitute a new indicator, and check for proper operation.

34.7.4 REMOVAL AND REPLACEMENT.

34.7.4.1 WEATHER RADAR ANTENNA.

a. Extend and remove the nose cone from the airplane (see Section 34.0).

b. Disconnect the antenna cabling connector from the weather radar.

c. Remove the four screws from the flexible waveguide at the antenna end of the waveguide, and disconnect the flexible waveguide.

CAUTION

Position at least one man at the side of the weather radar antenna to hold the antenna while performing the next steps. Do not grasp any part of the antenna dish assembly. Hold the antenna by the housing only, and be careful not to insert fingers into the waveguide openings.

d. Remove the four nuts and four washers that secure the radar antenna to the antenna mount.

e. Remove the four bolts from the antenna mount and antenna, and carefully remove the radar antenna from the airplane electronic equipment rack.

f. To install the weather radar antenna, reverse the procedures in steps (b) through (e).

g. Perform an operational checkout of the weather radar system.

h. Replace the nose cone. (See Section 34.10.)

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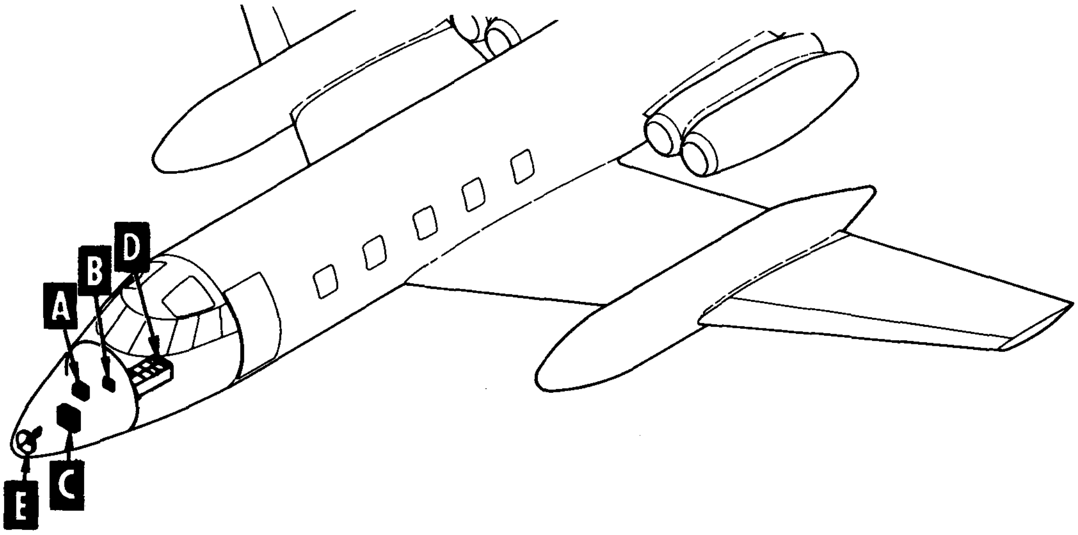

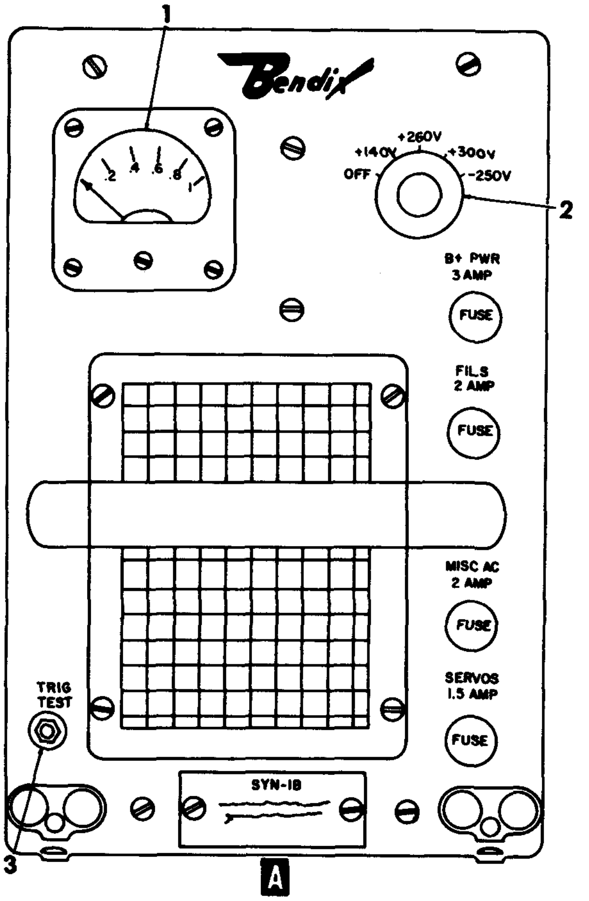

A

RADAR SYNCHRONIZER
(LOOKING INBOARD)

1. PANEL METER
2. METER CIRCUIT SELECTOR SWITCH
3. TRIGGER TEST JACK
4. DIMMER CONTROL
5. FAST ERASE BUTTON
6. ERASE RATE CONTROL
7. FILTER ADJUSTING CONTROL
8. TRACE ADJUSTING CONTROL
9. RANGE AND MARKS SELECTOR SWITCH
10. FUNCTION SELECTOR SWITCH
11. GAIN CONTROL
12. ANTENNA TILT SWITCH
13. CURRENT METER
14. METER CIRCUIT SELECTOR SWITCH
15. DIRECTIONAL COUPLER OUTPUT TEST
16. AZ SWITCH
17. EL SWITCH
18. AZIMUTH INDEX
19. TILT INDEX

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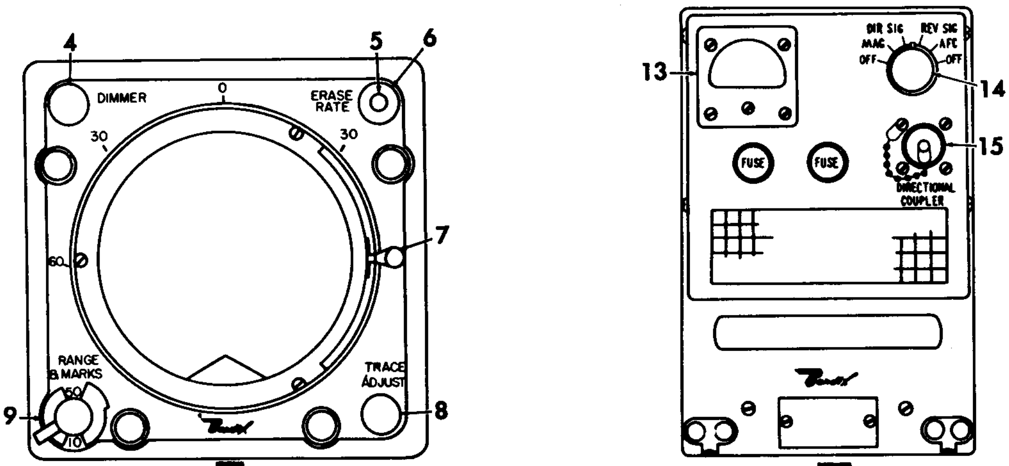
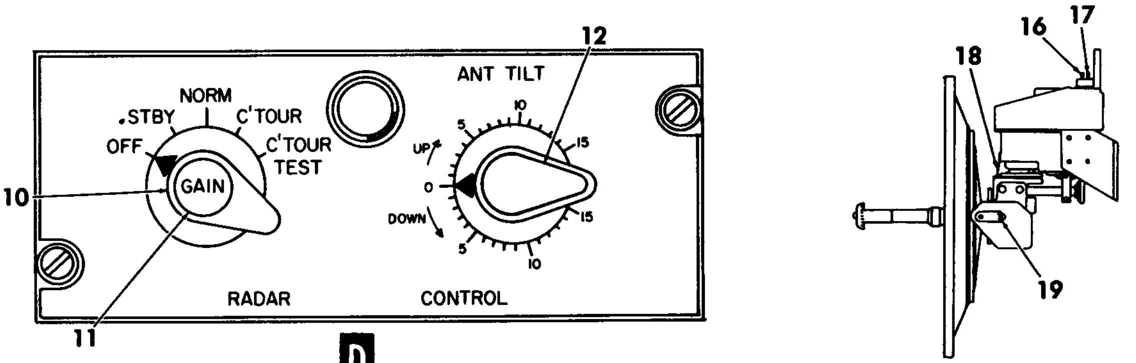
Figure 34-17. Weather Radar System Components (Sheet 1 of 2)



B
WEATHER RADAR SCOPE
(LOOKING FORWARD)

C
TRANSMITTER-RECEIVER
(LOOKING INBOARD)



D
RADAR CONTROL PANEL
(LOOKING DOWN)

E
WEATHER RADAR ANTENNA

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Figure 34-17. Weather Radar System Components (Sheet 2 of 2)

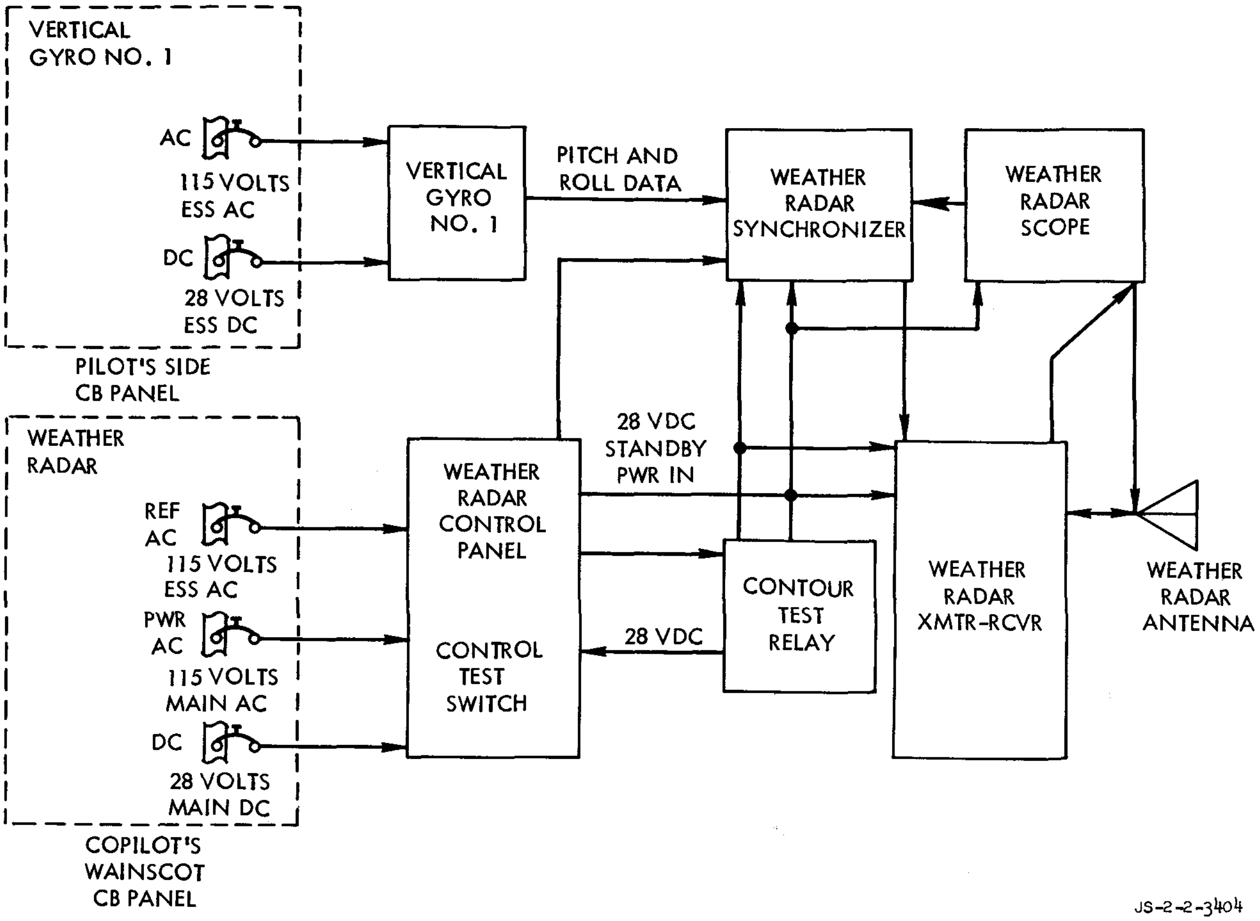


Figure 34-18. Weather Radar System Block Diagram

SECTION 34.8
DISTANCE MEASURING EQUIPMENT SYSTEM

Contents

Contents	Page
DESCRIPTION AND OPERATION	34-71
General	34-71
List of Components	34-72
Control Panel	34-72
Receiver Transmitter	34-72
Distance Indicators	34-72
Antenna	34-72
OPERATIONAL CHECKOUT	34-72
Test Equipment Required	34-72
Distance Measuring Equipment	34-72
TROUBLESHOOTING	34-73
FIGURES	
34-19. DME System Components	34-75
34-20. DME System Block Diagram	34-76

34.8.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.8.1.1 GENERAL. (See figures 34-19 and 34-20.) By measuring the round trip time of radio signals between the airplane and a ground station, a Collins 860E-1 radio set determines the distance of the airplane from the ground station. Random interrogation pulses, which during operation of the system are continuously transmitted from the airplane, are received by a ground station which transmits the same pulse sequence in reply. When the ground station reply signals are received at the airplane, an automatic range search process enables the airborne radio set to lock in on the reply pulses corresponding to the set's own interrogation pulses. Thus, a time delay is measured and translated into a distance reading which appears on two distance indicators on the main instrument panel. After the automatic range search is locked onto the appropriate signal, a tracking process is initiated so that gradual changes in distance can be indicated. A control panel and an antenna are also included in the system. Main bus power at 115 volts, AC and 28 volts, DC is supplied through a DME AC and a DME DC circuit breaker, respectively, on the copilot's wainscot circuit breaker panel. The system has a range of 196 nautical miles.

34.8.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Radio Set	1	Collins	860E-1	522 1228 00 [1]
Indicator	2	Collins	339D-1	522 1859 003 [1]
Antenna	1	Electronic Speciality		AS-L-647
Control Panel	1	Gables		G-906V [1]

[1] On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

Note

The abbreviations DME and DMET are synonymous as used in this manual.

34.8.1.3 CONTROL PANEL. The distance measuring system is controlled from a VORTAC-VHF panel on the center console. A VORTAC VOL control, a STBY-DMET SWITCH, a SEARCH O'RIDE-NORM switch, and a VORTAC frequency selector are the DME controls located on this panel. When the VORTAC VOL control is turned on, the No. 1 VOR power relay is energized, thereby connecting power to the distance measuring radio set. With the STBY-DMET switch in the "STBY" position, the radio set is ready for use; in the "DMET" position the radio set operates. When a VOR frequency is dialed on the VORTAC frequency selector, a corresponding DME station is simultaneously selected. The SEARCH O'RIDE-NORM switch, when in the "NORM" position, reduces possible signal interference by limiting the range search to 35-70 miles, according to a preset adjustment. In the "SEARCH O'RIDE" position, the search process is effective to the maximum range of the system.

34.8.1.4 RADIO SET. A Collins 860E-1 radio set is composed of a chassis and four modules: AN RF, a decoder, a distance, and a distance mechanism module. The unit transmits in the frequency range of 1,025-1,150 megacycles with 1-megacycle spacings between channels. Reception is in the frequency ranges of 962 to 1,024 and 1,151 to 1,213 megacycles. Both the transmitting and receiving frequencies are tuned to the DME station corresponding to the VOR station dialed with the VORTAC frequency selector. A DME station identification code is received and routed through the AUDIO SELECTOR panel.

34.8.1.5 DISTANCE INDICATORS. On both the pilot's and the copilot's side of the main instrument panel the distance of the airplane from the selected station appears on a Collins 339D-1 distance indicator. The distance is given in nautical miles. The indicator is equipped with a warning flag which appears in the viewing window to indicate the 860E-1 radio set is in search and not locked on, or is inoperative.

34.8.1.6 ANTENNA. (See figure 34-3.) The same antenna, an L-band, annular slot type, is used for both transmitting and receiving.

34.8.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
PBN-4123	Radio Test Set	Supplies distance signals to receiver.

34.8.2.1 DISTANCE MEASURING EQUIPMENT.

a. Connect external power to the airplane (see Chapter 24), and energize the AC buses.

b. Be sure the following circuit breakers are closed: DMET AC, DMET DC, VOR NO. 1 AC, VOR NO. 1 DC, INPH, and COPILOT ISOLATION AMP.

c. Place the STBY-DMET switch in the "STBY" position, and the VORTAC VOL control in the on position.

d. Set up and adjust a Federal Telephone and Radio test set, Model PBN-4123, (or equal) according to the instructions with the set. The test set must be within effective operating distance of the DME antenna, and the path between them must be unobstructed.

34.8.2.1 *(Cont)*.

e. Using the VORTAC selector switch, dial a frequency other than a local one and other than the test set frequency.

f. Place the STBY-DMET switch in the "DMET" position, and the SEARCH O'RIDE-NORM switch in the "NORM" position.
RESULT: The numbers on both distance indicators should spin.

g. Set the VORTAC frequency selector at 134.6 or 117.1 megacycles, and the distance calibrated dial on the test set at 5 miles.
RESULT: The numbers on both distance indicators should rotate and stop at 5 (±2) miles.

h. Place the INT monitor switch on the pilot's AUDIO SELECTOR panel in the on position, and listen to the pilot's headset.
RESULT: A distinct tone should be heard.

i. Set the distance calibration dial at 100 (±2) miles, and place the SEARCH O'RIDE-NORM switch in the "SEARCH O'RIDE" position.
RESULT: The numbers on both indicators should rotate and stop at 100 (±3) miles.

j. Repeat step i for 20, 50, 150, and 180 miles.

k. While keeping it in proper operating position, move the test set away from the airplane DME antenna until the range begins to search.
RESULT: The range should not begin to search at less than 20 feet.

l. Place the STBY-DMET switch in the "STBY" position.
RESULT: Within 10 seconds the distance indicator flags should appear.

m. Move the test set back to the original test position, and place the STBY-DMET switch in the "DMET" position.
RESULT: Both distance indicators should indicate the distance selected on the test set.

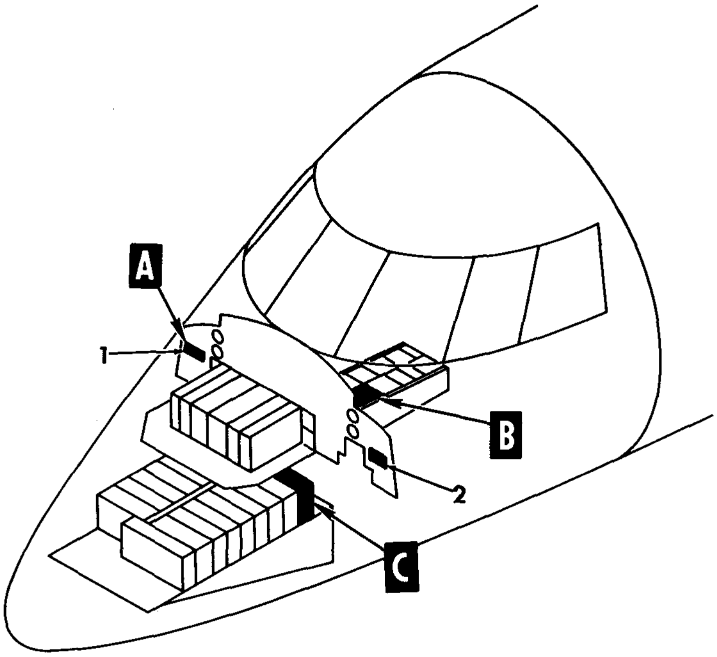
n. Disconnect external power and restore the airplane to normal.

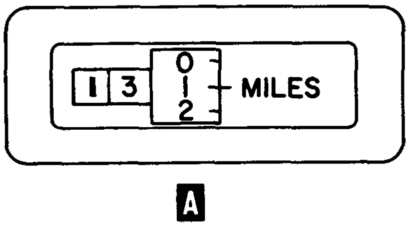
34.8.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

DME Radio Set manuals, Collins part number 523 0050 006, TD 519A, 523 0050 003, 523 0050 004.

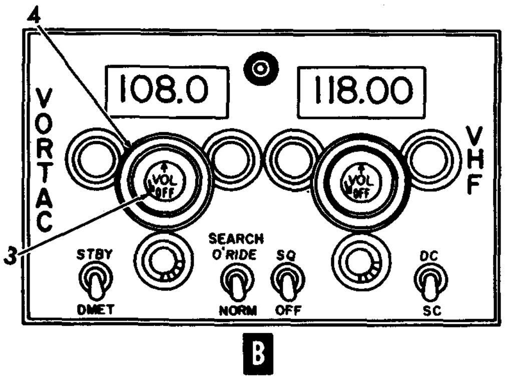
There are no manuals available for the Gables control panels. These should be returned to the manufacturer for repair.

PROBABLE CAUSE	ISOLATION PROCEDURE
DME SYSTEM INOPERATIVE.	
Open circuit breaker.	Check setting of 28VDC and 115 VAC DME circuit breakers on copilot's circuit breaker panel.
Defective VOR NO. 1 power relay.	Check VOR NO. 1 power relay for faulty contacts.
Defective DME radio set.	Substitute a new DME radio set, and check for proper operation.
Defective VOR TAC-VHF control panel.	Substitute a new control panel, and check for proper operation.
Defective antenna.	Connect ramp test set directly to DME radio set. If proper operation is then obtained, the trouble is probably in the antenna or coaxial cable.
Defective wiring.	Check wiring and connections with reference to system wiring diagram.
BOTH DME AND VOR SYSTEMS INOPERATIVE.	
Defective VOR NO. 1 power relay.	Check VOR NO. 1 power relay for proper operation.
Defective VORTAC-VHF control panel.	Substitute a new control panel, and check for proper operation.
WARNING FLAG EXPOSED ON ONE INDICATOR ONLY.	
Defective indicator.	Substitute a new indicator, and check for proper operation.

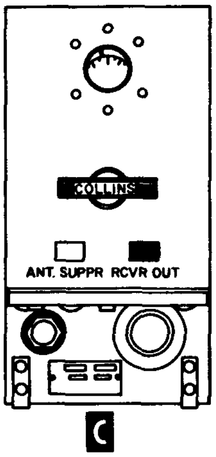




DISTANCE INDICATOR (TYPICAL)
(LOOKING FORWARD)



VORTAC-VHF CONTROL PANEL
(LOOKING DOWN)



RADIO SET
(LOOKING INBOARD)

1. COPILOT'S DISTANCE INDICATOR
2. PILOT'S DISTANCE INDICATOR
3. VORTAC VOL CONTROL
4. VORTAC FREQUENCY SELECTOR

JS-2-2-3418

Figure 34-19. DME System Components

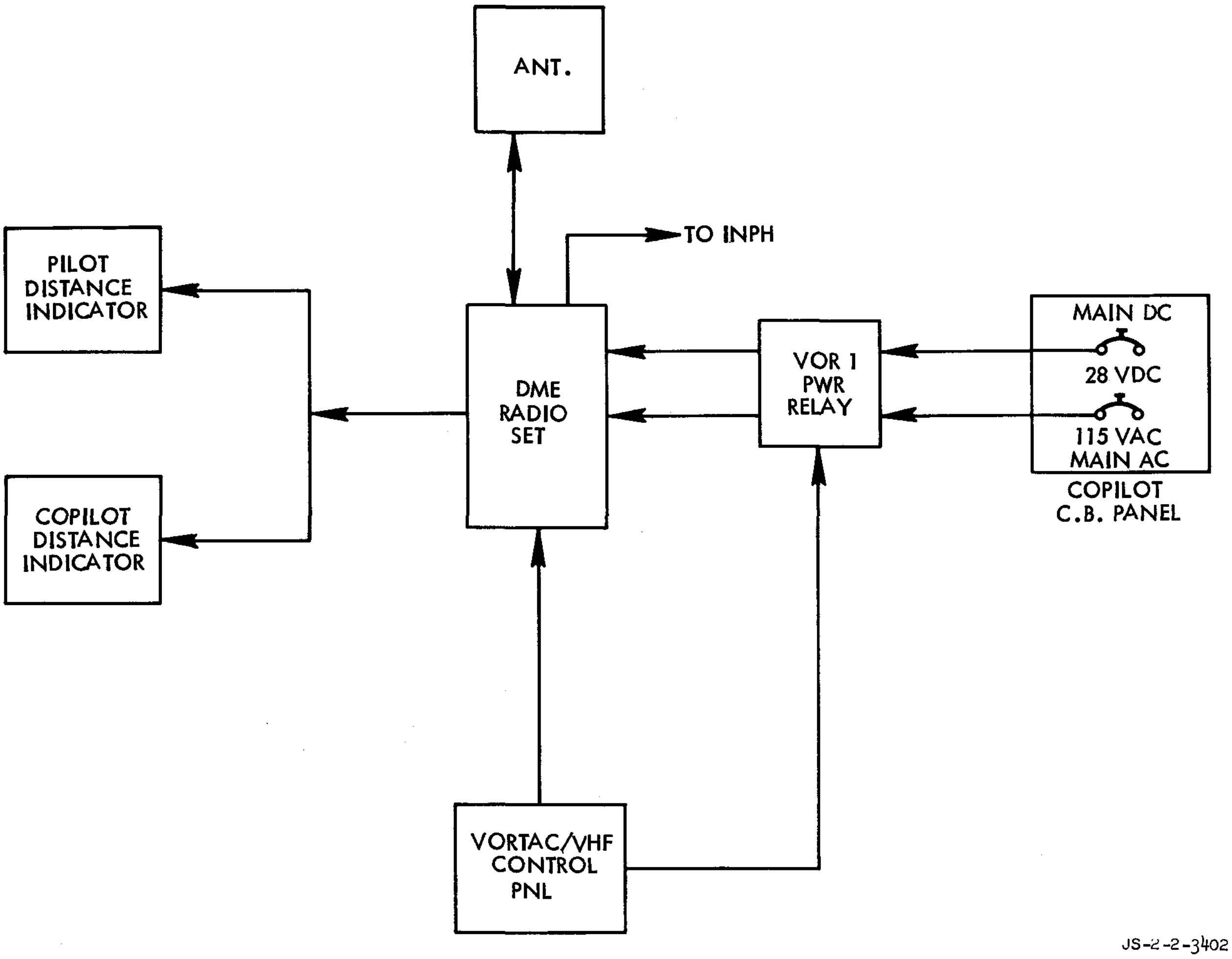


Figure 34-20. DME System Block Diagram

SECTION 34.9
AIR TRAFFIC CONTROL TRANSPONDER SYSTEM

Contents Page

DESCRIPTION AND OPERATION 34-77
General 34-77
List of Components 34-77
Control Panel 34-78
Transponder 34-78
Antenna 34-78

OPERATIONAL CHECKOUT 34-78
Test Equipment Required 34-78
ATC Transponder System 34-78

TROUBLESHOOTING 34-79

FIGURES
34-21. ATC Transponder System Components 34-81
34-22. ATC Transponder System Block Diagram 34-82

34.9.1 DESCRIPTION AND OPERATION.

Note

This section applies to airplanes 5001 thru 5045 which have the complete system described in this section installed, and to airplanes 5046 thru 5049, 5052, 5056, and 5059 which have installation provisions and wiring provisions for the system described in this section. This section does not apply to airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 and up which have installation provisions and wiring provisions for "solid state" equipment. Refer to Section 34.11 for information pertaining to these airplanes.

34.9.1.1 GENERAL. (See figures 34-21 and 34-22.) There are two types of transponders used in JetStar airplanes. Airplanes 5001 thru 5035 use a Collins 621A-2, and airplanes 5036 and up use a Collins 621A-2A. The 621A-2 transponder includes circuitry for two-pulse sidelobe suppression, and the 621-2A includes circuitry for three-pulse sidelobe suppression. Upon receiving an interrogating signal from a ground station transmitter, a Collins air traffic control transponder transmits a reply signal. The interrogating signal received is a pair of spaced pulses at 1,030 megacycles; the reply signal is 2 to 8 accurately spaced, coded pulses and an identification pulse transmitted at 1,090 megacycles. Depending upon the spacing of the interrogating signal pulse pairs, either of two modes of operation can be used.

34.9.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Transponder	1	Collins	621A-2	522 1571 014 △1
Transponder	1	Collins	621A-2A	522 2305 014 △2 △3
ATC TRANSP-MKR Control Panel	1	Gables		G 907 △3
Antenna	1	Electronic Specialty		LB 422

△1 Airplanes 5001 thru 5035

△2 Airplanes 5036 thru 5049, 5052, 5056 and 5059.

△3 On airplanes 5046 thru 5049, 5052, 5056 and 5059, this equipment is not installed by Lockheed when the airplane is manufactured.

34.9.1.3 CONTROL PANEL. An ATC-TRANSP-MKR control panel is located towards the aft end of the center console. The air traffic control transponder portion of the control panel includes a mode selector, a sensitivity switch, a code selector, a selected-code indicator, and an IDENT button. The transponder is turned on and off with the four-position ("OFF," "STBY," "A," "B") mode selector. In the "STBY" position, the transponder warms up in 80 seconds, and then remains ready for immediate use. Further, in the "A" and "B" positions, respectively, an 8-microsecond and a 17-microsecond spacing between interrogation pulses is provided. For use especially in areas in which several stations are operating, the two-position ("NORM," "LO SENS") sensitivity switch can be placed in the "LO SENS" position to reduce the gain of the receiver when close to the interrogating stations. Any of 64 different coded replies, as shown on the selected-code indicator, can be dial with the code selector. When the IDENT button is pressed momentarily, identification pulses are transmitted continuously for a preset length of time.

34.9.1.4 TRANSPONDER. A Collins transponder consists of a chassis-panel assembly and six modules: an r-f module, a decoder module, an encoder module, an encoder gate module, a modulator module, and a power supply module. On the front panel are four connectors: an antenna ("ANT"), a transmitter-trigger ("TRIGGER"), a receiver output ("RCVR OUT"), and an external suppression ("SUPPR") connector. An audio monitor circuit is used to provide an audible tone when the transponder is interrogated at the proper rate by the ground station. The transponder operates at 500 watts peak pulse power, and has a range of approximately 200 nautical miles at an altitude of 30,000 feet, and 105 nautical miles at an altitude of 10,000 feet.

39.9.1.5 ANTENNA. (See figure 34-3.) Both transmission and reception are achieved through a non-directional L-band, annular slot type antenna.

34.9.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Collins 476X-1	Ramp Test set	Supplies interrogating signals to transponder and receives coded reply.

34.9.2.1 ATC TRANSPONDER SYSTEM.

a. Place a Collins 476X-1 ramp test set approximately 8 feet from the airplane antenna with the test set antenna pointed toward the airplane antenna, and with an unobstructed, line-of-sight path between the two antennas.

Note

Steps b through h are for self-testing of the test set.

b. Place the MODE selector on the test set in the "SELF TEST" position.
RESULT: The battery voltage indication should be in the green portion of the panel meter, and the IDENT flag should appear.

CAUTION

If the battery voltage indication is below the green portion of the panel meter, use of the batteries might damage them. Recharging can be accomplished by placing the function switch in the "OFF" position and plugging the ramp test set into a 115-volt, AC source. The set operates normally with the AC source connected.

c. Dial "00" with the code selector on the test set.
RESULT: The code flag should appear.

d. Dial "77" with the code selector.
RESULT: The code flag should reappear.

e. Place the MODE selector in the "A" position.
RESULT: The A flag should appear.

f. Place the MODE selector in the "B" position.
RESULT: The B flag should appear.

g. Place the MODE selector in the "AB" position.
RESULT: The A and B flags should appear alternately at 15-second intervals.

h. Dial "00" with the code selector, and place the MODE selector in the "A" position. Place the CALIBRATE OUTPUT-BATTERY VOLTAGE switch in the "CALIBRATE OUTPUT" position, and adjust output to the center of the green portion of the panel meter to set the r-f level. Set the T/R ATTENUATOR at "10," though further adjustment might be necessary.

i. Connect external power to the airplane (see Chapter 24), and energize the AC buses.

j. Make sure the TRANSPONDER AC and the TRANSPONDER DC circuit breakers are closed.

k. On the ATC TRANSP-MKR control panel, place the mode selector in the "STBY" position, and allow at least an 80-second warmup.

l. On the ATC TRANSP-MKR control panel, dial "00" with the code selector. Place the sensitivity switch in the "LO SENS" position and place the mode selector in the "A" position.

34.9.2.1 (Cont).

RESULT: The A flag and the code flag should appear on the ramp test set.

m. Place both the ATC TRANSP-MKR control panel and the ramp test set mode selectors in the "B" position.
RESULT: The B flag and the code flag should appear on the ramp test set.

n. Increase attenuation at the ramp test set by one step, and place the ATC-marker control panel sensitivity switch in the "NORM" position.
RESULT: The code flag should appear on the ramp test set.

o. On the ramp test set, adjust the transponder control to the code complement, 77 minus the selected code.
RESULT: The code flag should appear on the ramp test set.

p. Depress the IDENT button on the ATC-marker control panel.
RESULT: The IDENT flag should appear on the ramp test set for approximately 10 seconds.

q. Repeat steps l through p for the following code settings for the ATC-marker control panel and the ramp test set code selectors:

Code	Complement
01	76
12	65
23	54
34	43
45	32
56	21
67	10
70	07

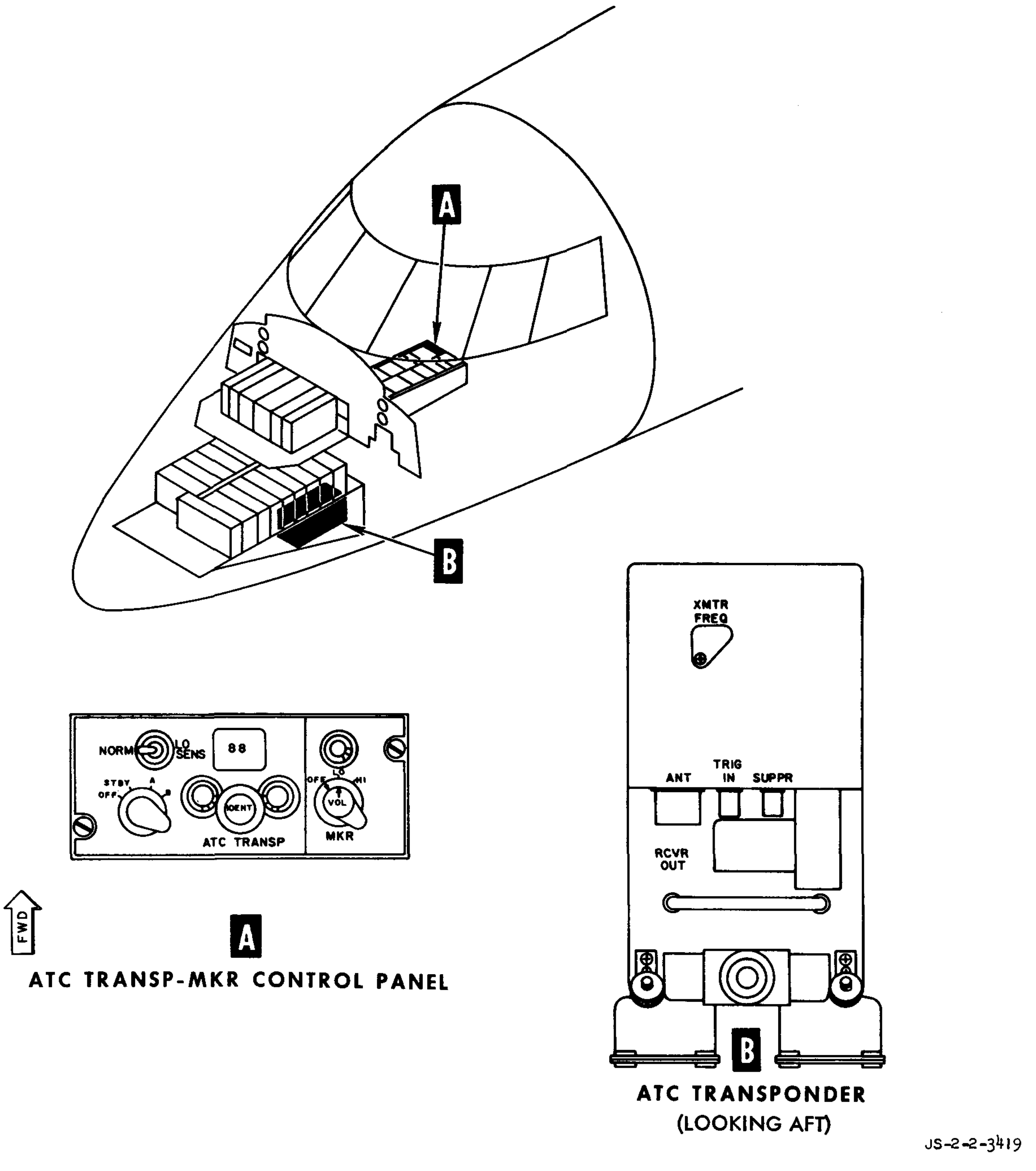
r. Disconnect external power and restore the airplane to normal.

34.9.3 TROUBLESHOOTING. The following troubleshooting chart is intended as an aid in isolating trouble to a particular system component. For troubleshooting a defective component, refer to the following handbooks:

Transponder Manuals, Collins part numbers 523 0091 006, 523 0091 004, 523 0091 003.

There are no manuals available for the Gables control panels. These should be returned to the manufacturer for repair.

PROBABLE CAUSE	ISOLATION PROCEDURE
TRANSPONDER INOPERATIVE.	
Open circuit breaker.	Check Setting of 28 VDC and 115 VAC transponder circuit breaker on copilot's circuit breaker panel.
Defective transponder.	Substitute a new transponder, and check for proper operation.
Defective control panel.	Substitute a new control panel, and check for proper operation.
Defective antenna.	Connect ramp test set 476X-1 directly to the 621A-2 or 621A-2A. If proper operation is then obtained, the trouble is probably in the antenna or coaxial cables.
Defective wiring.	Check wiring and connections with reference to system wiring diagram.



ATC TRANSP-MKR CONTROL PANEL

ATC TRANSPONDER (LOOKING AFT)

Figure 34-21. ATC Transponder System Components

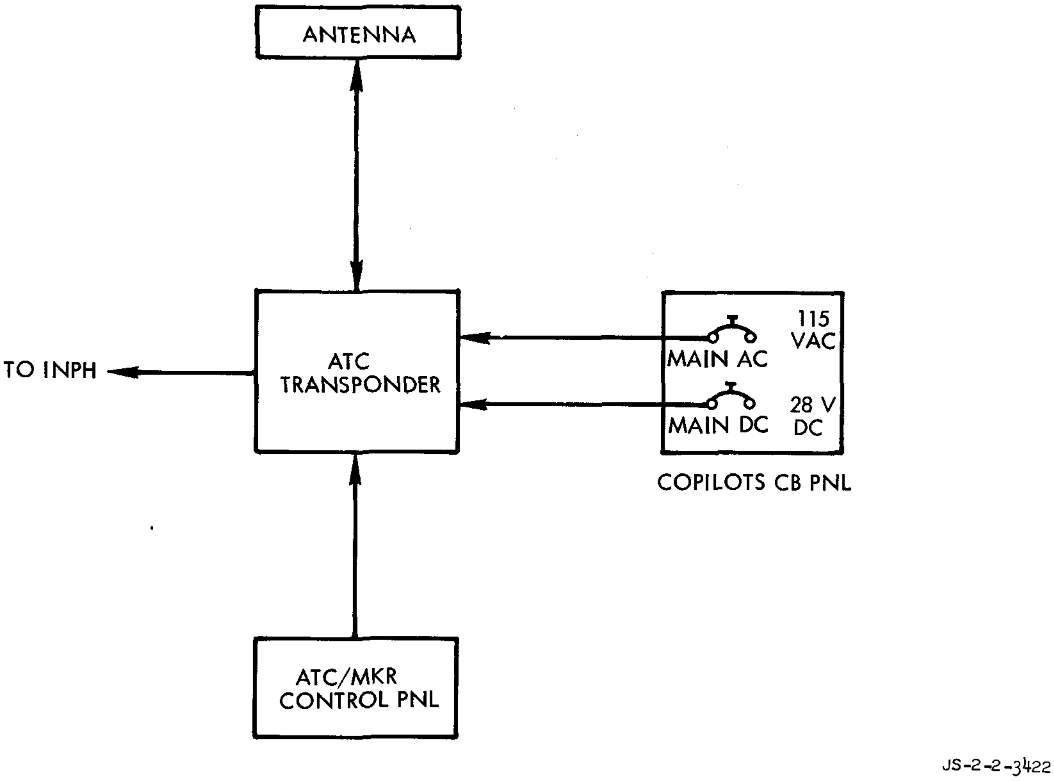


Figure 34-22. ATC Transponder System Block Diagram

SECTION 34.10

AIR DATA, ATTITUDE, AND DIRECTION INSTRUMENTATION

Contents Page

DESCRIPTION AND OPERATION 34-83
  General 34-83
  Airspeed Indicators 34-84
  Altimeters 34-84
  Dual Altimeter and Cabin Differential Pressure Indicator 34-84
  Mach Meter 34-84
  Outside Air Temperature Indicating System 34-84
  Overspeed Warning System 34-84
  Pitot-Static System 34-85
  Turn and Slip Indicating System 34-85
  Vertical Velocity (Rate-of-Climb) Indicators 34-85
OPERATIONAL CHECKOUT 34-86
  Airspeed Indicators 34-86
  Altimeters 34-86
  Mach Meter 34-86
  Outside Air Temperature Indicating System 34-86
  Overspeed Warning System 34-86
  Pitot-Static Systems 34-87
  Pilot's Pitot-Static System Leak Test 34-87
  Pilot's Pitot System 34-87
  Pilot's Static System 34-87
  Copilot's Pitot-Static System Leak Test 34-88
  Copilot's Pitot System 34-88
  Copilot's Static System 34-88
  Third Pitot System 34-88
  Third Static System Leak Test 34-88A
  Third Static System 34-88A
  Autopilot System Components 34-88A
  Dual Altimeter and Differential Pressure Indicator 34-88B
  Cabin High Differential Pressure Switch 34-88B
  Stall Warning Altitude Pressure Switches No. 1 and No. 2 . . . . . 34-88B
  Mach Trim Compensator Control Unit 34-88B
  Restoring System to Normal 34-88B
  Turn and Slip Indicating System 34-88C
  Vertical Velocity (Rate-of-Climb) Indicators 34-88C
TROUBLESHOOTING 34-88C
  Airspeed Indicators 34-88C
  Altimeters 34-89
  Mach Meter 34-89
  Outside Air Temperature Indicating System 34-89
  Overspeed Warning System 34-90
  Turn and Slip Indicating System 34-91
  Vertical Velocity (Rate-of-Climb) Indicators 34-91
EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION 34-91
REMOVAL AND REPLACEMENT 34-91
  Removal of Static Port Heaters 34-91
  Replacement of Static Port Heaters 34-91

FIGURES

  34-23. Pitot-Static System Plumbing 34-93
  34-24. Pitot-Static System Schematic Diagram 34-94
  34-24A. Static Port Heaters Installation 34-96
  34-25. Air Data and Directional Instruments 34-97

34.10.1 DESCRIPTION AND OPERATION.

34.10.1.1 GENERAL. Dual flight instruments (pilot and copilot), a single mach meter and an outside air temperature indicator are provided on the main instrument panel. The mach meter is centrally located on the main instrument panel and visible to the pilot and copilot. The outside air temperature indicator is located on the copilot's side of the main instrument panel, visible to the pilot and copilot. The dual flight instruments are:

34.10.1.1 (Cont).

a. Airspeed indicators.

b. Altimeters.

c. Turn and slip indicators.

d. Vertical velocity (rate of climb) indicators.

A pitot-static system is provided to supply a source of pitot pressure and static pressure to flight instruments and secondary systems. Secondary flight systems are:

a. The autopilot system (refer to Chapter 22).

b. The cabin pressurization system (refer to Chapter 21).

c. The mach trim compensator control system (refer to Chapter 27).

d. The overspeed warning system (refer to this section).

e. The stall warning altitude pressure switches systems No. 1 and No. 2 (refer to Chapter 27).

34.10.1.2 AIRSPEED INDICATORS. One airspeed indicator on the pilot's side of the main instrument panel is connected to the pilot's pitot system and to the pilot's static system. Another identical indicator on the copilot's side is connected to the copilot's pitot system and to the copilot's static system. Airspeed is determined by the pitot static differential pressure and indicated by a long, white airspeed pointer, in hundreds of knots. A vernier scale in the window in the top portion of the dial face indicates the speed in knots between 0 and 100. Maximum allowable speed at any altitude is given by a striped, maximum allowable speed pointer, set with a screw at the lower left-hand corner of the indicator to stop at 350 knots for an air pressure of 29.92 inches of mercury. A mach number index pointer, which is a small triangular pointer on the perimeter of the dial of the airspeed indicator, is set with an adjustment screw on the back of the indicator, to a preset value of mach 0.82. Up to an altitude of approximately 24,500 feet the maximum allowable speed pointer will remain at 350 knots, but above that altitude, this limit will decrease. Thus, the indicated airspeed should never be greater than the given maximum allowable airspeed, or the safe operating limitations of the airplane will be exceeded.

34.10.1.3 ALTIMETERS. Both the pilot and the copilot are provided with an altimeter on their side of the main instrument panel. The diaphragms in the pilot's altimeter are actuated by the pilot's static system, and the diaphragms in the copilot's altimeter by the copilot's static system. In each altimeter, movement of the diaphragms moves the pointers. On each indicator dial, three concentrically mounted pointers in combination present an altitude reading. The long pointer indicates hundreds of feet, the short pointer thousands of feet, and the long, thin pointer tens of thousands of feet. A correction for local barometric conditions can be made with an adjustment knob on the lower left-hand corner of the altimeter. The corrected barometric pressure selected (from 28.1 to 31.0 inches of mercury) appears in a window on the right-hand side of the altimeter. At ground level, the altimeter indication should be zero (±30) feet if the proper barometric pressure correction has been made. A black and white striped warning area is visible at the bottom of the altimeter up to about 16,600 feet.

34.10.1.4 DUAL ALTIMETER AND CABIN DIFFERENTIAL PRESSURE INDICATOR. A combination dual altimeter and cabin differential pressure indicator is connected to the third static system and is located on the pressurization control panel in the flight station's overhead console (refer to Section 21.1).

34.10.1.5 MACH METER. Indication of the mach number is provided in the range of 0.3 through 1.0 by a mach meter on the pilot's side of the main instrument panel. The mach number represents the ratio of the speed of the airplane to the speed of sound, both at a given condition of altitude pressure and temperature. Both a pitot pressure and a static pressure source are connected to the indicator. An aneroid diaphragm and a differential pressure diaphragm respond, respectively, to altitude pressure and differential pressure. The diaphragm actions are mechanically resolved through an arrangement of levers and gears to position the indicator pointer. A mach number of .82 should never be exceeded, as indicated on the speed limitations placard mounted on the main instrument panel to the left of the machmeter (pilot's position).

34.10.1.6 OUTSIDE AIR TEMPERATURE INDICATING SYSTEM. An outside air temperature indicator on the copilot's side of the main instrument panel provides an outside air temperature reading in the range from minus 70 to plus 50° Centigrade. Outside air temperature is measured by a resistance bulb, resistance-bridge type circuit. Temperature is sensed by an air temperature bulb whose resistance varies with temperature changes. The sensing element in the bulb is a ribbon of nickel whose resistance depends on temperature. The sensing element is encased for flush mounting and is on the fuselage skin forward of the entrance door. The bulb is wired into one side of the bridge type circuit. The varying resistance of the air temperature bulb, which constitutes one leg of the bridge type circuit, creates an imbalance across the bridge, causing an indicator pointer deflection by a deflection coil in the indicator. Main 28-volt, DC bus power is supplied to the system through an EXT AIR TEMP circuit breaker on the copilot's side circuit breaker panel.

34.10.1.7 OVERSPEED WARNING SYSTEM. Main DC bus power at 28 volts is supplied to the overspeed warning system through an OVERSPEED WARN circuit breaker on the copilot's wainscot circuit breaker

34.10.1.7 (Cont).

panel. If an equivalent airspeed of 350 (+6,-0) knots is exceeded, a warning bell, located beneath the pilot's side circuit breaker panel, sounds. Voltage is applied to the warning bell when an overspeed warning switch in the nose section closes. This occurs when a pitot-static differential pressure corresponding to 350 (+6,-0) knots EAS is present to actuate the switch. The switch reopens and the bell stops sounding before the airspeed drops to 348 knots EAS. For test purposes, the warning bell can be sounded when an OVERSPEED WARNING TEST switch on the copilot's inclined panel is depressed.

Note

The equivalent airspeeds as noted above are applicable on airplanes 5046 and up and on airplanes 5001 thru 5045 with the intent of Service Bulletin No. 329-125 (Mach-Airspeed Warning Switch Installation) incorporated.

34.10.1.7 (Cont).

The overspeed warning switch operates from the third static system and the third source pitot system, and requires a 28-volt, DC power supply. The switch contains an aneroid and an airspeed diaphragm which sense altitude and airspeed respectively. At the limit speed corresponding to a given altitude, the contacts close the 5-amp relay circuit which allows current to flow to the warning bell, which is internally grounded.

34.10.1.8 PITOT-STATIC SYSTEM. (See figures 34-23 and 32-24). Two pitot heads on the underside of the nose of the airplane provide pitot pressure. The pitot head on the left side provides pitot pressure for the pilot's pitot system which consists of the pilot's pitot pressure lines connected to the pilot's airspeed indicator and to the mach meter. The pitot head on the right side provides pitot pressure for the copilot's pitot and the third source pitot system. The copilot's pitot system is the copilot's pitot pressure lines connected to the copilot's airspeed indicator.

The third source pitot system is controlled through a manually operated shutoff valve. A two-position ("OPEN" "CLOSE") AUTOPILOT & "Q" SWITCH PITOT SHUTOFF handle on the copilot's inclined panel, controls this valve. Pitot pressure is routed through this valve to the overspeed warning switch, the mach trim compensator control unit, and the autopilot airspeed differential pressure switch.

Two flush static ports are located on each side of the airplane at fuselage station 161. The top ports on each side of the airplane provide static pressure to the pilot's static system which consists of the pilot's static pressure lines connected to the pilot's altimeter, vertical velocity indicator, airspeed indicator, and the mach meter. The bottom ports on each side of the airplane provide static pressure to the copilot's static system which consists of the copilot's static pressure lines connected to the copilot's altimeter, vertical velocity indicator, and airspeed indicator.

Three flush static ports are located on each side of the airplane at fuselage station 294. The middle ports on each side of the airplane provide static pressure to the third static system which consists of third static system pressure lines connected to the overspeed warning switch, the dual altimeter and differential pressure indicator (see Chapter 21), the autopilot airspeed differential pressure switch (see Chapter 22), the autopilot altitude control (see Chapter 22), the cabin high differential pressure warning switch (see Chapter 21), the mach trim compensator control unit (see Chapter 27), and the stall warning altitude pressure switches for systems No. 1 and No. 2 (see Chapter 27). The top ports on each side of the airplane provide static pressure for the airplane pressurization system (Chapter 21). On aircraft 5126 and subsequent, the upper port is omitted on the left-hand side of the aircraft. Provisions are retained in the heater assembly for port installation if required. The bottom ports on each side of the airplane are provided for customer installed instrumentation in the cabin area. The airplane is delivered with the bottom ports plugged. Refer to Section 30.4 for anti-icing of the pitot heads and the static ports.

34.10.1.9 TURN AND SLIP INDICATING SYSTEM. The turn and slip indicating system is composed of a turn and slip indicator and a turn and slip power indicator on both the pilot's and copilot's side of the main instrument panel. The system presents information which can be used for reference in keeping the airplane level laterally, in maintaining straight flight, and in establishing the proper angle of bank for a given rate of turn. DC power at 28 volts is supplied to the appropriate indicator through a PILOT TURN AND SLIP IND circuit breaker on the pilot's side circuit breaker panel. The turn and slip indicator is a four-minute-turn, gyroscope driven, and 28-volt, DC power operated instrument. It is comprised of two separate instruments, operating independently of each other, but mounted in the same case. The slip indicator portion is comprised of a curved glass tube in which is placed a white ball-type inclinometer. The inside of the tube is filled with clear fluid which serves to deaden vibrations between the ball and the tube and to assure smooth, accurate movements of the ball. During level flight, the ball remains centered in the glass tube. When a low-wing condition exists, the ball moves toward the low wing side and remains in the lowest part of the tube until a level attitude is established by the airplane. When a perfectly banked turn is being made, the two forces (gravity and centrifugal), which act on the ball, are equalized, and the ball stays at the center of the tube. However, if one wing is lower than it should be for the turn, gravity forces the ball to roll to the side of the low wing. If the turn is not steep enough, centrifugal force pulls the ball towards the higher wing. The turn indicator portion consists of an electrically driven gryo, mechanically linked to a pointer which indicates whether or not the airplane is turning, and in which direction and at what rate the turn is being made. The pointer moves from its neutral position at the top of the dial to either side of the dial, depending upon the direction of the turn. Thus, if the airplane is making a left turn, the pointer moves to the left of the dial. The turn and slip power indicator is a D'Arsonval-type DC voltmeter with a lance-type pointer which indicates on a scale (0 to 30-volts) arc-type dial. The primary function is to monitor the DC electrical power supply to the turn and slip indicator in terms of volts. The turn and slip power indicator is connected to the electrical wiring of the turn and slip indicator in such a way as to enable an accurate measurement of DC power to be reported to both the pilot and copilot. If the pointer is registering in the green sector of the arc, the turn and slip indicator is receiving an adequate DC power supply to run the electrically driven gyro. If the pointer is out of the green sector of the arc, the turn and slip indicator is not receiving sufficient DC power for the turn indicator portion to accurately reflect the rate at which the turn is being made and whether or not the airplane is turning.

34.10.1.10 VERTICAL VELOCITY (RATE-OF-CLIMB) INDICATORS. One vertical velocity indicator is on the pilot's side of the main instrument panel and is connected to the pilot's static system. Another

34.10.1.10 (Cont).

vertical velocity indicator is on the copilot's side of the main instrument panel and is connected to the copilot's static system. Each indicator measures the rate of change of the atmospheric pressure. This measurement is done by admitting static pressure directly into the interior of a diaphragm, while the exterior of the diaphragm is acted upon by the pressure within the instrument case. The same static pressure that is fed into the interior of the diaphragm is also fed into the case of the indicator, through restrictions, so the change of pressure in the case lags the change of pressure in the diaphragm. The resulting differential pressure on the diaphragm actuates the pointer. A mechanical linkage from the diaphragm translates the differential pressure into a calibrated pointer deflection. The dial face is logarithmically graduated from zero to 6,000 feet per minute of vertical velocity, either up or down. A setscrew at the lower left-hand corner of the indicator is used to zero the pointer and should not be disturbed during flight. If altitude changes rapidly, an overspeed diaphragm opens a valve to admit static pressure into the case to immediately reduce the differential pressure and thereby avoid possible damage to the indicator. An insulated instrument case minimizes the effects of temperature change.

34.10.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Equipment	Description	Use and Application
Auxiliary Power Unit	28V DC Electrical Power Type MD-3A	To energize circuits.
Pitot-Static Test Unit	A source of pitot pressure and a source of static pressure/vacuum.	For tube leakage test and for instrument and equipment calibration checkout tests.
Centigrade Thermometer	Certified and calibrated in centigrade from -70° to +50°	To record temperature outside airplane.
Voltmeter	To measure 28 Volts DC with an accuracy of 5 percent	To measure voltage at test points during checkout.

34.10.2.1 AIRSPEED INDICATORS. (Refer to 34.10.2.6.)

34.10.2.2 ALTIMETERS. (Refer to 34.10.2.6.)

34.10.2.3 MACHMETER. The following procedure can be used to test the mach meter for operation and accuracy:

a. Perform step 34.10.2.6.2. a and slowly apply a pressure of 10.89 inches of mercury absolute.
RESULTS: The mach meter should indicate .68 (±.05) mach assuming a condition of 0 feet altitude during a normal day of 15° centigrade and ambient barometric pressure of 29.92 inches of mercury.

Note

Mach number changes in relationship to ambient barometric pressure and temperature. Thus, if the mach meter is checked at conditions other than those stated, recalculate the mach number using the following formula:

$$\frac{\Delta P}{P} = \left(1 + \frac{M^2}{5}\right)^{\frac{7}{2}} - 1$$

ΔP = Differential pressure corresponding to given airspeed.

P = Pressure corresponding to given altitude (same unit as ΔP).

M = Mach number

b. Allow the system to bleed to ambient pressure, and restore the system to normal.

34.10.2.4 OUTSIDE AIR TEMPERATURE INDICATING SYSTEM. The following procedure can be used to test the outside air temperature indicating system:

a. Connect the auxiliary power unit to the airplane external 28 volt, DC electrical power receptacle.

b. Close the EXT AIR TEMP circuit breaker on the copilot's side circuit breaker panel.

c. Place test thermometer near the flush mounted bulb, located on the fuselage skin forward of the main entrance door and below the clear vision window. Allow the thermometer to remain in position for approximately five minutes. Record the temperature and compare with the outside air temperature indicator.
RESULTS: The outside air temperature indicator should read within ±10°C (18°F) of the test thermometer.

34.10.2.5 OVERSPEED WARNING SYSTEM. (Refer to 34.10.2.6.)

34.10.2.6 PITOT-STATIC SYSTEMS. The following operational checkout procedure may be used to assure proper operation of the pilot's pitot and static system, the copilot's pitot and static system, and the third pitot and static system.

CAUTION

Vacuum must not be applied to pitot pressure components unless an equivalent vacuum is applied, simultaneously, to the corresponding system's static pressure components. Pressure must not be applied to the static systems. Any application or release of pressure or vacuum must not be made at a rate of more than 3,000 ft. per minute. If any components have been removed, the tubing must be securely capped.

34.10.2.6.1 PILOT'S PITOT-STATIC SYSTEM LEAK TEST.

a. Set test unit altimeter and pilot's altimeter to indicate 29.92 inches of mercury.

Note

Lightly tap the instruments before taking readings.

RESULTS: The pilot's altimeter and the test unit altimeter should agree within 20 feet.

b. Connect a suction sleeve over the pilot's pitot head (left-hand side), and seal the drain hole in the pitot head.

c. Connect a suction cup over the pilot's static port (top port) on the left-hand side, and seal the pilot's static port (top port) on the right-hand side of the airplane.

d. Connect the suction sleeve (pitot head) and the suction cup (static port) to the vacuum system of the pitot-static system test unit.

e. Slowly evacuate the system until the test unit altimeter indicates 25,877 feet (10.68 inches of mercury absolute) altitude. Allow the vacuum to stabilize for 2 minutes.
RESULTS:
1. Pilot's rate of climb indicator should indicate an "up" condition during altitude increase.
2. Pilot's airspeed indicator should not indicate an increase in speed.
3. Pilot's altimeter and test unit altimeter should agree within 160 feet.
4. Machmeter should not indicate an increase in speed.

f. Close the vacuum supply valve on the test unit.
RESULTS: The pilot's altimeter should not change more than 518 feet within the next minute, and the altitude indicator should not rise.

g. Allow the system to bleed to ambient pressure, and restore the system to normal.

34.10.2.6.2 PILOT'S PITOT SYSTEM.

a. Connect a calibrated pressure source from the test unit to the pilot's pitot head (left-hand side), and seal the drain hole in the pitot head.

b. Slowly apply pressure until the test airspeed indicator shows 425 knots (9.59 inches of mercury absolute). Allow the pressure to stabilize for 2 minutes.

Note

Lightly tap the instruments before taking readings.

RESULTS: The pilot's airspeed indicator and the test unit's airspeed indicator should agree within 4 knots.

c. Close the pressure supply valve.
RESULTS: The pilot's airspeed indicator should not change more than 3 knots within the next minute.

d. Decrease the pressure until the test unit's airspeed indicator shows 300 knots (4.54 inches of mercury absolute). Allow the pressure to stabilize for 2 minutes.
RESULTS: The pilot's airspeed indicator and the test unit's airspeed indicator should agree within 4 knots.

e. Allow the system to bleed to ambient pressure, and restore the system to normal.

34.10.2.6.3 PILOT'S STATIC SYSTEM.

a. Set the test unit's altimeter and the pilot's altimeter at 29.92 inches of mercury.

Note

Lightly tap the instruments before taking readings.

RESULTS: The pilot's altimeter and the test unit's altimeter should agree within 20 feet.

b. Connect a calibrated vacuum source from the test unit to the left-hand pilot's static port (top port), and seal the right-hand pilot's static port (top port).

c. Slowly evacuate the system until the test unit's altimeter indicates 14,000 feet (17.57 inches of mercury absolute). Allow the vacuum to stabilize for 2 minutes.
RESULTS:
1. Pilot's rate of climb indicator should indicate an "up" condition during altitude increase.
2. Pilot's airspeed indicator should indicate an increase in speed.

34.10.2.6.3 (Cont).

3. Machmeter should indicate an increase in speed.
4. Pilot's altimeter and the test unit's altimeter should agree within 100 feet.

d. Close the vacuum supply valve.
RESULTS: The pilot's altimeter should not drop more than 200 feet within the next minute.

e. Decrease the vacuum until the test unit's altimeter indicates 8,000 feet (22.22 inches of mercury absolute). Allow the vacuum to stabilize for 2 minutes.
RESULTS:
1. Pilot's rate of climb indicator should indicate a "down" condition during altitude decrease.
2. Pilot's airspeed indicator and the machmeter should indicate a decrease in speed.
3. Pilot's altimeter and the test unit's altimeter should agree within 60 feet.

f. Allow the system to bleed to ambient pressure.
RESULTS:
1. Pilot's rate of climb indicator should indicate a "down" condition during altitude decrease.
2. Pilot's airspeed indicator and the machmeter should indicate a decrease in speed.
3. Pilot's altimeter should indicate a decrease in altitude.

g. Remove the seal from the pilot's right-hand static port, and apply a vaccum to the pilot's left-hand static port.
RESULTS: A vacuum build-up should not occur.

h. Remove the vacuum source from the pilot's left-hand static port, and restore the system to normal.

34.10.2.6.4 COPILOT'S PITOT-STATIC SYSTEM LEAK TEST.

a. Perform step 34.10.2.6.1.a except use copilot's altimeter in lieu of pilot's.

b. Connect a suction sleeve over the copilot's pitot head (right-hand side), and seal the drain hole in the pitot head.

c. Connect a suction cup over the copilot's static port (bottom port) on the right-hand side, and seal the copilot's static port (bottom port) on the left-hand side of the airplane.

CAUTION

The AUTO-PILOT & "Q" SWITCH PITOT SHUT-OFF valve must be in the "CLOSE" position.

d. Perform steps 34.10.2.6.1.d through 34.10.2.6.1.g except use copilot's instruments instead of pilot's and delete machmeter requirements.

34.10.2.6.5 COPILOT'S PITOT SYSTEM

CAUTION

The AUTO-PILOT & "Q" SWITCH PITOT SHUT-OFF valve must be in the "CLOSE" position.

a. Connect a calibrated pressure source from the test unit to the copilot's pitot head (right-hand side), and seal the drain hole in the pitot head.

b. Perform steps 34.10.2.6.2.b through 34.10.2.6.2.e except use copilot's instruments instead of pilot's and delete machmeter requirements.

34.10.2.6.6 COPILOT'S STATIC SYSTEM

a. Set the test unit's altimeter and the copilot's altimeter at 29.92 inches of mercury.

Note

Lightly tap the instruments before taking readings.

RESULTS: The copilot's altimeter and the test unit's altimeter should agree within 20 feet.

b. Connect a calibrated vacuum source from the test unit to the right-hand copilot's static port (bottom port), and seal the left-hand copilot's static port (bottom port).

c. Perform steps 34.10.2.6.3.c through 34.10.2.6.3.h except use copilot's instruments and delete machmeter requirements.

34.10.2.6.7 THIRD PITOT SYSTEM

Note

Steps e, f, g, h, i.2, and o apply only to aircraft equipped with the Sperry SP-40 Autopilot system.

a. Connect the auxiliary power unit to the airplane external 28-volt, DC electrical power receptacle, and energize the MAIN DC BUS with 28-volts.

b. Place the AUTOPILOT AND "Q" SWITCH PITOT SHUT-OFF valve in the "OPEN" position.

c. Perform step 34.10.2.6.5.a.

d. Engage the following circuit breakers:

1. OVERSPEED WARNING	Copilot wainscot panel (forward)
2. AUTOPILOT DC	Copilot wainscot panel (aft)

e. Remove the autopilot computer from the nose electronic equipment mounting rack.

f. Connect the voltmeter positive lead to pin 22 of the mounting rack connector and the voltmeter negative lead to ground.
RESULTS: The voltage at pin 22 should read 28 V DC.

34.10.2.6.7 (Cont).

g. Slowly apply pressure until the test unit's airspeed indicator indicates 200 knots. Allow the pressure to stabilize for 2 minutes.
RESULTS: The voltage at pin 22 should disappear between 195 and 205 knots.

h. Connect the voltmeter positive lead to pin 77 of the mounting rack connector and the voltmeter negative lead to ground.
RESULTS: There should be no voltage.

i. Slowly increase the pressure until the test unit's airspeed indicator indicates 300 knots. Allow the pressure to stabilize for 2 minutes.
RESULTS:
1. The copilot's airspeed indicator and the test unit's airspeed indicator should agree within 8 knots.
2. The voltage at pin 77 should read 28V DC.

j. Close the pressure supply valve.
RESULTS: The copilot's airspeed indicator should not drop more than -2 knots in the next minute.

k. Press the OVERSPEED WARNING TEST switch on the copilot's incline panel.
RESULTS: The warning bell should sound. Release the switch and the bell should silence.

l. Slowly increase the pressure until the test unit's airspeed indicator shows 350 knots. Allow the pressure to stabilize for 2 minutes.
RESULTS: The copilot's airspeed indicator and the test unit's airspeed indicator should agree within 8 knots.

m. Slowly increase the pressure.
RESULTS: The overspeed warning bell should sound between 356 and 363 knots.

n. Slowly decrease the pressure.
RESULTS: The overspeed warning bell should silence before 352 knots.

Note

The indicated airspeed values of steps m and n include airspeed indicator and pitot-static system error. Steps m and n are applicable on airplanes 5046 and up, and on airplanes 5001 through 5045 with the intent of Service Bulletin 329-125 (Mach-Airspeed Warning Switch Installation) incorporated.

o. Slowly decrease the pressure.
RESULTS: The voltage at pin 77 of the mounting rack connector should disappear at 5 to 15 knots less than the actuation point of increasing pressure (ref. step h) and should appear at pin 22 at 5 to 15 knots less than the previous actuation point (ref. step f).

p. Perform step 34.10.2.6.2.e.

Note

See 27.3.2.5 for mach trim compensator system operational checkout procedure.

34.10.2.6.8 THIRD STATIC SYSTEM LEAK TEST

a. Set the test unit's altimeter at 29.92 inches of mercury.

b. Connect a calibrated vacuum source from the test unit to the left-hand third static port (middle port), and seal the right-hand third static port (middle port).

c. Slowly evacuate the system until the test unit's altimeter indicates 8,000 feet. Allow the vacuum to stabilize for 2 minutes.

d. Close the vacuum supply valve.
RESULTS: The test unit's altimeter should not change more than 350 feet within the next minute.

CAUTION

Never evacuate the third static system beyond 8,000 feet simulated altitude with the over-warning switch, the autopilot system components, and the mach trim compensator component connected to the system tubing.

e. Allow the system to bleed to ambient pressure.

f. Remove the seal from the right-hand third static system port (middle port), and apply a vacuum to the left-hand third static system port (middle port).
RESULTS: A vacuum build-up should not occur.

g. Remove the vacuum source from the left-hand third static system port (middle port), and restore the system to normal.

Note

Continue with step 34.10.2.6.9 if a complete third static system operational checkout is to be performed.

34.10.2.6.9 THIRD STATIC SYSTEM. Perform steps 34.10.2.6.8.a, 34.10.2.6.8.b, and 34.10.2.6.7.a.

34.10.2.6.9.1 AUTOPILOT SYSTEM COMPONENTS. (Refer to 22.0.2.1.ba.)

a. Engage the following circuit breakers:

1. VERT GYRO NO. 2 AC

2. VERT GYRO NO. 2 DC

34.10.2.6.9.1 (Cont).

3. AUTOPILOT AC

4. AUTOPILOT DC

b. Energize the AC buses.

CAUTION

Remove control surface locks.

c. Slowly evacuate the system until the test unit's altimeter indicates 5,000 feet.

d. Position the AFCS switch in the "ENG" position, and depress the ALT mode selector switch (switches are located on the autopilot control panel in the flight station's center console).
RESULTS: The ALT mode selector switch should engage and its light should illuminate.

CAUTION

After the autopilot ALT mode selector switch has been engaged, DO NOT change the simulated altitude more than 100 feet without disengaging ALT mode, to avoid damage to the altitude control unit.

e. Slowly start increasing the vacuum. DO NOT increase beyond 5,100 feet.
RESULTS: The pilot's and copilot's control column should move forward.

f. Slowly start decreasing the vacuum. DO NOT decrease beyond 4,900 feet.
RESULTS: The pilot's and copilot's control column should move aft.

g. Place the PITCH control in the "UP" position and allow the system to bleed to ambient pressure.
RESULTS: The ALT mode selector switch should disengage and its light should extinguish.

h. Restore airplane to step 34.10.2.6.9 condition if a complete third static system operational checkout is to be performed. If a complete autopilot system operational checkout is to be performed continue with step 22.0.2.1.bb.

34.10.2.6.9.2 DUAL ALTIMETER AND DIFFERENTIAL PRESSURE INDICATOR.

a. Slowly evacuate the system until the test unit's altimeter indicates 8,000 feet. Allow the vacuum to stabilize for 2 minutes.
RESULTS:
1. The "FLIGHT" pointer of the dual altimeter and differential pressure indicator and the test unit's altimeter should agree within 350 feet.
2. The "CABIN" pointer of the dual altimeter and differential pressure indicator and the pilot's and copilot's altimeter should agree within 200 feet.

b. Allow the system to bleed to ambient pressure which returns the airplane to step 34.10.2.6.9 condition.

34.10.2.6.9.3 CABIN HIGH DIFFERENTIAL PRESSURE SWITCH.

CAUTION

Disconnect and plug the third static system tubing at the following components:

1. Autopilot's altitude control unit (Chapter 22).
2. Autopilot airspeed differential pressure switch (Chapter 22).
3. The Overspeed Warning Switch (Chapter 34.10).
4. The Mach Trim Compensator Control Unit (Chapter 27.3).
5. The Stall Warning Altitude Pressure Switches No. 1 and No. 2 (Chapter 27.6).

a. Engage the WARNING INTERIOR LIGHTS circuit breaker which is located on the pilot's side console, center panel.

b. Slowly evacuate the system until the "CABIN PRESSURE HIGH" indication on the annunciator panel illuminates.
RESULTS: The test unit's gage should indicate a differential pressure of 18.3 to 19.3 inches of mercury absolute.

c. Slowly start decreasing the vacuum.
RESULTS: The "CABIN PRESSURE HIGH" indication on the annunciator panel should extinguish between a differential pressure of 18.3 to 17.3 inches of mercury absolute.

d. Allow the system to bleed to ambient pressure which returns the airplane to step 34.10.2.6.9 condition.

34.10.2.6.9.4 STALL WARNING ALTITUDE PRESSURE SWITCHES NO. 1 AND NO. 2 (refer to 27.6.2.1.t).

34.10.2.6.9.5 MACH TRIM COMPENSATOR CONTROL UNIT (refer to 27.3.2.5).

34.10.2.6.10 RESTORING SYSTEM TO NORMAL.

a. Allow the system to bleed to ambient pressure.

34.10.2.6.10 (Cont).

b. Remove the seal from the right-hand third static system port (middle port), and remove the vacuum source from the left-hand third static system port (middle port).

c. Disconnect the auxiliary power unit.

d. Check all system component connections.

e. Restore the system to normal.

34.10.2.7 TURN AND SLIP INDICATING SYSTEM. The following procedure can be used to test the system for proper operation:

a. Position the airplane on a level parking area.

b. Connect the auxiliary power unit to the airplane external 28-volt, DC electrical power receptacle.

c. Close the PILOT TURN AND SLIP IND. Circuit breaker on the pilot's side circuit breaker panel.
RESULTS: Both the pilot's and copilot's pointer of the turn indicator and the inclinometer ball of the bank indicator should be in the center. The pointer of the pilot's and copilot's turn and slip power indicator should be indicating in the green sector of the dial.

34.10.2.8 VERTICAL VELOCITY (RATE-OF-CLIMB) INDICATORS. The following procedure can be used to test the vertical velocity indicators:

a. Perform steps 34.10.2.6.3.a through 34.10.2.6.3.c.
RESULT: The pointer movement should be steady and smooth.

b. Slowly decrease the vacuum at a constant rate.
RESULT: The pointer movement should be steady and smooth.

c. Repeat steps (a) and (b) for the copilot's vertical velocity (rate of climb) indicator.

d. Restore the system to normal.

34.10.3 TROUBLESHOOTING.

34.10.3.1 AIRSPEED INDICATORS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
AIRSPEED POINTER DOES NOT RESPOND.		
Pitot tubing obstructed or improperly connected.	Check connections.	If connections are correct, disconnect tubes from instruments and clear with compressed air.
AIRSPEED POINTER OSCILLATES OR INDICATES INCORRECTLY.		
Leak in pitot tubing.	Perform pitot-static operational checkout.	Isolate and repair leaks.
Instrument case leak.	Examine instrument case.	Replace instrument.
STRIPED POINTER INDICATES INCORRECTLY.		
Do not attempt to service the instrument for incorrect reading of striped pointer.		

34.10.3.2 ALTIMETERS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER DOES NOT RESPOND.		
Static line obstructed or improperly connected.	Check connections.	If connections are correct, disconnect tubes from instruments and clear with compressed air.
POINTER INDICATES INCORRECTLY.		
Static lines or indicator leaking.	Test for leaks.	Repair tube leak or replace instrument.
Indicator faulty.	If no leaks are found, the indicator is faulty.	Replace instrument.

34.10.3.3 MACH METER.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER DOES NOT RESPOND		
Pitot-static lines obstructed or improperly connected.	Check connections.	If connections are correct, disconnect tubes from instruments and clear with compressed air.
POINTER INDICATES INCORRECTLY.		
Pitot-static lines or indicator leaking.	Test for leaks.	Repair tube leak or replace instrument.
Indicator faulty.	If no leaks are found, the indicator is faulty.	Replace instrument.
STRIPED POINTER INDICATES INCORRECTLY.		
Do not attempt to service the instrument for incorrect reading of striped pointer.		

34.10.3.4 OUTSIDE AIR TEMPERATURE INDICATING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER DOES NOT RESPOND.		
Loose or improper electrical connections.	Check wiring against wiring diagram.	Isolate and repair
Indicator faulty.	If no electrical problems are found, the indicator may be faulty.	Replace instrument
Sensing element faulty.	If no electrical problems are found and no obstructions are found, the element may be faulty.	Replace sensing element.

34.10.3.4 OUTSIDE AIR TEMPERATURE INDICATING SYSTEM. (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER INDICATES INCORRECTLY		
Loose or improper electrical connections.	Check wiring against wiring diagram.	If wiring correct, replace instrument or sensing element.
Sensing element improperly mounted.	Check for flush condition of sensing bulb.	Correct to insure flush mounting with outside skin.
Do not attempt to service the instrument for incorrect reading.		

34.10.3.5 OVERSPEED WARNING SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
WARNING BELL DOES NOT SOUND		
Pitot-static tubing obstructed or improperly connected.	Check connections and tubing.	If connections are correct, disconnect tubes from warning switch and clear with compressed air.
Loose or improper electrical connection.	Check wiring against wiring diagram.	Isolate and repair.
Warning bell obstructed.	Check warning bell for obstructions	Remove obstructions.
Overspeed warning test switch inoperative.	Check continuity.	If failed, replace.
Warning switch inoperative.	Check continuity.	If failed, replace.
Warning switch case leakage.	Test for leaks.	If leaking, replace.
INCORRECT SOUNDING OF WARNING BELL.		
Leak in pitot-static tubing.	Perform pitot-static operational checkout.	Isolate and repair leaks.
Warning switch case leak.	Examine instrument case.	Replace warning switch.
Loose or improper electrical wiring.	Check wiring against wiring diagram.	Isolate and repair.
Do not attempt to service the warning switch for warning bell sounding incorrectly.		

34.10.3.6 TURN AND SLIP INDICATING SYSTEM.

34.10.3.6.1 TURN AND SLIP INDICATOR AND TURN AND SLIP POWER INDICATOR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER DOES NOT RESPOND.		
Loose or improper electrical wiring.	Check wiring against wiring diagram.	Isolate and repair.
Indicator faulty.	Perform an operational checkout.	If failed, replace.

34.10.3.7 VERTICAL VELOCITY INDICATORS.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
POINTER DOES NOT RESPOND.		
Static pressure line obstructed.		Disconnect tubing from instrument and blow clear with compressed air.
POINTER OSCILLATES OR INDICATES INCORRECTLY.		
Leak in static line or instrument.	Check static tubing and/or instrument for leaks.	Replace instrument if leaking. Repair tubing if leaking.

34-10.4 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. In cold weather, drain the tubing daily or after each flight. Cover the pitot heads and static ports to prevent ice from blocking passages. Make sure air passages and drain holes in pitot heads are open before flight. Drain tubing after heating the airplane before flight. When sand or dust is blowing, cover pitot heads and static ports while the airplane is on the ground. Do not allow any foreign matter to enter tubing.

34.10.5 REMOVAL AND REPLACEMENT.

34.10.5.1 REMOVAL OF STATIC PORT HEATERS.

a. Disengage four pitot heater circuit breakers.

b. Remove static tubes from static port fitting.

c. Remove static port fittings by removing four screws in each fitting.

d. Remove static port heater ground wire from stud.

e. Cut static port heater power wire and remove static port heater.

34.10.5.2 REPLACEMENT OF STATIC PORT HEATERS.

a. Clean sealant from static port fitting and inside skin of airplane with clean cloths saturated with TT-N-95 aliphatic naphtha. Wipe surfaces dry with additional clean cloths. Do not allow naphtha to air dry.

b. Seal between skin, heater, and fittings and around edges with a uniform coat of MIL-S-8802 sealant and install heater and fittings while wet. (See figure 34-24A.)

c. Fill over screw heads on outer skin with aerodynamic smoothing compound and buff smooth with skin contour.

d. Reconnect static tubes to static port fittings.

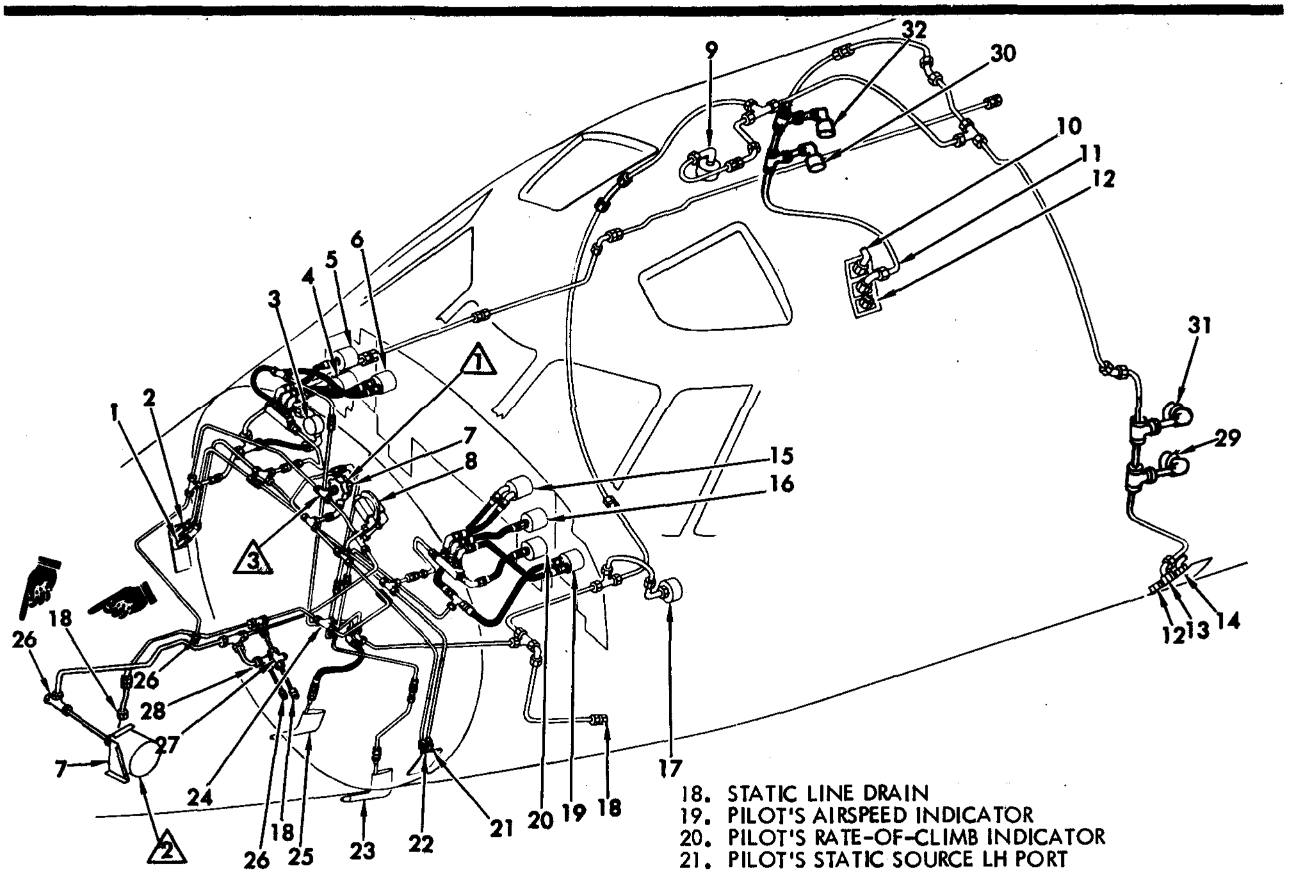
e. Identify heater wires with vinyl sleeves per wiring diagram JR119.

f. Connect ground wire to ground stud and splice the other wire to power wire that was cut in step 34.10.5.1.e.

g. Perform operational checkout, as necessary, per paragraph 30.4.2 and 34.10.2.6 when sealant is sufficiently cured.

34.10.5.2 (Cont).

h. Curing time for class of sealant used as indicated on figure 51-6 should be used. See figure 34-24A for notes and flushness requirements.



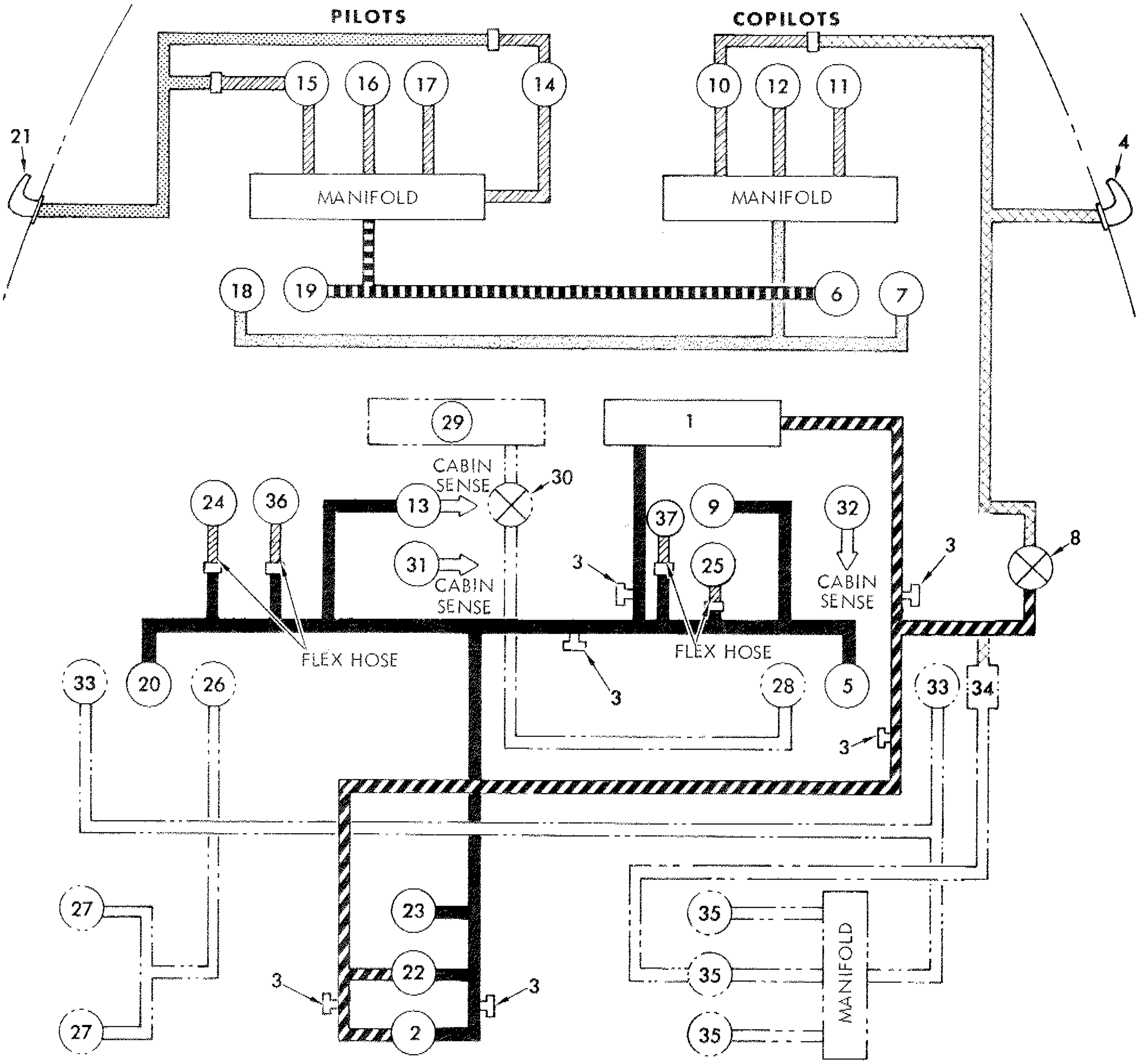
1. COPILOT'S STATIC SOURCE RH PORT
2. PILOT'S STATIC SOURCE RH PORT
3. AUTOPILOT & "Q" SWITCH PITOT SHUTOFF VALVE
4. COPILOT'S ALTIMETER
5. COPILOT'S RATE-OF-CLIMB INDICATOR
6. COPILOT'S AIRSPEED INDICATOR
7. OVERSPEED WARNING SWITCH
8. AUTOPILOT AIRSPEED DIFFERENTIAL PRESSURE SWITCH
9. DUAL INDICATOR
10. CABIN PRESSURE SYSTEM STATIC PORT
11. THIRD STATIC SOURCE RH PORT
12. CAPPED STATIC PORT
13. THIRD STATIC SOURCE LH PORT
14. CABIN SAFETY VALVE STATIC PORT
15. MACH METER
16. PILOT'S ALTIMETER
17. CABIN HIGH DIFFERENTIAL PRESSURE WARNING SWITCH
18. STATIC LINE DRAIN
19. PILOT'S AIRSPEED INDICATOR
20. PILOT'S RATE-OF-CLIMB INDICATOR
21. PILOT'S STATIC SOURCE LH PORT
22. COPILOT'S STATIC SOURCE LH PORT
23. PILOT'S PITOT HEAD
24. STATIC LINE CONNECTION TO AUTOPILOT ALTITUDE CONTROL
25. COPILOT'S PITOT HEAD
26. PITOT LINE DRAIN
27. STATIC LINE CONNECTION TO MACH TRIM COMPENSATOR CONTROL UNIT
28. PITOT LINE CONNECTION TO MACH TRIM COMPENSATOR CONTROL UNIT
29. ALTITUDE PRESSURE SWITCH, STALL WARNING SYSTEM NO. 1, 25,000 FT △4 AND 20,000 FT △5
30. ALTITUDE PRESSURE SWITCH, STALL WARNING SYSTEM NO. 2, 25,000 FT △4 AND 20,000 FT △5
31. ALTITUDE PRESSURE SWITCH, STALL WARNING SYSTEM NO. 1, 32,500 FT △5 .
32. ALTITUDE PRESSURE SWITCH STALL WARNING SYSTEM NO. 2, 32,500 FT △5 .

NOTE

△1 AIRPLANES 5001 THRU 5045 NOT HAVING SERVICE BULLETIN 329-125 INCORPORATED.
△2 AIRPLANES 5046 AND UP AND AIRPLANES 5001 THRU 5045 HAVING SERVICE BULLETIN 329-125 INCORPORATED.
△3 TEES CAPPED OFF ON AIRPLANES 5001 THRU 5045, WITH SERVICE BULLETIN 329-125 INCORPORATED.
△4 AIRPLANES 5001 THRU 5092, 5094 THRU 5096 NOT HAVING SERVICE BULLETIN 329-230 INCORPORATED.
△5 AIRPLANES 5093, 5097 AND UP AND AIRPLANES 5001 THRU 5092, 5094 THRU 5096 HAVING SERVICE BULLETIN 329-230 INCORPORATED.

JS-34-0-023

Figure 34-23. Pitot-Static System Plumbing



JS-34-0-024-1

Figure 34-24. Pitot Static System Schematic Diagram (Sheet 1 of 2)

1. MACH TRIM COMPENSATOR CONTROL UNIT
2. OVERSPEED WARNING SWITCH
3. LINE DRAIN
4. COPILOT'S PITOT HEAD
5. THIRD STATIC SOURCE RH PORT
6. PILOT'S STATIC SOURCE RH PORT
7. COPILOT'S STATIC SOURCE RH PORT
8. AUTOPILOT & "Q" SWITCH PITOT SHUTOFF VALVE
9. CABIN HIGH DIFFERENTIAL PRESSURE WARNING SWITCH
10. COPILOT'S AIRSPEED INDICATOR
11. COPILOT'S RATE-OF-CLIMB INDICATOR
12. COPILOT ALTIMETER
13. DUAL ALTIMETER AND DIFFERENTIAL PRESSURE IND.
14. MACH METER
15. PILOT'S AIRSPEED INDICATOR
16. PILOT'S ALTIMETER
17. PILOT'S RATE-OF-CLIMB INDICATOR
18. COPILOT'S STATIC SOURCE LH PORT
19. PILOT'S STATIC SOURCE LH PORT
20. THIRD STATIC SOURCE LH PORT
21. PILOT'S PITOT HEAD
22. AUTOPILOT AIRSPEED DIFFERENTIAL PRESSURE SWITCH
23. AUTOPILOT ALTITUDE CONTROL
24. ALTITUDE PRESSURE SWITCH, 25,000 FT, STALL WARNING SYSTEM NO. 1 △1
    ALTITUDE PRESSURE SWITCH, 20,000 FT, STALL WARNING SYSTEM NO. 1 △2
25. ALTITUDE PRESSURE SWITCH, 25,000 FT, STALL WARNING SYSTEM NO. 2 △1
    ALTITUDE PRESSURE SWITCH, 20,000 FT, STALL WARNING SYSTEM NO. 2 △2
26. CABIN SAFETY VALVE STATIC PORT (SEE CHAPTER 21) △3
27. CABIN SAFETY VALVE, PRESSURIZATION SYSTEM (SEE CHAPTER 21)
28. CABIN PRESSURE SYSTEM STATIC PORT (SEE CHAPTER 21)
29. CABIN PRESSURE CONTROLLER (SEE CHAPTER 21)
30. DIFFERENTIAL SHUTOFF TEST VALVE (SEE CHAPTER 21)
31. CABIN RATE OF CLIMB INDICATOR
32. CABIN PRESSURE INDICATOR
33. STATIC SOURCE PROVISIONS FOR CABIN INSTRUMENTS.
34. PITOT SOURCE PROVISIONS FOR CABIN INSTRUMENTS.
35. CABIN INSTRUMENTS
36. ALTITUDE PRESSURE SWITCH, 32,500 FT, STALL WARNING SYSTEM NO. 1 △2
37. ALTITUDE PRESSURE SWITCH, 32,500 FT, STALL WARNING SYSTEM NO. 2 △2

NOTE

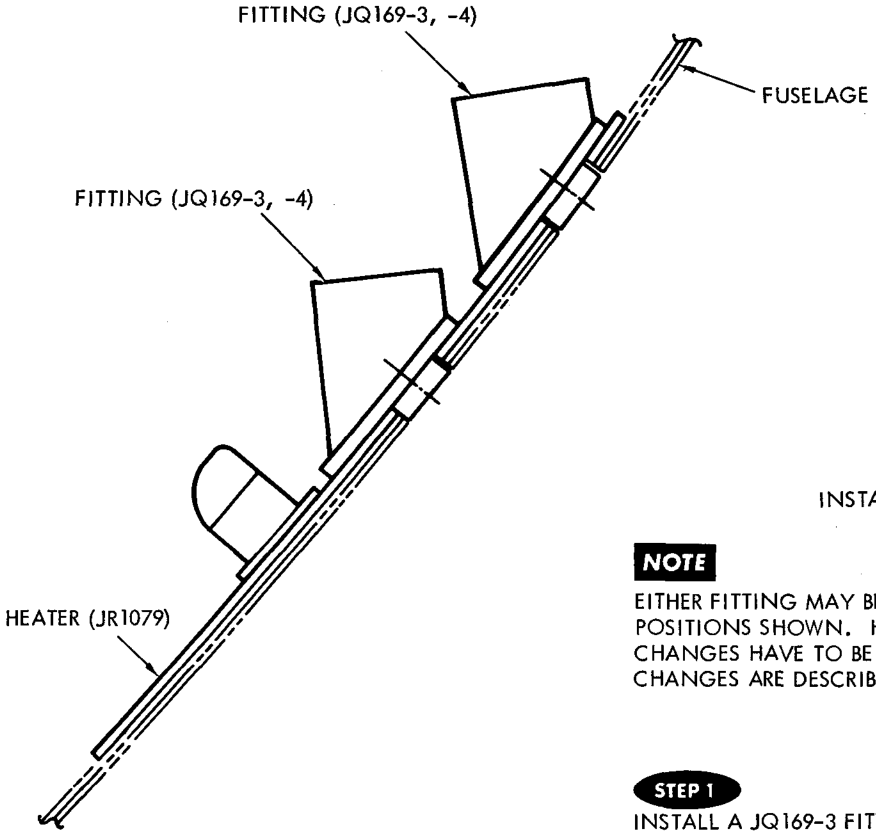
△1 AIRPLANES 5001 THRU 5092, 5094 THRU 5096 NOT HAVING SERVICE BULLETIN 329 -230 INCORPORATED

△2 AIRPLANES 5093, 5097 & UP & AIRPLANES 5001 THRU 5092, 5094 THRU 5096 HAVING SERVICE BULLETIN 329 - 230 INCORPORATED.

△3 AIRPLANES 5001 THRU 5125

■ 3RD STATIC SOURCE
▥ PILOT'S STATIC
▦ COPILOT'S PITOT PRESSURE
▨ FLEX HOSE
░ COPILOT'S STATIC
▒ PILOT'S PITOT PRESSURE
▧ 3RD PITOT SOURCE

Figure 34-24. Pitot Static System Schematic Diagram (Sheet 2 of 2)



INSTALLATION

NOTE

EITHER FITTING MAY BE INSTALLED IN THE POSITIONS SHOWN. HOWEVER, MINOR CHANGES HAVE TO BE MADE. THESE CHANGES ARE DESCRIBED BELOW.

STEP 1

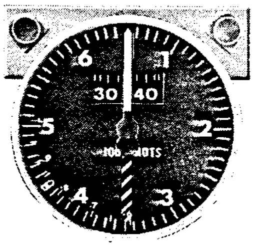
INSTALL A JQ169-3 FITTING FLUSH WITH THE FUSELAGE CONTOUR WITHIN $^{+0.002}_{-0.001}$ INCH BY USING ALUMINUM SHIM STOCK AS REQUIRED.

STEP 2

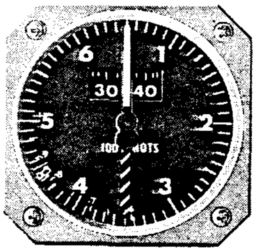
INSTALL A JQ169-4 FITTING FLUSH WITH THE FUSELAGE CONTOUR WITHIN $^{+0.003}_{-0.000}$ INCH BY MICRO-SHAVING THE BOSS TO NOTED FINISH. AFTER MICRO-SHAVING, APPLY A CHEMICAL FILM PROTECTION AS DIRECTED IN SECTION 51.7 IN THIS MANUAL.

Figure 34-24A. Static Port Heaters Installation

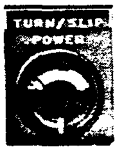
JS-1-2-X0/2-3424A



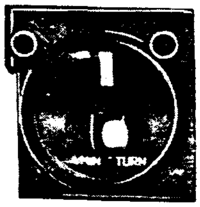
AIRSPEED INDICATOR AIRPLANES 5001 THROUGH 5045



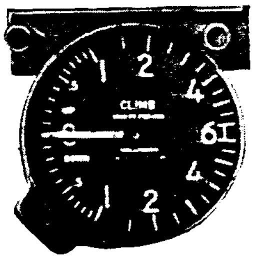
AIRSPEED INDICATOR AIRPLANES 5046 AND UP



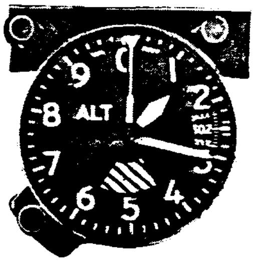
TURN AND SLIP POWER INDICATOR



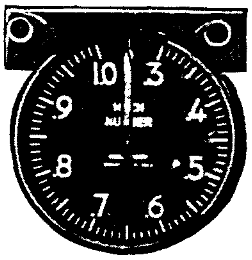
TURN AND SLIP INDICATOR



RATE OF CLIMB INDICATOR



ALTIMETER



MACH METER

Figure 34-25. ***Air Data and Directional Instruments***

SECTION 34.11
ALTERNATE ELECTRONIC SYSTEMS
Airplanes: 5050, 5051, 5053 thru 5055, 5057, 5058, 5060 & up

Contents **Page**

DESCRIPTION 34-98A
General 34-98A
Provisions for Equipment 34-98A
Antennas 34-98A
FD-108 Flight Director System 34-98A
Interchangeability of Systems 34-98A
Spare Wiring 34-98B
List of alternate Systems for Which Complete Provisions are Installed 34-98B

34.11.1 DESCRIPTION

34.11.1.1 GENERAL. The installation of provisions for "solid state" and other advanced electronic systems, the installation of provisions to accept equipment available from different manufacturers, and the installation of "spare" wiring to minimize rework to the airplane if additional equipment is added, have given the owners of the above airplanes a wide selection of equipment which may be installed. Due to the variations in choice of equipment to be installed in these airplanes, no attempt has been made in this handbook to cover the operation and maintenance of systems where these features exist. For operating and maintenance information refer to the equipment manufacturers handbook applicable to the equipment chosen for installation in the airplane.

34.11.1.2 PROVISIONS FOR EQUIPMENT. Complete provisions, including wiring, antennas, and mounting facilities are installed for the electronic systems listed below. These provisions are suitable for equipment available from several different manufacturers.

QTY	DESCRIPTION
2	VHF Communication
2	VHF Navigation (VOR-ILS)
1	Distance Measuring System (DME)
1	Marker Beacon
2	Automatic Direction Finder (ADF)
1	Air Traffic Control (ATC)
1	Weather Radar
2	Flight Director

34.11.1.3 ANTENNAS. All antennas shall be installed as part of the equipment provisions except the weather radar antenna and the ADF loop antennas.

34.11.1.4 FD-108 FLIGHT DIRECTOR SYSTEM. As noted in 34.11.1.7.H, complete provisions shall be provided for two basic FD-108 Flight Director systems. Partial provisions for additional FD-108 components that may be added in the future are installed as outlined below.

> Separate Mode Selector. Complete wiring provisions are terminated inside the center stand with the appropriate connector.
>
> Altitude Hold Units: Space provisions only are provided in the nose equipment rack. Spare wiring described in paragraph 34.11.1.6 may be utilized for these units.
>
> Control-Wheel-Mounted "Speed Control" Switch: Complete wiring shall be provided through the control wheels.
>
> Collins 54W-1 Comparator System: Spare wiring as described in paragraph 34.11.1.6 is designed to be adequate for this system.

34.11.1.5 INTERCHANGEABILITY OF SYSTEMS. All systems listed as "Alternate No. 1" in paragraph 34.11.1.7 have been installed and tested by the Lockheed Aeronautical Systems Company in a flight test aircraft. The JetStar provisions have also been designed to accommodate all systems listed in alternates 2 and 3 in paragraph 34.11.1.7, although these systems may not have been flight tested by the Lockheed Aeronautical Systems Company. It should be noted that only one system is listed for each manufacturer. It may be possible to install any one of the several models of the particular manufacturer's system without change to aircraft wiring. For instance, aircraft wiring is suitable for VHF communication systems that have expanded frequency ranges or 25 KC frequency spacing, and aircraft wiring will accommodate for DME sets that feature "2 out of 5" tuning with a simple change.

34.11.1.6 SPARE WIRING. On airplanes 5050, 5051, 5053 thru 5055, 5057, 5058, and 5060 thru 5120 over one hundred spare wires have been installed in the flight station to facilitate the addition of future electronic systems. These wires extend from connectors mounted on the bulkhead at Fuselage Station 164 to connectors stowed behind the main instrument panel, in the center pedestal, and under the side consoles.

34.11.1.7 LIST OF ALTERNATE ELECTRONIC SYSTEMS FOR WHICH COMPLETE PROVISIONS ARE INSTALLED.

A. Dual VHF Communications Systems

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
		ALTERNATE NO. 1		
Collins	Transceiver	618M-1	522 2466 004 △1	2
Collins	Control Panel (VOR-VHF-DME)	313 N-3D	522 2448 025 △2 or 522 2448 057 △1	1
Collins	Control Panel (VOR-VHF)	313 N-3	522 2448 015 △1 or 522 2448 056 △2	1
		ALTERNATE NO. 2		
Bendix	Transceiver	RT-41A		2
Gables	Control Panel (VOR-VHF-DME)	G 1347		1
Gables	Control Panel (VOR-VHF)	G 1774		1
		ALTERNATE NO. 3		
Wilcox	Transceiver	807A		2
Gables	Control Panel (VOR-VHF-DME)	G 1347		1
Gables	Control Panel (VOR-VHF)	G 1774		1

B. Dual VHF Navigation Systems

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
		ALTERNATE NO. 1		
Collins	Receiver	51RV-1	522 2450 014	2
NOTE:	51RV-1 uses the same control panels as listed above under Alternate No. 1 of VHF Communications Systems.			
		ALTERNATE NO. 2		
Bendix	Receiver	RNA-26A		2
NOTE:	RNA-26A uses same control panels as listed above under Alternates Nos. 2 & 3 of VHF Communications Systems.			
		ALTERNATE NO. 3		
Wilcox	Receiver	806A		2
NOTE:	806A uses same control panels as listed above under Alternates Nos. 2 & 3 of VHF Communications Systems.			

△1 Red lighted panel △2 White lighted panel

C. Distance Measuring Equipment

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
		ALTERNATE NO. 1		
Collins	Transceiver	860E-2	522 2702 054	1
NOTE: Same control panel as listed for DME under VHF Communication System.				
		ALTERNATE NO. 2		
Bendix	Transceiver	DMA-29AL		1
NOTE: DMA-29AL uses same control panels as listed under Alternates Nos. 2 and 3 of VHF Communication Systems.				
		ALTERNATE NO. 3		
RCA	Transceiver	AVQ-60E		1
NOTE: AVQ-60E uses same control panels as listed under Alternates Nos 2 and 3 of VHF Communication Systems.				

D. Dual Automatic Direction Finder

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
		ALTERNATE NO. 1		
Collins	Receiver	51Y-4	522 1836 00	2
Collins	Control Panel	614L-8	522 2357 00	2
Collins	Loop Antenna	137A-4	522 2301 005	2
Collins	QEC	582A-4	522 2237 004	1
Collins	Line Simulator	316A-35	522 1042 005	2
Collins	Antenna Coupler	179J-1A	522 1860 004	2
Collins	QEC	582A-10	522 2740 004	1
		ALTERNATE NO. 2		
Bendix	QEC	QCA-73A-30U-6N		1
Bendix	Receiver	DFA-73A		2
Bendix	Control Panel	CNA-73( )		2
Bendix	Loop Antenna	LPA-73C		2
Bendix	QEC	QCA-73A-30U-0		1
Bendix	Line Simulator	LSA-73A-34P		2
Bendix	Antenna Coupler	CUA-73-195		2

E. Air Traffic Control System

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
		ALTERNATE NO. 1		
Collins	Transponder	621A-3	522 2703 00	1
Gables	ATC - Marker Beacon Control Panel	G 1524 △1 or G 1524A △2		1
		ALTERNATE NO. 2		
Bendix	Transponder	TRA-61AL		1
Bendix	ATC Control Panel	CNA-61		1
		ALTERNATE NO. 3		
Wilcox	Transponder	714C-2		1
Gables	Control Panel	G 1524 △1 or G 1524A △2		1

△1 Red lighted panel. △2 White lighted panel.

F. Marker Beacon

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
ALTERNATE NO. 1				
Collins	Receiver	51Z-4	522 2996 011	1
NOTE: 51Z uses G1524 control panel listed under Air Traffic Control System, Alternate No. 1				
ALTERNATE NO. 2				
Bendix	Receiver	MKA-28A		1
NOTE: MKA-28A uses G 1524 control panel listed under ATC, Alternate No. 1. If Bendix ATC control is used (see ATC, Alternate No. 2), a separate Marker Beacon control panel is needed.				

G. Weather Radar System

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
ALTERNATE NO. 1				
Bendix	Transmitter-Rec.	RDR-1D		1
Bendix	Synchronizer	SYN-1B		1
Bendix	Indicator	PPI-1E-1 or PPI-1E-2		1
Bendix	Antenna	ANT-1G		1
Gables	Control Panel	G 908		1
ALTERNATE NO. 2				
Bendix	Synchronizer-T.R. Unit	RDR-1E		1
Bendix	Indicator	PPI-1G		1
Bendix	Antenna	ANT-1G		1
Gables	Control Panel	G 908A		1
NOTE: Minor rework to nose rack necessary to install Alternate No. 2.				

H. Dual Flight Director Systems

MFR.	EQUIPMENT	TYPE	MANUFACTURER PART NO.	QTY
ALTERNATE NO. 1 - COLLINS FD-108				
Collins	Course Indicator	331A-6A	522 2782 004	2
Collins	F. D. Indicator	329B-7A	522 3206 004	2
Collins	Inst. Amplifier	344C-1D	522 3120 004	2
Collins	Steering Comp.	562A-5F	522 3121 024	2
NOTE: See Flight Director System, paragraph 34.11.1.4 for description of additional FD-108 space and wiring provisions.				
ALTERNATE NO. 2 - SPERRY IIS				
Sperry	Pictorial Deviation Ind.	R4A	1384457-601	2
Sperry	Horizon Indicator	HZ4	1776365-602	2
Sperry	Computer	Z5	1780607-3	2
Sperry	Instrument Rack		1783180-7	1
Sperry	Mode Selector		1778905-2	2
NOTE: Certain aircraft wiring and mounting provisions must be changed if this system is installed since the Collins FD-108 and the Sperry IIS are not interchangeable systems.				

SECTION 34.12

DOPPLER RADAR NAVIGATION AND COMPUTER SYSTEM

> Note
>
> The Collins Doppler is installed on some airplanes as optional equipment. Refer to this section for information if the complete system, provisions for the system or a partial system is installed.

Contents

Page

DESCRIPTION AND OPERATION 34-101
- General 34-102
- List of Components 34-102
- Control Panels 34-102
- Transmitter-Receiver 34-102
- Antenna 34-102
- Computer-Tracker 34-102
- Doppler Radar Indicator 34-102
- Doppler Navigation Computer Control Panel 34-102
- Doppler Navigation Computer 34-103

OPERATIONAL CHECKOUT 34-103
- Doppler Navigation and Computer System 34-103

DETAILED MAINTENANCE INSTRUCTIONS 34-103
- Doppler Radar Navigation System 34-104
- Doppler Navigation Computer System 34-104

REMOVAL AND REPLACEMENT 34-104
- Doppler Antenna 34-104
- Doppler Transmitter-Receiver 34-104
- Doppler Computer-Tracker 34-104

FIGURES
- 34-26 Doppler Radar Navigation and Computer System Components 34-105
- 34-27 Doppler Radar Navigation and Computer System Block Diagram 34-106

34.12.1 DESCRIPTION AND OPERATION.

34.12.1.1 GENERAL.

(See figures 34-26 and 34-27). A Collins Doppler Radar Navigation System DN-101 presents computed groundspeed and drift angle information on a Doppler radar indicator. In addition to the indicator, the system is composed of a DOP radar control panel, a transmitter-receiver, an antenna, and a computer-tracker. The system transmits three beams of X-band energy downward at three different angles and measures the Doppler frequency shift of each of the three reflected beams. The frequency shifts are translated into groundspeed and drift angle information in the computer-tracker. Besides the indicator presentation, related information is also furnished to a Collins Doppler Navigation Computer System NC-103, which includes a DOP NAV computer control panel and a Doppler navigation computer. Selected on the DOP NAV computer control panel, a desired ground track and distance is continuously compared with the actual ground track information from the computer-tracker. This provides a display of the remaining distance to go along the selected track as well as the cross-track distance on the DOP NAV computer control panel. The Doppler navigation computer can also furnish cross-track deviation and track angle error information to the autopilot.

34.12.1.2 LIST OF COMPONENTS.

UNIT	NUMBER USED	TYPE	TYPE NUMBER	PART NUMBER
Transmitter-Receiver	1	Collins	621C-1B	522 2359 015
Computer-Tracker	1	Collins	562F-1B or 562F-1C	522 2358 00
DOP Radar Control Panel	1	Collins	714G-2A	522 2520 00
Doppler Radar Indicator	1	Collins	338B-1A	552 2127 005
Doppler Antenna	1	Collins	137Y-1A	522 2239 00
DOP-NAV Computer Control Panel	1	Collins	162C-1	522 1589 013 △2 522 1589 00 △1
Doppler Navigation Computer	1	Collins	560E-1	522 1588 00

△1 Aircraft 5001 thru 5065 (red lighted).

△2 Aircraft 5066 and up (white lighted).

34.12.1.3 CONTROL PANEL. Operation of the DN-101 navigation system is controlled from a DOP radar control panel which contains the following switches:

A Land-Sea switch, an OFF-DOP-STBY-TEST switch, and a four-position slewing switch. The Land-Sea switch selects the circuitry compatible with the earth's surface characteristics. The OFF-DOP-STBY-TEST switch controls the internal functioning of the system. The "OFF" position renders the system inoperative. The "DOP" position activates the DN-101 Doppler radar navigation system. The "STBY" position allows any associated computer to operate on information selected with the slewing switch and keeps the sensor equipment activated for instantaneous operation. The test position provides the self-test function. The slewing switch is used to select input data to associated computers during dead reckoning navigation.

34.12.1.3 TRANSMITTER-RECEIVER. A Collins 621C-1B transmitter-receiver is composed of four plug-in modules; a modulator, a 3-MC amplifier, a regulator, and a power supply and klystron module. A blower mounted on the front panel cools the unit. The transmitter portion produces an 8,800 megacycle signal which is frequency-modulated at 1 megacycle. In the antenna, part of this signal is mixed with the Doppler-shifted return signal, and the resultant signal is routed back to the transmitter-receiver.

34.12.1.4 ANTENNA. Three separate radiating horns, one for each of the three beam directions, are included in a Collins 137Y-1A antenna. The transmitted signal is sequentially routed to each of the horns for a 1-second duration, with a 1/3-second pause from the time one horn is de-energized and another energized. The switching is accomplished with ferrite switching devices which respond to timing signals from the computer-tracker. Reflected signals received are routed through a mixing stage before being conducted to the transmitter-receiver.

34.12.1.5 COMPUTER-TRACKER. Twelve plug-in modules and a chassis compose a Collins 562F-1B or 562F-1C computer-tracker. This unit serves to control the sequencing of the antenna horn switching, to measure and simplify the frequency shift, and to compute the groundspeed and drift angle. Computed information is routed to the Doppler radar indicator and to the NC-130 navigation computer system. The No. 1 directional gyro furnishes a stable heading reference to the computer-tracker. The 562F-1B is for use with a single Doppler system. The 562F-1C has capabilities for use with a dual Doppler system and also contains SELF-TEST circuitry. To SELF-TEST the computer-tracker, place the OFF-DOP-STBY-TEST switch on the control panel to "TEST." The 338B-1A indicator should display 541.0 (± 5.0) knots and 0 (± 1) degree.

34.12.1.6 DOPPLER RADAR INDICATOR. Receiving synchro position signals from the computer-tracker, a Collins 338B-1A Doppler radar indicator displays drift angle and groundspeed information. The drift angle pointer is directly positioned by a synchro signal, but the groundspeed counter-type display is servo-driven according to the groundspeed synchro. The amplifier for this servo loop is contained within the indicator. When an OFF warning flag appears, the information presented on the indicator is unreliable.

34.12.1.7 DOPPLER NAVIGATION COMPUTER CONTROL PANEL. A Collins 162C-1 DOP NAV computer control panel provides control, selection, and display of computed information for the Collins NC-103 navigation computer system. The computer is turned on with a two-position ("ON," "OFF") control switch, but until the nose landing gear scissors switch is closed, the computer simply warms up and remains on standby. Two stages of operation are provided, and transition from stage A to B is controlled by a two-position ("MAN," "AUTO") MAN-AUTO switch. In the "MAN" position, the activation of a stage is effected by depressing an appropriate pushbutton. The stage of operation is indicated by a STAGE A and a STAGE B stage indicating flag, which read either "ON" or "OFF," depending upon which stage is in use. In the "AUTO" position, the non-operating stage is automatically turned on when an ALONG TRACK counter for the other stage reads zero. For either stage, there is an ALONG TRACK and a TRACK ANGLE dial, each having a corresponding selector knob. Thus a combination of any heading and ground distance up to 999 nautical miles can be selected for each stage. As the airplane proceeds along course, the ALONG TRACK indication for the activated stage decreases to show the remaining miles. Deviation to either side of the track is indicated on a CROSS TRACK dial which also reads either "LEFT" or "RIGHT." Up to 88 nautical miles can be registered. An associated selector knob is used to set any desired cross-track or to correct the indicator if necessary.

34.12.1.8 DOPPLER NAVIGATION COMPUTER. A Collins 560E-1 computer receives actual ground track and groundspeed information from the computer-tracker and receives selected information from the DOP NAV computer control panel. After computer calculation, signals are returned to the control panel to provide ALONG TRACK and CROSS TRACK readings. If the DOP NAV mode selector switch on the autopilot flight controller is depressed, the autopilot will respond to the Doppler navigation computer information. A track angle error signal and a cross-track deviation are routed to the autopilot flight control computer.

34.12.1 OPERATIONAL CHECKOUT.

34.12.2.1 OPERATIONAL CHECKOUT OF THE DOPPLER RADAR NAVIGATION AND COMPUTER SYSTEM.

a. Remove pin from nose gear scissors, and lower upper arm to de-actuate the nose landing gear scissors switch.

b. Connect external power to the airplane (see Chapter 24).

c. Place the MAIN INV switch in the "ON" position.

d. Be sure the DOPPLER CONT DC, the DOPPLER XMTR 115 VAC, the DOPPLER EXCITATION 115 VAC, and the DOPPLER SYNCHROS 26 VAC circuit breakers are closed.

e. Place the control switch on the DOP control panel in the "STBY" position, and using the slewing switch, set the Doppler radar indicator at zero degrees drift angle and 600 knots groundspeed. On the DOP NAV computer control panel, set the ALONG TRACK at 60 miles and the TRACK ANGLE at the actual airplane heading for both STAGE A and STAGE B, and set the CROSS TRACK at zero.

Note

Be sure the No. 1 gyrosyn compass system is operating.

f. Place the control switch on the DOP NAV computer control panel in the "ON" position, depress, the STAGE A pushbutton, and place the MAN-AUTO switch in the "AUTO" position.
RESULT: The STAGE A ALONG TRACK should decrease at the rate of 10 miles per minute. Thus, after 6 minutes the STAGE A ALONG TRACK should be decreased to zero, and the STAGE B ALONG TRACK should still be at 60. At this point the STAGE B indicating flag should change from "OFF" to "ON," and the STAGE B ALONG TRACK should begin decreasing at the rate of 10 miles per minute. The CROSS TRACK should remain constant during this time.

g. Before the STAGE B ALONG TRACK has reached zero, reset STAGE A ALONG TRACK to 10 and increase the TRACK ANGLE by 90 degrees.
RESULT: When STAGE B ALONG TRACK reaches zero, which should require 6 minutes, the STAGE A indicating flag should change from "OFF" to "ON." The STAGE A ALONG TRACK should not decrease, and the CROSS TRACK should increase at the rate of 10 miles per minute LEFT.

h. Set the STAGE B ALONG TRACK at 40 miles, and decrease the TRACK ANGLE by 90 degrees. Place the MAN-AUTO switch in the "MAN" position, and depress the STAGE B pushbutton.
RESULT: The STAGE B indicating flag should change from "OFF" to "ON," the CROSS TRACK should decrease at rate of 10 miles per minute, and the STAGE B ALONG TRACK should remain constant.

i. Reset the STAGE A TRACK ANGLE at the airplane heading. Allow the CROSS TRACK to decrease to zero, begin increasing to the RIGHT, and then depress the STAGE A pushbutton.
RESULT: The STAGE A indicating flag should change from "OFF" to "ON," the CROSS TRACK should remain constant, the STAGE A ALONG TRACK should decrease to zero, and the STAGE A indicating flag should change to "OFF."

j. Reset the STAGE B TRACK ANGLE at the airplane heading, and adjust the CROSS TRACK to zero.

k. Using the slewing switch, set the Doppler radar indicator at a drift angle of 37 degrees left, depress the STAGE B pushbutton, and set the STAGE A ALONG TRACK at 40 miles.
RESULT: The STAGE B indicating flag should change from "OFF" to "ON," and the STAGE B ALONG TRACK should decrease to zero, at which time the CROSS TRACK should be 30 (±1) miles LEFT.

l. Reset the drift angle at 37 degrees right, and depress the STAGE A pushbutton.
RESULT: The STAGE A indicating flag should change from "OFF" to "ON," and the STAGE A ALONG TRACK should decrease to zero, at which time the CROSS TRACK should be zero (±2) miles LEFT or RIGHT.

m. Disconnect external power from the airplane, and restore it to normal.

Note

Be sure to replace pin in nose gear scissors.

34.12.3 DETAILED MAINTENANCE INSTRUCTIONS.

34.12.3.1 DOPPLER RADAR NAVIGATION SYSTEM. For detailed information concerning the components of the system, together with bench test procedures, refer to Instruction Book for the DN-101 Doppler Radar Navigation System, Publication No. 523-0755-834, Collins Radio Company.

34.12.3.2 DOPPLER NAVIGATION COMPUTER SYSTEM. For detailed information concerning the components of the system, together with bench check and calibration procedures, refer to Instruction Book for the NC-103 Doppler Navigation Computer System, Publication No. 520-5998-004 Collins Radio Company.

34.12.4 REMOVAL AND REPLACEMENT.

34.12.4.1 DOPPLER ANTENNA.

a. Between fuselage station 698.5 and 739.0 remove the 55 screws from the Doppler radome flange, and remove the radome from the airplane.

b. Remove the cover.

c. Disconnect the flexible waveguide at the Doppler antenna and the antenna cabling connector.

d. Hold the antenna, remove the four nuts and four washers from the bolts that secure the antenna to the support installation, and lower the antenna out of the airplane.

e. To install the Doppler antenna, reverse the procedures of steps (a) through (d).

f. Perform an operational checkout of the Doppler Radar Navigation System.

34.12.4.2 DOPPLER TRANSMITTER-RECEIVER.

a. After removal of the Doppler antenna, the transmitter-receiver can be removed through the radome opening. The transmitter-receiver can also be removed from inside the aft compartment. Access to this area is gained through the speed brake opening when the brake is fully extended.

WARNING

Make sure the speed brake safety locks are installed after extending the brakes.

b. Disconnect the transmitter-receiver cabling connectors.

c. At the airplane forward end of the transmitter-receiver shockmount, loosen and disengage the shockmount retainer assembly from the transmitter-receiver mounting lugs.

d. Slide the transmitter-receiver toward the forward end of the airplane until the transmitter-receiver is disengaged from the flexible waveguide, and remove the transmitter-receiver from the airplane.

e. To install the Doppler transmitter-receiver, reverse the procedures of steps (a) through (d).

f. Perform an operational checkout of the Doppler Radar Navigation System.

34.12.4.3 DOPPLER COMPUTER TRACKER.
(See figure 34-26.)

a. Gain access to the center floor area on the forward side of the pressure bulkhead at fuselage station 570.

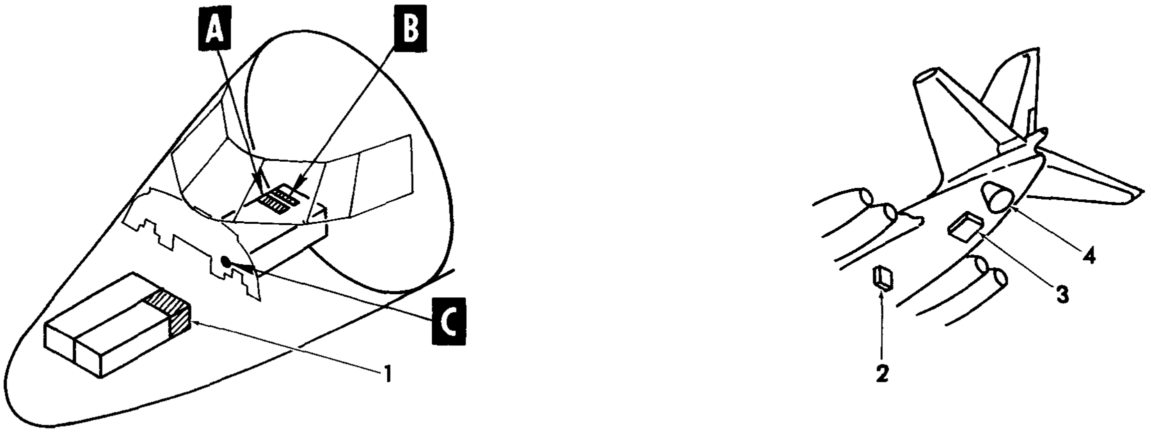
b. Loosen the two small setscrews on the right side of the computer-tracker mount.

c. Loosen the large setscrew on the right side of the computer-tracker mount. The loosening action of this screw releases a guide assembly on the mount which pulls the computer-tracker to the right and electrically disconnects it from the mount.

d. Lift the computer-tracker from the mount and remove it from the airplane.

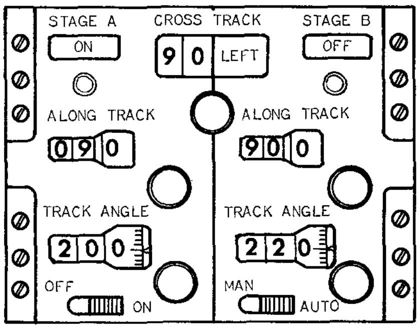
e. To install the Doppler computer-tracker, reverse the procedure of steps (b) through (d).

f. Perform an operational checkout of the Doppler Radar Navigation System.



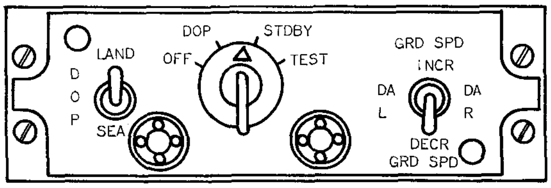
1. DOPPLER NAVIGATION COMPUTER [1]
2. COMPUTER TRACKER
3. TRANSMITTER-RECEIVER
4. ANTENNA

[1] ON AIRPLANES WITH SOLID STATE ELECTRONIC EQUIPMENT INSTALLED, THIS UNIT IS LOCATED IN THE BAGGAGE COMPARTMENT.



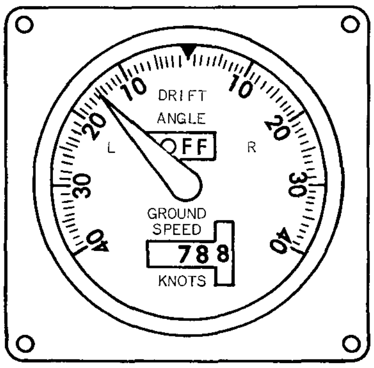
A

DOPPLER NAVIGATION COMPUTER CONTROL PANEL
(LOOKING DOWN)



B

DOPPLER RADAR CONTROL PANEL
(LOOKING DOWN)



C

DOPPLER RADAR INDICATOR
(LOOKING FORWARD)

Figure 34-26. Doppler Radar Navigation and Computer System Components

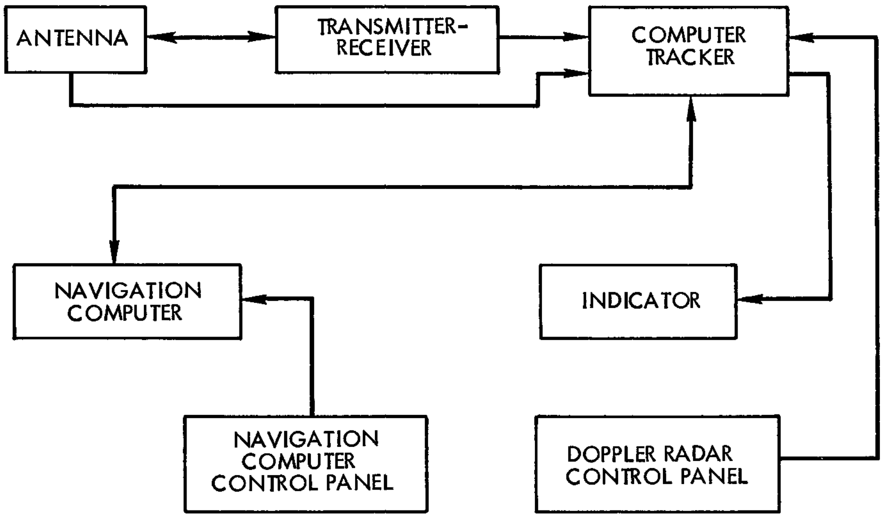


Figure 34-27. Doppler Radar Navigation and Computer System Block Diagram

CHAPTER 35
OXYGEN SYSTEM

Contents Page

DESCRIPTION AND OPERATION 35-2
General 35-2
Oxygen Cylinders 35-2
Pressure Reducer-Shutoff Valve 35-2
Oxygen Warning System 35-2
Passenger Oxygen Control Panel 35-2
Blowout Indicator Discs 35-2
Selector Valves 35-3
Door Actuator Valves 35-3
Lanyard Operated Valves 35-3
Passenger Masks 35-3
Crew Oxygen Masks 35-3
Smoke Goggles 35-3
Crew Pressure-Demand Regulators 35-3
Portable Oxygen Cylinder 35-3
Oxygen Duration 35-4

OPERATIONAL CHECKOUT 35-4

TROUBLESHOOTING 35-5

REMOVAL AND REPLACEMENT 35-6
General 35-6
Replacement of System Components 35-6
Restowing of Passenger Oxygen Masks 35-6
Replacement of Oxygen Cylinder Pressure Reducer Rupture Disc 35-7
Replacement of Oxygen Selector Valves 35-7
Torque Values for Oxygen Plumbing 35-7

PRESERVATION/DEPRESERVATION DURING PROLONGED INACTIVITY 35-7
Preservation 35-7
Maintenance of Preservation 35-7
Depreservation 35-7

EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION 35-7

FIGURES

35-1. Oxygen System Components 35-8
35-2. Oxygen Duration Charts 35-10
35-3. Passenger Oxygen Mask and Hose Replacement 35-12
35-4. Oxygen System Schematic Diagram (Airplanes 5001 through 5085) 35-13
35-5. Oxygen System Schematic Diagram (Airplanes 5086 and Up) 35-15

35.1 DESCRIPTION AND OPERATION.

35.1.1 GENERAL. (See figure 35-1.)

a. The oxygen system is provided for use by passengers and crew if pressurization fails at high altitudes, or if cabin air becomes contaminated. A high-pressure, gaseous oxygen system, reduced to low pressure at the cylinder is used. A manually operated system is used for the pilot and copilot, while operation of the passenger oxygen system is semi-automatic. Oxygen warning systems are incorporated in all airplanes.

b. The oxygen system supply is carried in two cylinders, one for use by the pilot and copilot, and the other for the remaining occupants of the airplane. Cross-feeding between the two systems is made possible by use of a manual crossfeed valve. Airplanes 5001, 5002, and 5004 thru 5035 have plumbing provisions for two long-range oxygen cylinders. Airplanes 5003, 5036 and up incorporate a capped-tee fitting in the passenger supply line which can be used to connect long-range system plumbing. On all airplanes, a portable oxygen cylinder is provided which incorporates quick-connecting fittings that enable crew oxygen masks to be disconnected from the regulator supply and to be connected to the portable cylinders. The portable oxygen cylinder is stowed in the aft end of the pilot's side console.

c. The pilot's and copilot's oxygen masks are stowed on the wainscots adjacent to their seats. Passenger masks are stowed in recessed compartments, one above each passenger seat, in the lavatory and adjacent to the galley. The oxygen masks, when stowed in their recessed compartments, are contained behind a spring-loaded door which opens automatically when cabin altitude exceeds 13,000 to 14,000 feet, as in failure of the pressurization system. Unlatching of the stowage doors is accomplished by oxygen pressure released into the passenger distribution system by the opening of an automatic turn-on valve actuated by cabin pressure. Each mask then falls from its stowage compartment. The action of donning the mask pulls a lanyard connected to a pin in a lanyard operated valve. Extraction of the pin opens the valve which releases oxygen to the mask. A manual turn-on valve (on the PASSENGER OXYGEN control panel in the flight station) permits the stowage doors to be opened and the masks to be deployed at lower altitudes or for ground test.

d. On airplanes 5046 and up, two remote oxygen cylinder pressure gages are mounted on the forward right side of the bulkhead at fuselage station 232.

35.1.2 OXYGEN CYLINDERS.

The oxygen cylinders are charged to 1,850 PSI. This pressure is reduced to 70 to 90 PSI by a pressure regulator on each cylinder. The capacity of the oxygen cylinder for the pilot and copilot is 39.0 cubic feet, while the one for passenger use has a capacity of 76.6 cubic feet. The portable oxygen cylinder is charged to 1,850 PSI, and has a capacity of 11 cubic feet.

35.1.3 PRESSURE REDUCER-SHUTOFF VALVE.

Airplanes 5001 through 5045 have a lever-operated OPEN-CLOSE valve on the normal and long-range cylinders. Airplanes 5046 and up have a slow-opening rotating knob-operated OPEN-CLOSE valve, which also has remote pressure gage ports and a remote gage line venting port. The portable cylinder has an OPEN-CLOSE valve operated by a rotating knob. All valves have pressure gages.

35.1.4 OXYGEN WARNING SYSTEM.

Automatic or manual turning on of passenger oxygen closes an oxygen pressure switch; this completes circuits to an oxygen warning buzzer in the flight station, to the vestibule light, to the NO SMOKING ordinance light, and to an oxygen flow indicator light on the passenger oxygen control panel. The buzzer can be silenced by depressing a buzzer shutoff pushbutton on an oxygen system test panel. A warning system test pushbutton is also provided on this panel.

35.1.5 PASSENGER OXYGEN CONTROL PANEL.

This panel contains the manual controls for the oxygen system. The automatic turn-on valve for passenger oxygen is also incorporated in the panel. The controls are: a PASS-CREW CROSSFEED valve, a MANUAL TURN ON valve, a LINE PRESSURE indicator, and a PASSENGER OXYGEN ON light. The PASS-CREW CROSSFEED valve interconnects the passenger and crew supply lines for emergency use only. All of the oxygen supply can be interconnected and used at any station. The MANUAL TURN ON valve bypasses the automatic turn-on valve and supplies oxygen to the passenger distribution lines. The pressure gage indicates the pressure in the supply line. The PASSENGER OXYGEN ON light goes on when there is pressure in the passenger distribution lines. The automatic turn-on valve is opened automatically by an aneroid exposed to cabin air pressure, and thus supplies oxygen to the passenger distribution system if cabin pressure altitude exceeds 13,000 to 14,000 feet.

35.1.6 BLOWOUT INDICATOR DISCS.

A green indicator disc, visible from outside the airplane, is connected to the overboard vent lines from the oxygen cylinders. If cylinder pressure exceeds a safe limit because of high ambient temperature expanding the gas, bursting of the cylinder is prevented by breaking of a rupture disc at the cylinder, which vents the oxygen to atmosphere. The outlet of the vent line is covered by a frangible blowout disc, which is blown out by a relatively low pressure in the vent line, thus giving a visual indication that the cylinder has been depleted. On airplanes 5001 through 5035, there are two blowout indicator discs, one in the vent line from the normal oxygen cylinders and one in the vent line from the long-range cylinder plumbing provisions. On airplanes 5036 and up, only one blowout indicator disc is installed, forward in the vent line from the normal oxygen cylinders. The discs blow out at 60 (±20) PSI. The forward blowout disc is on the lower right of the fuselage station 287, and is on all airplanes. This disc vents the normal cylinders only on airplanes 5001 through 5035, but may also be used to vent a long-range system on airplanes 5036 and up. The aft disc is on the upper left side of the fuselage at station 560, and is on airplanes 5001 through 5035 only. This disc is connected to the long-range plumbing provisions vent line.

35.1.7 SELECTOR VALVES. Two manual selector valves in the oxygen compartment permit selection of either normal or long-range oxygen supply, if the long-range provisions are used. In the normal configuration, these valves are positioned to select only the normal supply.

35.1.8 DOOR ACTUATOR VALVES. An oxygen-pressure operated door actuator valve is installed between the passenger distribution line and each passenger mask. When oxygen is supplied through the line by opening of either the automatic or the manual turn-on valve, the latch of the mask stowage door is opened, permitting the mask to fall from the compartment.

35.1.9 LANYARD OPERATED VALVES. Oxygen is supplied to a passenger mask only when the mask has been pulled downward to don it. The action of pulling the mask downwards pulls a lanyard having an attached pin which holds the valve closed. Removal of the pin allows the valve to open and to supply oxygen to the mask. No oxygen is supplied to masks which have been ejected but which have not been donned, thus conserving the supply for occupied stations.

35.1.10 PASSENGER MASKS. The passenger masks are of the demand type, with manual selection for dilution or 100 percent oxygen with safety pressure. When stowed, each mask has its selector lever placed in the "100% OXYGEN" position. One mask is above each passenger seat, two in the toilet-lavatory area, and one extra mask near the galley. A flexible hose connects each mask to the distribution system.

35.1.11 CREW OXYGEN MASKS. Two crew oxygen masks are provided, one each for the pilot and copilot. The crew oxygen masks are equipped with a fast-donning feature. Quick-disconnects enable the mask to be detached from the regulator and connected to the portable oxygen cylinder for emergency use.

35.1.12 SMOKE GOGGLES. The pilot and copilot are each provided with a pair of smoke goggles, stowed adjacent to the copilot's seat, for eye protection against smoke or fumes which might enter the flight station during a fire or similar emergency. A supply of anti-fogging compound is included with each pair of goggles.

35.1.13 CREW PRESSURE-DEMAND REGULATORS. Two crew pressure-demand regulators of the same type are provided, one each for the pilot and copilot. Each regulator is installed between the crew distribution line and the mask. The regulator maintains automatic safety pressure (1 to 2 inches of water) from 30,000 feet to 40,000 feet cabin altitude. Automatic pressure breathing is actuated at 40,000 feet cabin altitude; it delivers increased oxygen pressure varying with altitude (8.5 to 12 inches of water at 45,000 feet cabin altitude). The regulator has a visual flow indicator, a system pressure gage, and three toggle-type selector levers. The flow indicator is a mechanical device which indicates flow of oxygen to the mask by movement of a flag across a window. The pressure indicator shows the pressure of oxygen being supplied to the regulator. A SUPPLY lever opens or closes the inlet valve to the regulator. A diluter lever has "100% OXYGEN" and "NORMAL OXYGEN" positions to control the amount of oxygen delivered to the mask at cabin altitudes below 30,000 feet. In the "100% OXYGEN" position, undiluted oxygen is delivered. In the "NORMAL OXYGEN" position, oxygen concentration varies in accordance with cabin altitude. At approximately 30,000 feet cabin altitude, 100 percent oxygen is delivered to the mask regardless of the lever position. A NORMAL-EMERGENCY lever has "NORMAL", "EMERGENCY", and "TEST MASK" positions. When the lever is in the "EMERGENCY" position, 100 percent oxygen is delivered under pressure to the mask. When in the "NORMAL" position, the lever has no effect on the ratio of oxygen to air, which then varies with cabin altitude. The "TEST MASK" position of the lever permits testing of the mask and regulator.

35.1.14 PORTABLE OXYGEN CYLINDER. The portable oxygen cylinder assembly is designed to provide a portable source of 100-percent breathing oxygen to crew members at high altitudes. It consists of an oxygen supply cylinder coupled to a pressure reducer and shutoff valve, an oxygen regulator, and a supply hose which attaches directly to a crew mask. To use the portable oxygen cylinder assembly, a crewman connects the unit to the inlet of his mask and activates it by turning the ON-OFF knob at the end of the pressure-reducing valve. The oxygen supply cylinders are charged to 1800 PSI. A pressure-reducing valve on each cylinder reduces the pressure to 50 to 90 PSI before delivering oxygen to the regulator. The regulator delivers a positive safety pressure of 100-percent oxygen not in excess of 2.0 inches of water to the crew members mask through the supply hose. At cabin altitudes from 35,000 to 43,000 feet, the regulator aneroid mechanism automatically varies the outlet pressure from 1.5 to 12.5 inches of water maximum, with values dependent upon cabin altitude. When the portable oxygen cylinder is not in use, the valve must be in the "OFF" position to preclude oxygen loss.

Note

Depending on factory adjustment of the regulator, a flow of oxygen will generally occur with the valve turned to "ON" although not in use. Adjustments can alter the performance of the regulator, but they must be made by the manufacturer or a manufacturer's approved repair station only.

35.1.15 OXYGEN DURATION. (See figure 35-2.) The quantity of oxygen stored in the supply cylinder and the oxygen required at altitude shown in the duration tables, figure 35-2, are computed or corrected to sea level conditions. The variation in required inspired oxygen versus altitude is affected by several physiological factors, but the major effect results from the decreased density (barometric pressure) of the ambient air at altitude. The required oxygen or combination of oxygen and air varies inversely with altitude. The volume inspired of course remains the same, since this is a function of lung capacity. Individuals vary in their normal inspired volume per minute due primarily to differences in physical lung capacity and to variations in physical activity. A person at rest will average about 8 liters per minute whether at sea level or at some altitude.

The variation of barometric pressure with altitude is shown in the following table:

(A) Altitude Feet	(B) Barometric Pressure PSI	(C) Volumetric Ratio of Barometric Pressure for Sea Level At Altitude 14.7/(B)	(D) 8 Liter Sea Level Unit Volume At Altitudes 8 × (C)
0	14.7	1.0	8.0
10,000	10.1	1.5	12.0
15,000	8.3	1.8	14.4
20,000	6.8	2.2	17.6
25,000	5.5	2.7	21.6
30,000	4.4	3.3	26.4
35,000	3.5	4.2	33.6
40,000	2.7	5.4	43.2

Therefore it can be seen that 8 liters of oxygen at sea level becomes 43.2 at 40,000 feet, providing an unused excess on a per minute basis which in turn permits greater duration at altitude. It should be noted that the oxygen requirements at altitude shown in figure 35-2 do not follow exactly the barometric variation with altitude because regulator design permits flows in excess of the minimum required.

35.2 OPERATIONAL CHECKOUT.

WARNING

During this checkout, small quantities of oxygen are released into the interior of the airplane. Observe NO SMOKING precautions. Permit no refueling, defueling, fluid servicing, or operation of electrical or electronic equipment. Personnel performing the checkout should wear clean clothing, free of oil or grease stains. Do not breathe oxygen with any type of frost or windburn protection grease on the face or the lips. Only qualified personnel should perform this checkout. Ground the airplane before starting the checkout, and do not disconnect the grounding cables until the checkout is completed and the airplane has been returned to normal.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

Note

Make sure cylinder valves are in "OPEN" position.

35.2 (Cont).

a. Check the pressure gages on the pilot's and copilot's regulators and the passenger oxygen control panel.
RESULT: The gage should indicate between 70 and 90 PSI.

b. With a mask connected to the pilot's regulator, turn the pilot's diluter lever to "NORMAL OXYGEN", and breathe into the mask. Observe the flow indicator.
RESULT: Black and white segments should appear in the window of the indicator.

c. Place the diluter lever in the "100% OXYGEN" position, and breathe into the mask. Observe the flow indicator.
RESULT: Black and white segments should appear in the window of the indicator.

d. Return the diluter lever to the "NORMAL OXYGEN" position.

e. Momentarily hold the NORMAL-EMERGENCY lever in the "EMERGENCY" position, and then return it to the "NORMAL" position.
RESULT: Oxygen should flow continuously from the regulator while the lever is in the "EMERGENCY" position. Flow should stop when the lever is returned to the "NORMAL" position.

f. Place the diluter lever in the "100% OXYGEN" position. Momentarily hold the NORMAL-EMERGENCY lever in the "EMERGENCY" position, and then return it to the "NORMAL" position.
RESULT: Oxygen should flow continuously from the regulator while the lever is in the "EMERGENCY" position. Flow should stop when the lever is returned to the "NORMAL" position.

g. Return the diluter lever to the "NORMAL OXYGEN" position, and place the EMERGENCY-NORMAL lever in the "TEST MASK" position. Return the EMERGENCY-NORMAL lever to the "NORMAL" position.
RESULT: Oxygen should flow continuously from the regulator while the EMERGENCY-NORMAL lever is in the "TEST MASK" position. Flow should stop when the EMERGENCY-NORMAL lever is returned to the "NORMAL" position.

h. Place the SUPPLY lever in the "OFF" position.

i. Repeat steps (b) through (h), using the co-pilot's regulator.

j. To test the PASS-CREW CROSSFEED valve, turn off the crew cylinder, and operate "TEST MASK" to bleed the crew system. Turn the valve to the "ON" position, and observe the pressure gage on the crew panel.
RESULT: The same pressure observed in step (a) should be indicated.

k. Place the MANUAL TURN ON valve in the "ON" position.
RESULT: The passenger oxygen mask doors should release, and the masks should fall free. The lanyard pin in each passenger mask should remain in the flow valve. The vestibule light, the NO SMOKING ordinance light, and the oxygen panel light should go on. The oxygen warning buzzer should sound.

l. Press the BUZZER SHUT OFF pushbutton switch on the oxygen test panel.
RESULT: The oxygen warning buzzer should be silenced. The lights which went on in step (k) should remain on.

Note

Accomplish steps (m) and (n) consecutively on each individual mask to conserve oxygen supply.

m. Check that the diluter lever on each passenger mask is in the "100% OXYGEN" position. Pull each passenger mask to disengage the lanyard pin.
RESULT: Oxygen should flow continuously from each passenger mask.

n. Place the diluter lever on each passenger mask in the "DILUTE" position.
RESULT: Oxygen flow from the masks should stop.

Note

There may be a low flow with the lever in "DILUTE". This is acceptable. Free flow from each mask, with the lever in "100% OXYGEN", should be limited to 1 minute.

o. With each passenger mask lever in the "DILUTE" position, breathe into each mask in turn.
RESULT: A flow of oxygen should be observed at each inhalation. Flow should stop during exhalation.

p. Return the MANUAL TURN ON valve to the "OFF" position. Place each passenger mask diluter lever in the "100% OXYGEN" position.
RESULT: With depletion of system pressure, the flow indicator light, the vestibule light, and the NO SMOKING ordinance lights should go off.

WARNING

It is essential that the passenger mask diluter lever be returned to the "100% OXYGEN" position.

q. Reinstall the passenger oxygen masks, and return the system to normal.

Note

To check the automatic oxygen warning feature, it is necessary to fly the airplane to an altitude above 14,000 feet with pressurization off, and observe that the buzzers sound, the warning lights go on, and the passenger masks drop out.

35.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
REGULATOR OR CONTROL PANEL GAGE SHOWS SYSTEM PRESSURE HIGHER THAN 90 PSI.		
Defective regulator.	Check pressure on other regulator(s).	Replace defective regulator.
REGULATOR OR CONTROL PANEL GAGE SHOWS PRESSURE LOWER THAN 70 PSI.		
Depleted cylinder.	Crossfeed from other cylinder and recheck pressure.	Replace depleted cylinder with fully charged cylinder.
FLOW INDICATOR NOT OPERATING WHEN CREW MASK BEING USED.		
Defective indicator or depleted system.	Crossfeed from other cylinder and recheck indicator.	Replace regulator or replace cylinder.
Regulator not releasing oxygen, or depleted cylinder.	Test by breathing through mask. Crossfeed from other cylinder.	Replace defective regulator or depleted cylinder as necessary.
NO PASSENGER MASKS RELEASED WHEN SYSTEM IS TURNED ON.		
Manual turn-on valve defective, if system was being operated manually, or depleted cylinder.	Check that system pressure is between 70 and 90 PSI.	Replace manual turn-on valve, or replace depleted cylinder.
Automatic turn-on valve defective, or cabin pressure aneroid not actuating turn-on valve, if system was in automatic mode.	Check that system pressure is between 70 and 90 PSI.	If pressure is satisfactory replace passenger oxygen panel. If pressure is low, replace depleted cylinder.
ALL PASSENGER MASKS NOT RELEASED WHEN SYSTEM IS TURNED ON.		
Defective release mechanism of inoperative masks, or loose connection in line between system and stowage compartment.	Inspect release mechanisms of inoperative masks, and check for loose line connections.	Repair, adjust, or replace release mechanisms, or tighten line connections, as necessary.
WARNING INDICATIONS NOT GIVEN WHEN SYSTEM IS TURNED ON.		
Defective pressure switch or circuitry.	Check circuitry continuity.	Repair circuit or replace pressure switch.
WARNING INDICATIONS DO NOT GO OFF WHEN PASSENGER DISTRIBUTION SYSTEM PRESSURE IS DEPLETED.		
Control panel pressure switch or circuit defective.	Perform continuity check of switch and circuit.	Replace switch or repair circuit as necessary.
PREMATURE DISCHARGE OF CYLINDER PRESSURE.		
Rupture disc blown.	Check high pressure relief port of reducer valve.	Replace ruptured disc (paragraph 35.4.4).

35.4 REMOVAL AND REPLACEMENT.

35.4.1 GENERAL.

WARNING

Gaseous oxygen can be a potentially dangerous material under some conditions. The following precautions must always be taken when performing maintenance work on the oxygen system:

a. Make sure the airplane is properly grounded, that all electrical and electronic equipment is turned off, that no engines will be run, and that no refueling or defueling operations will be performed while work is being done on the oxygen system.

WARNING

Ensure that static ground lines are firmly connected to earth ground before connecting them to the airplane. Failure to comply could result in personnel injury.

b. Use only clean tools, free from grease or oil. Remove grease from contaminated tools by an approved procedure before using them.

c. Wear clean clothing, free from grease or oil stains. If clothing becomes saturated with leaking oxygen, avoid smoking or going near any source of fire or sparks until the oxygen held in the clothing has dissipated.

d. Keep dust and dirt out of the system by using correct caps and plugs on disconnected lines and ports. Keep leak detector solution out of the system.

35.4.2 REPLACEMENT OF SYSTEM COMPONENTS. Removal and replacement of oxygen system components are mechanically simple operations. The following general procedures should be used, as applicable, when working on the system:

a. Before disconnecting lines, turn off the cylinder valves which supply the portion of the system to be disconnected; then deplete any pressure which might remain in that portion of the system. If the affected system is the passenger oxygen system, this may be done by manually dropping one of the passenger masks and pulling the lanyard pin. If the crew system is affected, place the NORMAL-EMERGENCY lever of the appropriate regulator in the "EMERGENCY" position.

b. When oxygen lines or components have been disconnected, immediately cap or plug all open lines and ports to exclude dust from the system, using plugs or caps of the correct size. Do not use masking tape; this can leave a sticky deposit, and it is not airtight.

c. When replacing oxygen lines, make sure that they are located correctly. Minimum clearances between the lines and other components and structure were established during manufacture of the airplane, and inadvertent relocation of lines may decrease these clearances to less than is permissible. In general, if replacement oxygen lines are installed in the original clamping and support provisions and the lines have not been bent or otherwise distorted, the correct clearances should exist.

d. The oxygen system lines have conventional flared fittings. Tighten tube nuts to the torque values given in paragraph 35.4.6. Test for leakage with LEAK-TEC or a solution of mild soap and water.

e. Inspect replacement oxygen lines before installing them, to be sure they are free from preservative compounds. If they are contaminated, or if they were not capped during storage, vapor-degrease them by an approved procedure. Install them, or cap the ends, as soon as they are cleaned. If the lines are left unprotected, oil vapor and other contaminants from the atmosphere can enter them.

f. Do not lubricate oxygen tubing fittings having parallel threads, such as tube nuts and unions. Lubricate only the first three threads of male pipe-threaded fittings (taper threads) with anti-seize compound approved for oxygen use.

g. After replacing any lines or components, the system should be purged with oxygen from an aircraft cylinder to remove air which normally contains contaminating moisture.

h. Perform an operational checkout of the affected portion of the system after line or component replacement.

35.4.3 RESTOWING OF PASSENGER OXYGEN MASKS. Extreme care must be taken in replacing passenger oxygen masks in the overhead stowage compartments, so that they do not become entangled and fail to fall free in an emergency. It is essential that the dimensions shown in figure 35-3 be maintained.

a. Place the mask selector lever in the "100% OXYGEN" position.

b. Insert the lanyard pin in the lanyard valve of the regulator.

c. Install the headstrap tab in the clip on the hose.

d. Coil the flexible hose to a bend diameter of approximately 3-1/2 inches.

35.4.3 (Cont).

e. Install the mask and regulator in the compartment with the headstrap at the side and the regulator downwards.

CAUTION

Do not place the headstrap inside the mask, otherwise damage to the mask may result.

f. Close the retaining door.

Note

The door latch is actuated by oxygen pressure of 10 to 12 PSI. The door cannot be latched unless system pressure is reduced below this value. System pressure can be depleted by breathing into one of the masks.

35.4.4 REPLACEMENT OF OXYGEN CYLINDER PRESSURE REDUCER RUPTURE DISC. The rupture disc described in 35.1.6 must be replaced every 12 months. Because of cold flow caused by continuous exposure to high pressure the disc may rupture below its design pressure causing premature loss of oxygen supply. (See Chapter 5 for further details.)

35.4.5 REPLACEMENT OF OXYGEN SELECTOR VALVES. If it is ever necessary to replace an oxygen selector valve, ensure that the pointer of the operating handle is aligned over the index mark on the valve shaft.

35.4.6 TORQUE VALUES FOR OXYGEN PLUMBING. Tighten tube nuts and fittings to the torque values shown in the table below.

Note

Torque 1/8 inch pipe thread fitting in D2 oxygen regulator to no more than 50 inch pounds maximum. In the case of solder type fittings, torque to 40-50 inch pounds. If connection leaks, disassemble fitting; reassemble and torque to 70 inch pounds.

TORQUE VALUES FOR OXYGEN FITTINGS

Tube Size (Inch)	Material	Straight Thread		Pipe Thread	
		Min.	Max.	Min.	Max.
1/8	Stainless Steel	85	100	-	-
1/8	Aluminum	-	-	40	150
3/16	Aluminum	40	60	-	-
1/4	Aluminum	-	-	60	250
5/16	Aluminum	100	125	-	-
3/8	Aluminum	200	250	100	400

35.5 PRESERVATION/DEPRESERVATION DURING PROLONGED INACTIVITY.

35.5.1 PRESERVATION. If the airplane is to be inactive for a prolonged period, observe the following precautions to protect the oxygen system from deterioration and to keep it clean:

WARNING

Oxygen system components and lines must always be kept free from grease or oil, to minimize possibility of spontaneous fire or explosion.

a. During storage maintain a minimum pressure of 50 PSI in the system.

b. If the airplane is to be inactive more than 30 days, enclose the following components in plastic bags:

    1. Each passenger and crew mask.
    2. The portable oxygen bottle hose end, pressure reducer shutoff valve, and regulator.
    3. The flexible hose end, if the crew mask is disconnected from the flexible hose.

35.5.2 MAINTENANCE OF PRESERVATION.

Make sure the plastic covers are not disturbed during the storage period.

35.5.3 DEPRESERVATION.

a. Remove plastic bags.

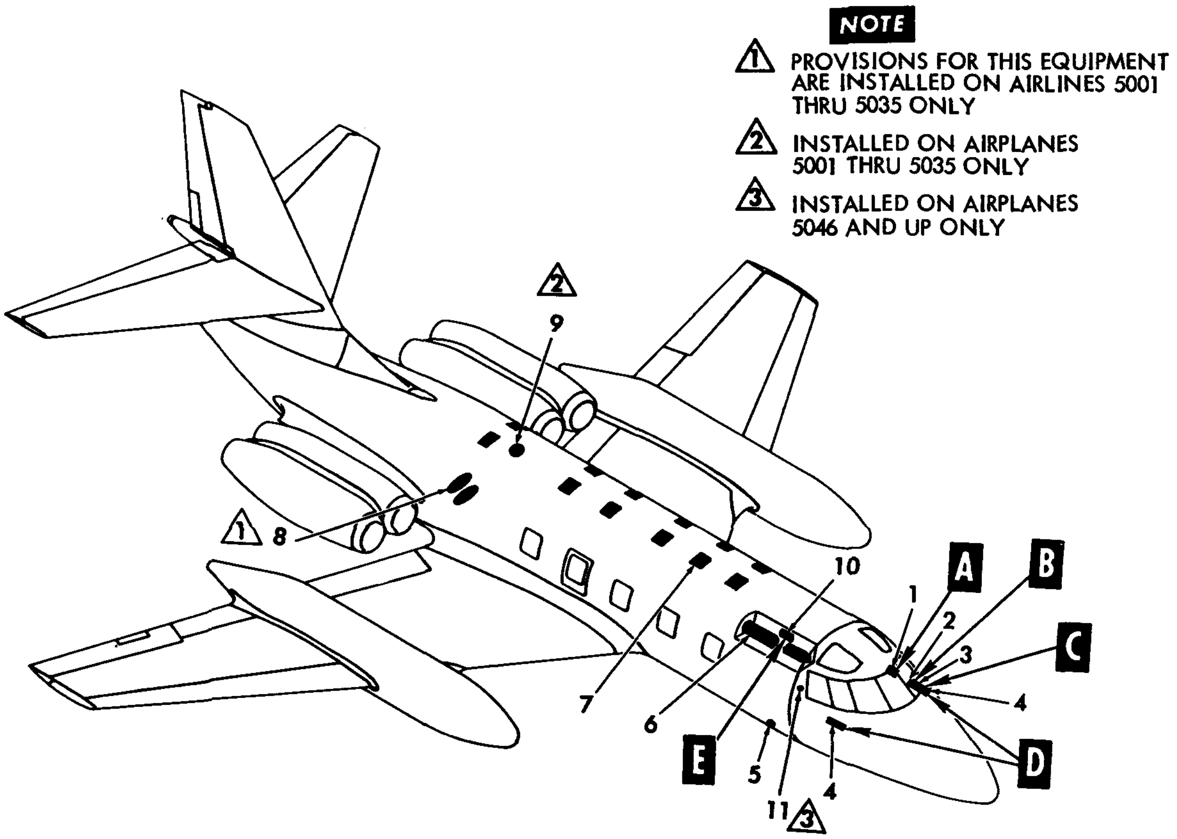
b. Reassemble system completely.

c. Recharge system. (Refer to Section 12.10.)

d. Place selector valves in proper position.

e. Check out crew and passenger systems to ensure proper operation.

35.6 EXTREME CLIMATIC AND ENVIRONMENTAL INFORMATION. The oxygen system does not require any special procedures in extremes of climate and environment. The blowout indicators should be checked when ambient temperatures are extremely high.



NOTE

△1 PROVISIONS FOR THIS EQUIPMENT ARE INSTALLED ON AIRLINES 5001 THRU 5035 ONLY

△2 INSTALLED ON AIRPLANES 5001 THRU 5035 ONLY

△3 INSTALLED ON AIRPLANES 5046 AND UP ONLY

1. PORTABLE OXYGEN CYLINDER STOWAGE
2. OXYGEN WARNING TEST PANEL
3. PASSENGER OXYGEN CONTROL PANEL
4. CREW OXYGEN CONTROL PANEL
5. FORWARD BLOWOUT INDICATOR DISC
6. SHORT-RANGE OXYGEN CYLINDERS
7. PASSENGER OXYGEN MASK CONTAINER
8. LONG-RANGE OXYGEN CYLINDERS △1
9. AFT BLOWOUT INDICATOR DISC △2
10. OXYGEN SELECTOR VALVE PANEL
11. REMOTE PRESSURE GAGES △3

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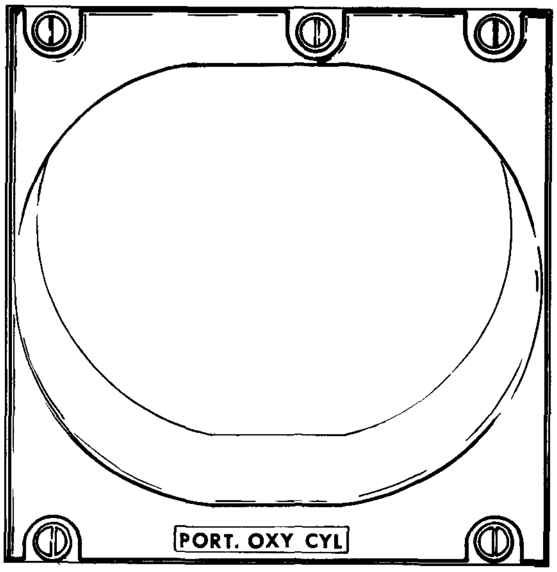
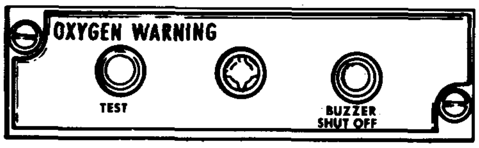
Figure 35-1. Oxygen System Components (Sheet 1 of 2)



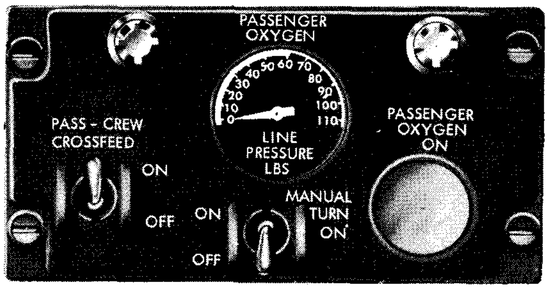
PORTABLE OXYGEN CYLINDER STOWAGE

A



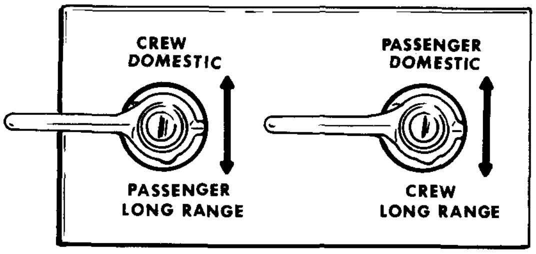
OXYGEN SYSTEM TEST PANEL

B



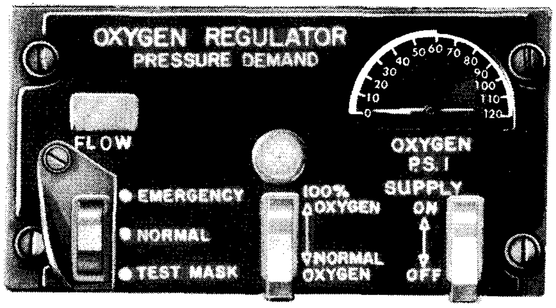
PASSENGER OXYGEN CONTROL PANEL

C



OXYGEN SELECTOR VALVE PANEL

E



CREW OXYGEN CONTROL PANEL

D

Figure 35-1. Oxygen System Components (Sheet 2 of 2)

CYLINDER "A" (39.0 CU. FT. AT 1850 PSI)								
CREW (DOMESTIC) OXYGEN DURATION (MAN HOURS)								
PRESSURE IN CYLINDER P.S.I.G.		CABIN ALTITUDE FT X 1000						
		8	12	15	20	25	30	35 & UP
1850	DILUTION	7.26	7.26	7.26	6.0	5.3	5.6	7.5
	100%			2.7	3.2	4.2	5.5	7.5
1800	DILUTION	7.0	7.0	7.0	5.8	5.1	5.4	7.3
	100%			2.5	3.1	4.1	5.25	7.3
1600	DILUTION	5.75	5.75	5.75	4.75	4.2	4.4	6.0
	100%			2.0	2.5	3.3	4.3	6.0
1400	DILUTION	5.2	5.2	5.2	4.3	3.8	4.0	5.4
	100%			1.8	2.3	3.0	3.9	5.4
1200	DILUTION	4.2	4.2	4.2	3.5	3.1	3.3	4.4
	100%			1.5	1.9	2.5	3.2	4.4
1000	DILUTION	3.4	3.4	3.4	2.8	2.4	2.6	3.5
	100%			1.2	1.5	1.9	2.5	3.5
800	DILUTION	3.0	3.0	3.0	2.5	2.2	2.4	3.2
	100%			1.0	1.3	1.8	2.3	3.2
600	DILUTION	1.6	1.6	1.6	1.4	1.2	1.3	1.7
	100%			0.6	0.7	1.0	1.25	1.7
400	DILUTION	0.8	0.8	0.8	0.7	0.6	0.6	0.9
	100%			0.3	0.4	0.5	0.6	0.9
200	EMERGENCY: 1. USE CROSSFEED VALVE OR ; 2. DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.							

CYLINDER "B" (76.6 CU. FT. AT 1850 PSI)								
CREW (LONG RANGE) OXYGEN DURATION (MAN HOURS)								
PRESSURE IN CYLINDER P.S.I.G.		CABIN ALTITUDE FT X 1000						
		8	12	15	20	25	30	35 & UP
1850	DILUTION	13.8	13.8	13.8	11.4	10.0	10.6	14.4
	100%			4.9	6.1	8.0	10.4	14.4
1800	DILUTION	13.5	13.5	13.5	11.1	9.8	10.4	14.0
	100%			4.75	6.0	7.8	10.1	14.0
1600	DILUTION	11.6	11.6	11.6	9.6	8.5	9.0	12.1
	100%			4.1	5.2	6.75	8.75	12.1
1400	DILUTION	9.3	9.3	9.3	7.7	6.8	7.2	9.7
	100%			3.3	4.1	5.4	7.0	9.7
1200	DILUTION	8.2	8.2	8.2	6.75	6.0	6.3	8.5
	100%			2.9	3.6	4.75	6.1	8.5
1000	DILUTION	6.5	6.5	6.5	5.3	4.7	5.0	6.75
	100%			2.3	2.9	3.75	4.9	6.75
800	DILUTION	4.7	4.7	4.7	3.8	3.4	3.6	4.8
	100%			1.6	2.0	2.7	3.5	4.8
600	DILUTION	3.2	3.2	3.2	2.6	2.3	2.5	3.3
	100%			1.1	1.4	1.9	2.4	3.3
400	DILUTION	1.5	1.5	1.5	1.3	1.1	1.2	1.6
	100%			0.5	0.7	0.9	1.2	1.6
200	EMERGENCY: 1. USE CROSSFEED VALVE OR; 2. DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.							

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Figure 35-2. Oxygen Duration Charts (Sheet 1 of 2)

CYLINDER "A" (39.0 CU. FT. AT 1850 PSI)						
PASSENGER (LONG RANGE) OXYGEN DURATION (MAN HRS)						
	SELECTOR LEVER ON PASS. MASK "DILUTED OXYGEN"				SELECTOR LEVER ON PASS. MASK "100% OXYGEN"	
PRESSURE IN CYLINDER P.S.I.G.	CABIN ALTITUDE FEET				CABIN ALTITUDE FEET	
	12,000	15,000	20,000	25,000	30,000	35,000
1850	2.7	3.4	4.25	5.4	3.8	4.5
1800	2.6	3.2	4.1	5.2	3.7	4.3
1600	2.1	2.7	3.4	4.3	3.0	3.5
1400	1.9	2.4	3.0	3.9	2.8	3.2
1200	1.5	1.9	2.5	3.2	2.25	2.6
1000	1.25	1.5	2.0	2.5	1.8	2.1
800	1.1	1.4	1.8	2.3	1.6	1.9
600	0.6	0.75	1.0	1.2	0.9	1.0
400	0.3	0.4	0.5	0.6	0.4	0.5
200	EMERGENCY: WHEN ALL "A" & "C" CYLINDERS ARE INADEQUATE: 1. TURN ON CROSSFEED VALVE OR ; 2. DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.					

CYLINDER "B" (76.6 CU. FT. AT 1850 PSI)						
PASSENGER (DOMESTIC) OXYGEN DURATION (MAN HRS)						
	SELECTOR LEVER ON PASS. MASK "DILUTED OXYGEN"				SELECTOR LEVER ON PASS. MASK "100% OXYGEN"	
PRESSURE IN CYLINDER P.S.I.G.	CABIN ALTITUDE FEET				CABIN ALTITUDE FEET	
	12,000	15,000	20,000	25,000	30,000	35,000
1850	5.1	6.4	8.1	10.3	7.3	8.5
1800	5.0	6.25	8.0	10.0	7.2	8.3
1600	4.3	5.4	6.9	8.7	6.2	7.2
1400	3.4	4.3	5.5	7.0	5.0	5.75
1200	3.0	3.8	4.8	6.1	4.3	5.0
1000	2.4	3.0	3.8	4.8	3.4	4.0
800	1.7	2.2	2.75	3.5	2.5	2.9
600	1.2	1.5	1.9	2.4	1.7	2.0
400	0.6	0.7	0.9	1.2	0.8	1.0
200	EMERGENCY: 1. USE CROSSFEED VALVE OR; 2. DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.					

CYLINDER "C" (114.0 CU. FT. AT 1850 PSI)						
PASSENGER (LONG RANGE) OXYGEN DURATION (MAN HRS) - PER CYLINDER						
	SELECTOR LEVER ON PASS. MASK "DILUTED OXYGEN"				SELECTOR LEVER ON PASS. MASK "100% OXYGEN"	
PRESSURE IN CYLINDER	CABIN ALTITUDE FEET				CABIN ALTITUDE FEET	
P.S.I.G.	12,000	15,000	20,000	25,000	30,000	35,000
1850	7.7	9.6	12.2	15.5	11.1	12.8
1800	6.8	8.5	10.9	13.8	9.8	11.4
1600	6.5	8.2	10.3	13.1	9.3	10.8
1400	5.5	7.0	8.9	11.25	8.0	9.3
1200	4.5	5.7	7.25	9.1	6.5	7.6
1000	3.6	4.5	5.7	7.25	5.1	6.1
800	2.6	3.25	4.1	5.25	3.75	4.3
600	1.8	2.25	2.8	3.6	2.6	3.0
400	0.9	1.1	1.4	1.8	1.3	1.5
200	EMERGENCY: WHEN ALL "A" & "C" CYLINDERS ARE INADEQUATE: 1. TURN ON CROSSFEED VALVE OR; 2. DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.					

Figure 35-2. Oxygen Duration Charts (Sheet 2 of 2)

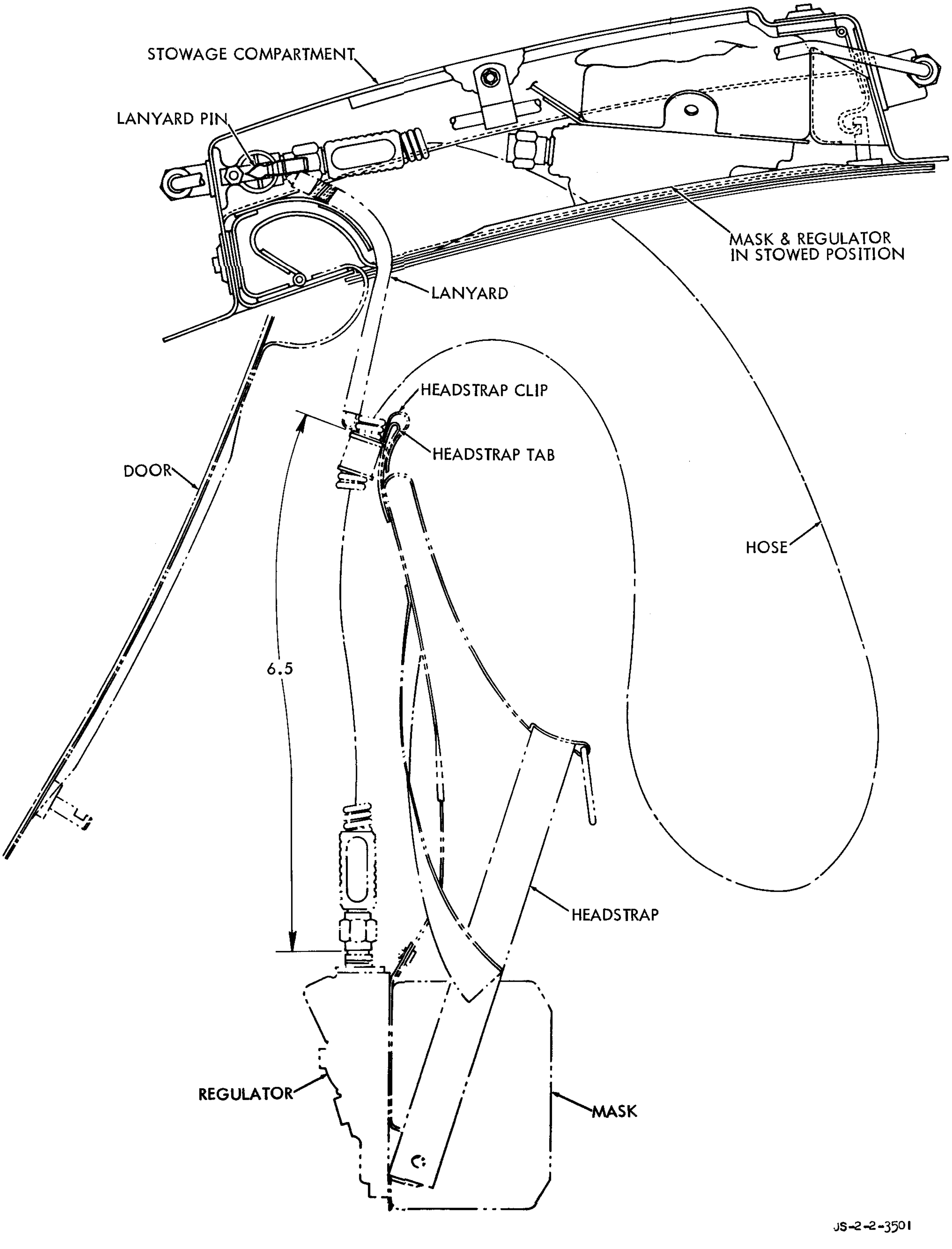
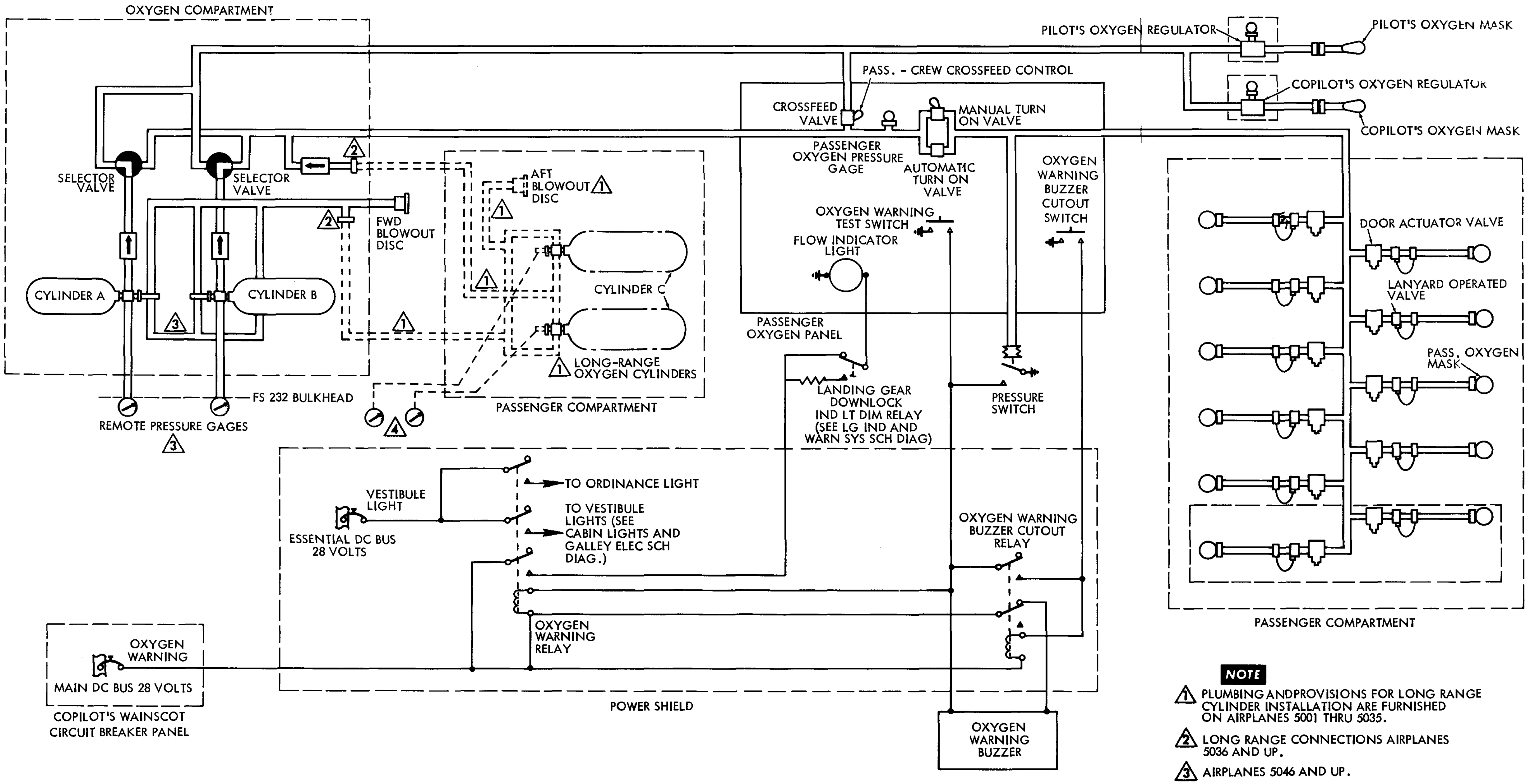


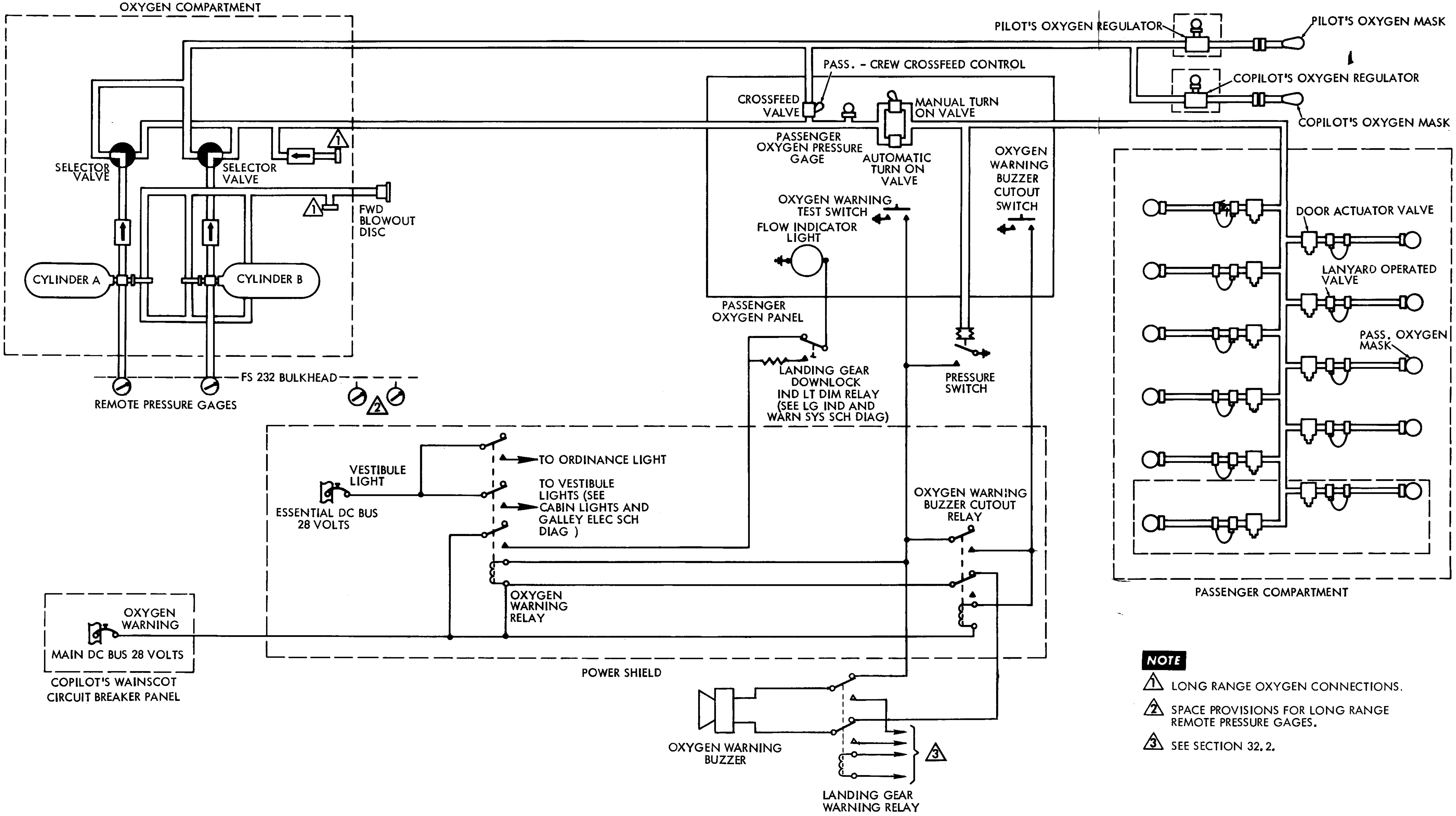
Figure 35-3. Passenger Oxygen Mask and Hose Replacement



NOTE

1. PLUMBING AND PROVISIONS FOR LONG RANGE CYLINDER INSTALLATION ARE FURNISHED ON AIRPLANES 5001 THRU 5035.
2. LONG RANGE CONNECTIONS AIRPLANES 5036 AND UP.
3. AIRPLANES 5046 AND UP.
4. SPACE PROVISIONS FOR LONG RANGE REMOTE PRESSURE GAGES.

Figure 35-4. Oxygen System Schematic Diagram
(Airplanes 5001 thru 5085)
Changed 1 June 1967



NOTE

1 LONG RANGE OXYGEN CONNECTIONS.

2 SPACE PROVISIONS FOR LONG RANGE REMOTE PRESSURE GAGES.

3 SEE SECTION 32.2.

Figure 35-5. Oxygen System Schematic Diagram (Airplanes 5086 and up)

Changed 1 June 1967

CHAPTER 38
WATER/WASTE

Contents Page

DESCRIPTION AND OPERATION 38-1
General 38-1
Water Storage Tank 38-1
Control Valve 38-1
Control Cable 38-1
Water Heater 38-1

OPERATIONAL CHECKOUT 38-2

REMOVAL AND REPLACEMENT 38-2

SERIALIZATION AND CONDITIONS 38-2

38.1 DESCRIPTION AND OPERATION.

38.1.1 GENERAL. The aircraft water system is of the gravity flow type, and is located on the right side in the aft part of the cabin, between FS 550 and the FS 570 pressure bulkhead. An external servicing panel is located at FS 560. This panel contains a filler coupling, with a garden hose adapter fitting, an overflow outlet, and a push-pull control handle which regulates a valve attached to the top of the water storage tank. When the handle is pulled, the storage tank can be filled to its capacity, and excess water will run out of the overflow outlet. After the tank is filled, remove the filling hose and push the handle into a locked position. This closes the filling and overflow lines, and vents the tank to the inside of the aircraft. Directly below the storage tank is a 2-quart water heater, through which the water passes before it reaches the faucet outlet. The water system is drained by turning the valve handle, on the bottom of the water heater, 90 degrees.

38.1.2 WATER STORAGE TANK. The four-gallon, barrel-shaped storage tank is secured to the fuselage structure by two strap clamps. The tank is constructed so that the control valve may be mounted to the upper outboard surface and a drain line may be installed on the lower surface.

38.1.3 CONTROL VALVE. Mounted to the top of the storage tank is a control valve, which is used to open and close the tank filling lines when servicing the water system. The valve also vents the tank to the inside of the aircraft when the filling lines are closed.

38.1.4 CONTROL CABLE. The control cable is secured to the regulating arm of the control valve on top of the storage tank. When the control cable tee handle is pulled, it opens the filling and overflow lines. Pushing the tee handle up closes both lines, and vents the storage tank to the inside of the aircraft.

38.1.5 WATER HEATER. The two-quart, 28-VDC water heater is controlled by a primary thermostat to maintain a water temperature of 110 to 135°F. The heater also contains a low-water thermostat which detects the heating element temperature. This thermostat, if the water level in the heater drops below the level of the element, interrupts the current flow through the element. A high-limit thermostat is provided in the heater to interrupt the power to the heating element in the event of malfunction of the primary thermostat. The high-limit thermostat contains an accessible reset button. The water system is drained through the water heater by turning the valve handle on the bottom of the heater. The circuit breaker for the water heater is located on the copilot's side panel.

38.2 OPERATIONAL CHECKOUT.

a. Close the drain valve on the bottom of the water heater.

b. Check all the water system plumbing and electrical connections for tightness.

c. Check attachment of the water valve to the storage tank.

d. With the tee handle on the servicing panel in the closed position, check the control cable between the upper fuselage attachment and the valve control to assure there is no excess length. Excessive cable length causes the cable to buckle and prevent the valve from being closed completely.

e. Fill the water system until the excess water runs out of the overflow line. Then disconnect the filling line and push the tee handle up into the locked position.

f. Check all of the plumbing connections for leaks.

g. Energize the water heater by pushing the circuit breaker in on the copilot's side panel.

h. Wait ten minutes to check the flow and the temperature of the water at the faucet.

38.3 REMOVAL AND REPLACEMENT.

38.3.1 TANK REMOVAL.

a. Drain the water system completely.

b. Disconnect the control cable from the water valve on top of the tank.

c. Disconnect the plumbing lines from the water valve and from the bottom of the tank.

d. Hold the tank in place, loosen the two strap clamps around the tank, and then remove the tank.

e. To re-install the tank, reverse steps (b) through (d).

Note

Check and adjust the rigging of the water valve remote control as necessary. Change in the tank position may necessitate rerigging.

f. Service the water system and check all plumbing fittings for leaks.

38.3.2 WATER HEATER REMOVAL.

a. Pull the water heater circuit breaker on the copilot's side panel.

b. Drain the water system completely.

c. Disconnect all the electrical and ground connections from the heater.

d. Disconnect all of the plumbing lines from the heater.

e. Remove the screws and nuts which secure the heater to the mounting bracket, and remove heater.

f. To re-install the water heater, reverse steps (b) through (d).

g. Service the water system and check all plumbing fittings for leaks.

h. Energize the water heater by pushing in the circuit breaker on the copilot's side panel.

i. Wait ten minutes to check the flow and temperature of the water at the faucet.

38.4 SERIALIZATION AND CONDITIONS.

38.4.1 SERIALIZATION. The contents of this chapter on the wash water system apply to the following JetStar airplanes: 5001, 5003 through 5045.

38.4.2 CONDITIONS. This chapter applies to the basic wash water system as installed by Lockheed Aeronautical Systems Company. It does not apply to the sink, faucets, drains, etc., or modifications made to the basic system by the customer or an interior conversion center.

CHAPTER 39
DRAG CHUTE

Contents Page

DESCRIPTION 39-1
General 39-1
Drag Chute 39-2
Drag Chute Controls 39-2
System Operation 39-2

OPERATIONAL CHECKOUT 39-2
Drag Chute Control System 39-2

REMOVAL AND INSTALLATION 39-2
Removal of the Drag Chute 39-2
Installation of the Drag Chute 39-2

ADJUSTMENT 39-3
Rigging of the Drag Chute Release System 39-3

PACKING 39-3
Tools Required 39-3
Preparation 39-3

FIGURES
39-1. Drag Chute Controls 39-5
39-2. Drag Chute Installation 39-6
39-3. Drag Chute Mechanism Locking Positions (Observed through Drag Chute Inspection Windows) 39-7
39-4. Drag Chute Release System Rigging Instructions 39-8
39-5. Drag Chute Components 39-10
39-6. Drag Chute Packing 39-11

39.1. DESCRIPTION.

39.1.1 GENERAL. (See figure 39-1.) Airplanes 5001 through 5120 were delivered with an operable drag chute system. Airplanes 5121 and subsequent have provisions for installing an operable system if required. The drag chute system provides a means of rapid deceleration and is a backup device to the wheel brakes and thrust-reversers. The drag chute is installed and attached in the airplane by means of a D-ring at the end of the chute pack riser, and is secured to the retaining hook of the locking mechanism at the bottom of the tail cone compartment. A mechanical linkage controls deployment and jettisoning of the drag chute. The drag chute is deployed after airplane touchdown at speeds up to 140 knots. An inspection window is provided on the left side of the fuselage, in the tail cone compartment, for checking the latch mechanism lock, on airplanes 5046 and up and on previous airplanes having Service Bulletin 329-86 incorporated.

39.1.2 DRAG CHUTE. The drag chute consists of a pilot chute and a main chute. The main chute is 18 feet in diameter and of ring-slot construction. The main chute canopy panels, or gores, make a pattern of circles from the outer circumference to the center vent, leaving circular slots between the panels. These slots are vents that enable the main chute to be an effective drag, without being destroyed at the high speeds of deployment. The components of the drag chute are fastened together with interlocking loops, for ease of maintenance without the use of tools. The riser, suspension lines, and canopy are combined into one component. The main chute and pilot chute are packed in a deployment bag. When the drag chute pack is installed in the airplane, the pilot chute is held in the deployment bag (under spring tension) by a rip cord pin. A safety feature of the drag chute installation is a shear pin which releases the drag chute before the airplane is damaged, if deployment should occur at excessively high speeds.

39.1.3 DRAG CHUTE CONTROLS. (See figure 39-1.) The controls for the drag chute consist of a chute release handle on the pilot's instrument panel, a cable and pulley linkage assembly, a chute locking mechanism, and a latch mechanism for the chute compatment doors.

39.1.4 SYSTEM OPERATION. To operate the drag chute system, pull the drag chute release handle out to the first stop. This initial movement positions a cam in the locking mechanism and locks a retaining hook in the closed position. The chute door latch trips, opening the doors, and allows the pilot chute to deploy. To jettison the drag chute, pull the DRAG CHUTE release handle out to the first stop, turn the handle 90° clockwise, and pull out to the second stop. This jettisons the drag chute by unlocking the drag chute retaining hook. The hook does not lock until the handle is pulled to the first stop; this allows free fall of the chute in event the compartment doors should accidentally open.

39.2 OPERATIONAL CHECKOUT.

39.2.1 DRAG CHUTE CONTROL SYSTEM. Perform the following checkout with the drag chute removed from the airplane. Two men are required for this checkout. Station man A in the pilots' compartment to operate the controls, and station man B at the tail cone to observe the operation of the control system.

a. Man A: Pull the DRAG CHUTE release handle to the first stop.
RESULT: The force required to pull the handle should not exceed 50 pounds, measured with a spring scale. The chute compartment doors should open.

b. Man B: Lift up on the retaining hook.
RESULT: The hook should be locked closed.

c. Man A: Rotate the control handle 90° clockwise, and pull it to the second stop.
RESULT: The force required to pull the handle should not exceed 50 pounds.

d. Man B: Lift up on the retaining hook.
RESULT: The hook should be held closed by spring tension, and should open without requiring excessive effort.

e. Man B: Install the drag chute in accordance with the instructions in this section.

39.3 REMOVAL AND INSTALLATION.

30.3.1 REMOVAL OF THE DRAG CHUTE. This procedure is for removing a drag chute from the airplane without permitting deployment. Two men are required for this removal operation. Station man A in the pilots' compartment, station man B at the tail cone and perform the following procedure:

a. Man B: Hold the chute doors closed while man A pulls the DRAG CHUTE release handle to its second stop and holds it there.

b. Man B: Allow the chute doors to open slightly, and insert a ripcord pin or temporary pin in the cone which extends through the pilot chute compartment flaps, to keep the pilot chute compartment closed.

Note

While inserting the pin, open the doors only enough to get a hand inside. If the doors are opened too much, the pilot chute may be ejected.

c. Man B: After the pin is in place, allow the door to open, and remove the chute from the tail compartment.

d. Man A: Return the DRAG CHUTE release handle to normal.

39.3.2 INSTALLATION OF THE DRAG CHUTE. (See figures 39-1, 39-2 and 39-3.)

a. Check that the plastic foam spacer has been installed at the forward end of the drag chute compartment.

b. Pull the DRAG CHUTE release handle located on the bottom of the pilot's instrument panel out to the first stop, turn the handle 90° clockwise and then pull out to the second stop. This will unlatch the compartment doors and set the anchor hook mechanism to the proper position for chute installation.

c. Return the DRAG CHUTE release handle to the normal stowed position on the pilot's instrument panel.

CAUTION

Ensure that the DRAG CHUTE release handle is in the fully stowed position. It must not be pulled out, except for chute inspection, removal, or deployment.

39.3.2 (Cont).

d. Inspect the drag chute pack riser locking loop for proper installation; ensure that the chute pack flaps are securely closed. The riser locking loop must not extend more than one inch through the flap loop.

Note

The drag chute pack riser locking loop should always lie against the pack, as it leads out and away from the pack towards the free end.

e. Stow all excess riser length at the forward end of the chute pack. The riser should lie flat and straight, and be centered under the chute pack.

f. Install the drag chute pack in the chute compartment, with the pilot chute towards the aft end of the airplane.

Note

If the anchor hook is locked repeat step (b).

g. Check the anchor hook to see that it can be moved freely. This safety feature will allow the chute pack to drop free, if the compartment doors should accidentally open.

h. Reach up and under the chute pack, grasp and pull the D-ring aft just far enough to connect it to the anchor hook.

CAUTION

To prevent damage to the aircraft skin, compartment doors must be closed by placing the palm of the hand over the latch indicator and squeezing the doors together.

i. Close the drag chute compartment doors. Allow the safety pin to remain in the pilot chute and the "REMOVE BEFORE FLIGHT" flag to hang between the doors.

j. On airplanes 5046 and up, and on previous airplanes having Service Bulletin 329-86 incorporated, look through the drag chute inspection window, located on the aft left side of the fuselage, check the chute lock (red) and the chute catch (white) for the proper latched position. (See figure 39-3.)

39.4.1 ADJUSTMENT.

39.4.2 RIGGING OF THE DRAG CHUTE RELEASE SYSTEM. (See figure 39-4.)

39.5.1 PACKING.

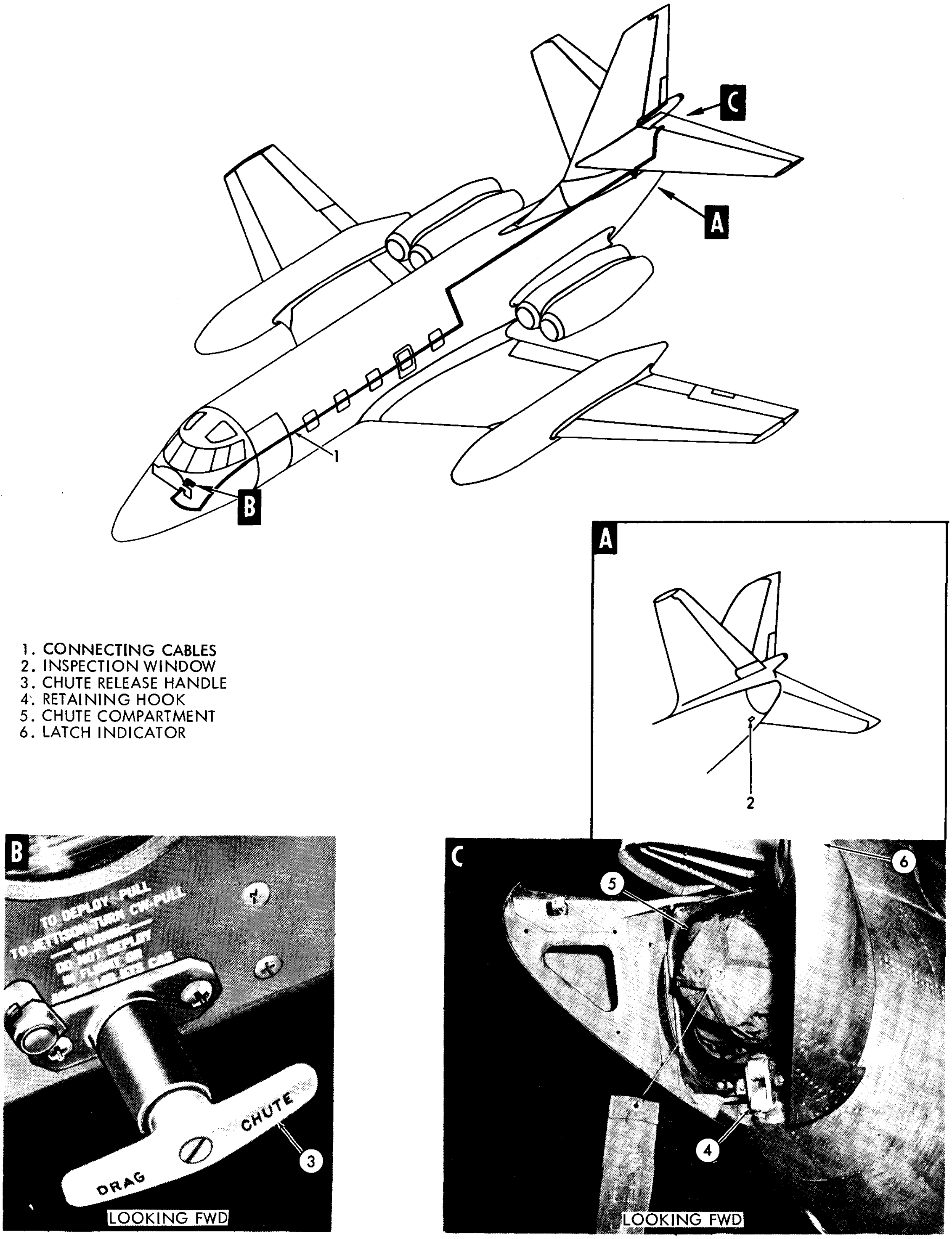
39-5.2 TOOLS REQUIRED. The following tools are required to pack the drag chute:

TOOL	AMOUNT/SIZE	APPLICATION
Rope	Two, 6 foot lengths	To stretch and apply tension to the main canopy prior to folding.
Box	One 10x10 inches	To hold the deployment bag while folding the main canopy.
Weights	Three, weighing 3 to 6 pounds each	To hold canopy in position during fold-operation. If shot bags or smooth weights are not available it is permissible to use clean wrenches as a substitute.
Line Separator	One	To keep the suspension lines properly separated during folding operation. If line separator is not available it is permissible to use a clean tool box as a substitute.

39.5.3 PREPARATION. To prepare the drag chute for packing inspect the chute for torn fabric, broken or friction-burned suspension lines and bridles, open seams, dirt and moisture. Any indication of broken or burned suspension lines is cause for replacing the drag chute. The presence of moisture is extremely critical since it will freeze at high altitude and cement the chute in the bag. If time permits, the chute should be aired for twelve hours following each deployment. Spread the chute on a clean surface protected from the elements and direct sunlight. The chute packing area must be a minimum of 45 feet long and 3 feet wide (preferably a table) with provisions at each and for attaching a length of rope to stretch and apply tension to the main canopy during packing. Refer to figure 39-5 for the drag chute components and to figure 39-6 for the correct method of packing the chute.

Note

Strict conformance to the instructions in figure 39-6 is mandatory to insure a safe and reliable installation.



1. CONNECTING CABLES
2. INSPECTION WINDOW
3. CHUTE RELEASE HANDLE
4. RETAINING HOOK
5. CHUTE COMPARTMENT
6. LATCH INDICATOR

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Figure 39-1. Drag Chute Controls

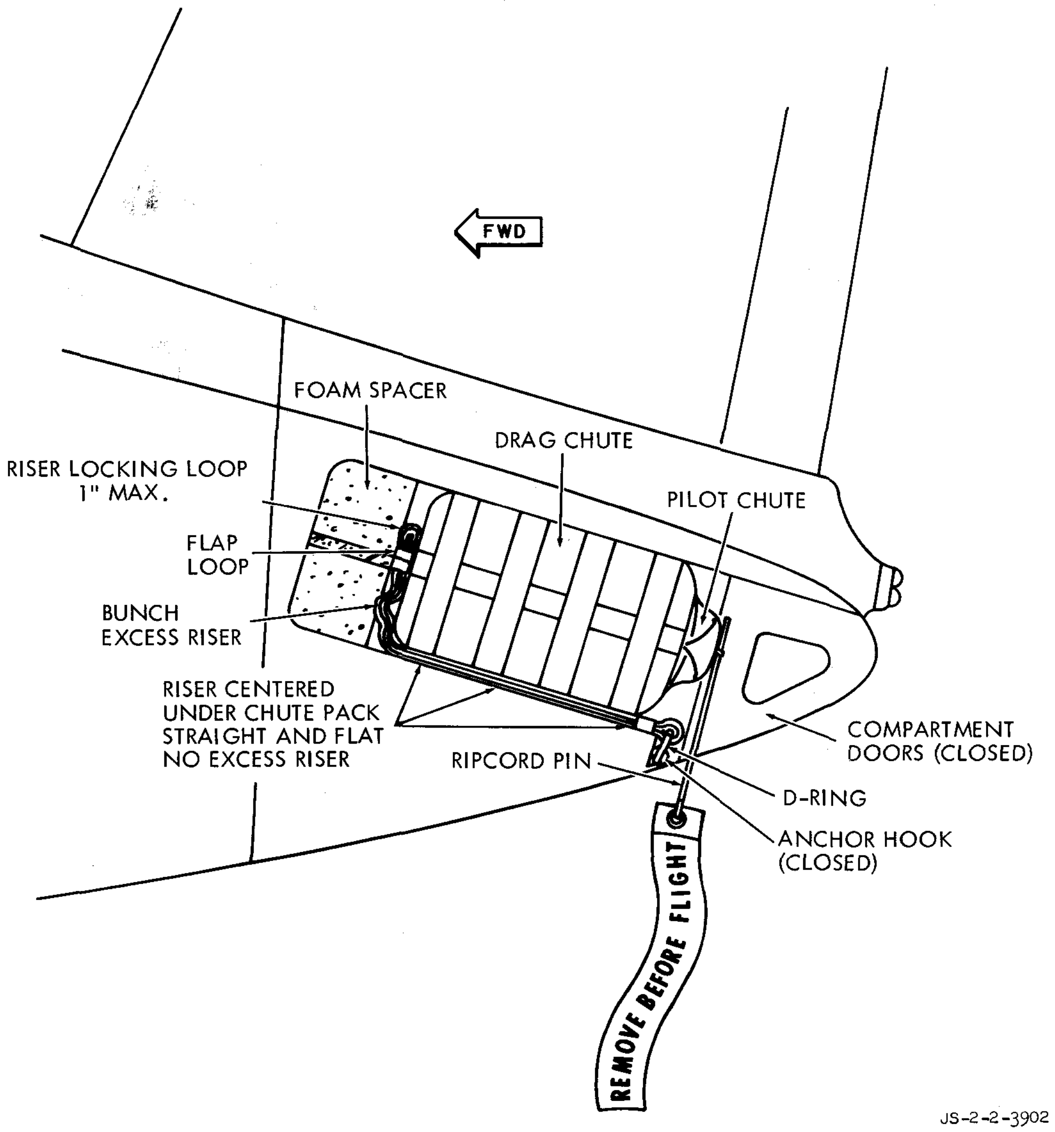
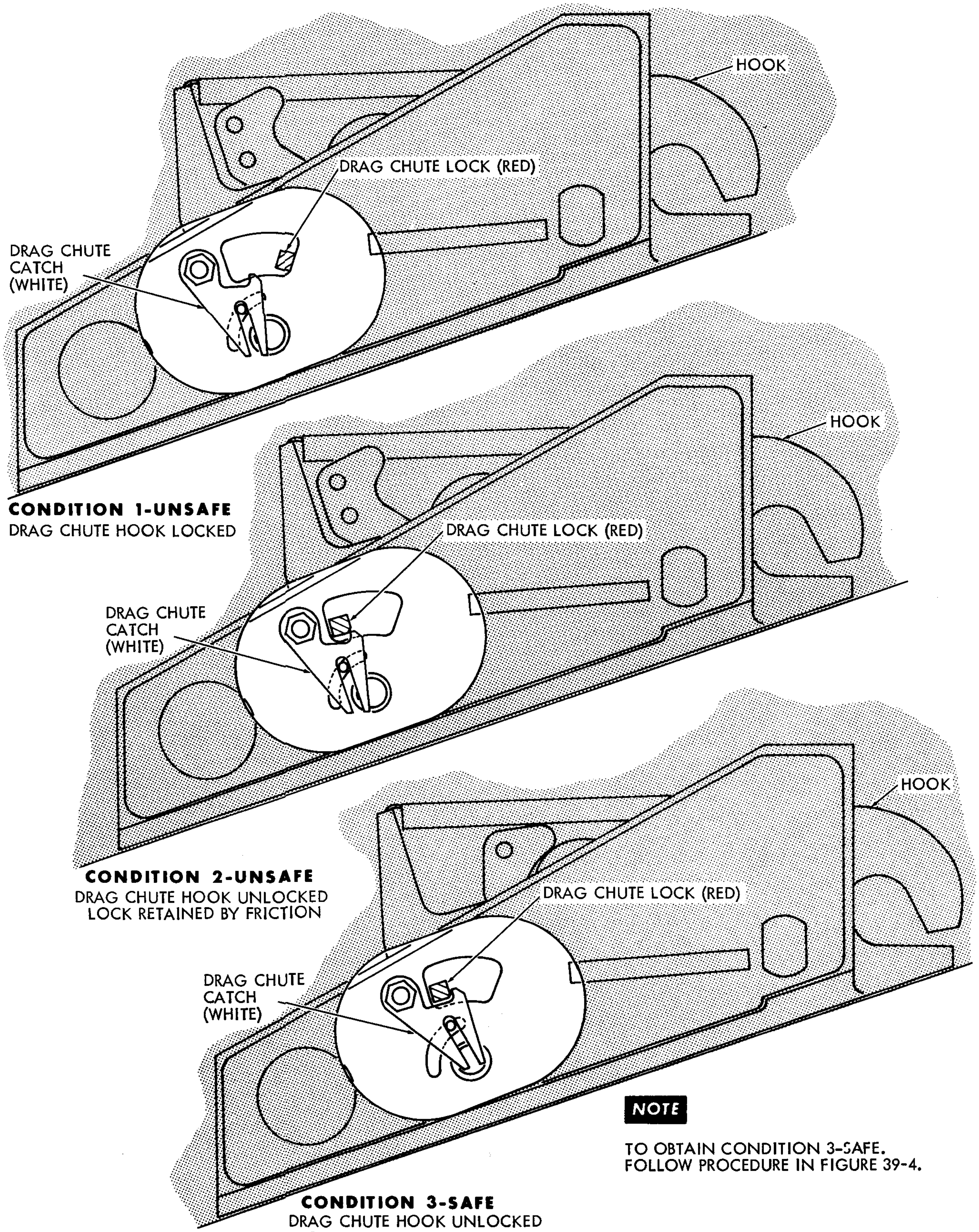


Figure 39-2. Drag Chute Installation



CONDITION 1-UNSAFE
DRAG CHUTE HOOK LOCKED

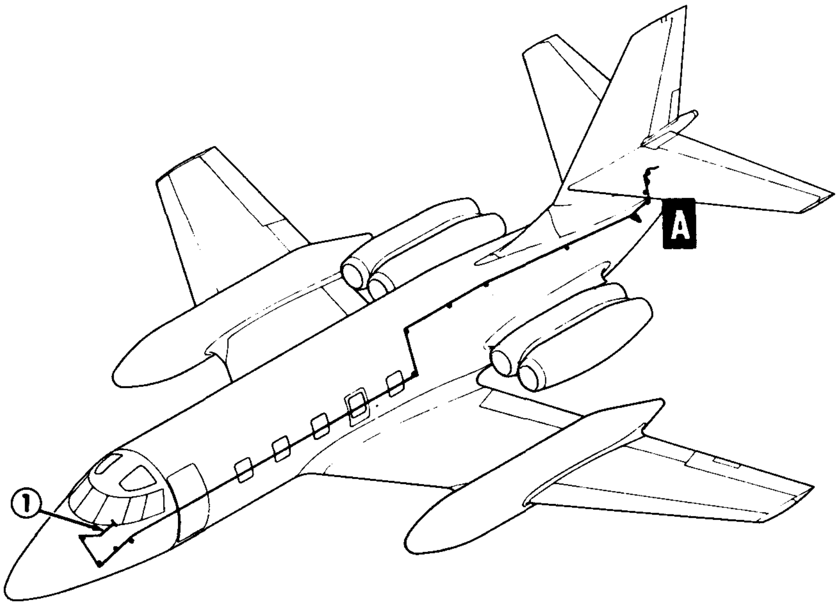
CONDITION 2-UNSAFE
DRAG CHUTE HOOK UNLOCKED
LOCK RETAINED BY FRICTION

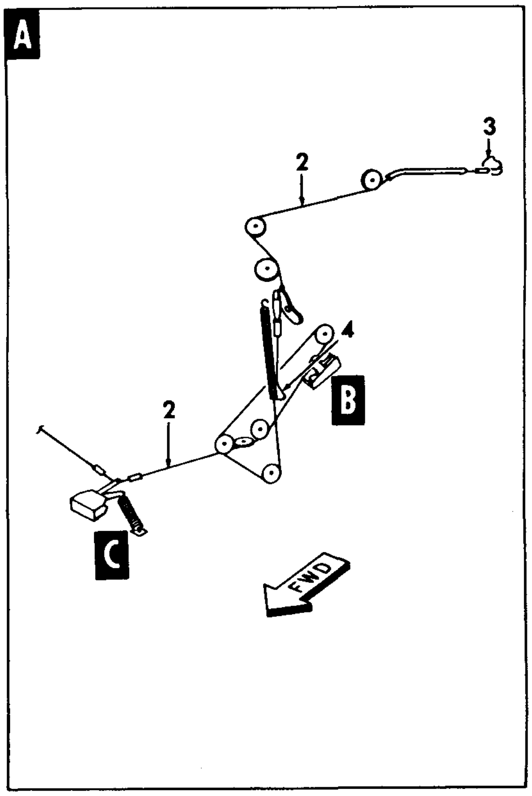
NOTE

TO OBTAIN CONDITION 3-SAFE. FOLLOW PROCEDURE IN FIGURE 39-4.

CONDITION 3-SAFE
DRAG CHUTE HOOK UNLOCKED
LOCK IN NORMAL POSITION

Figure 39-3. Drag Chute Mechanism Locking Positions (Observed Through Drag Chute Inspection Windows)





STEP 1

PLACE THE HOOK (8) IN THE CLOSED POSITION, CLAMPING ON A SPARE CHUTE RING OR ANY 0.75 INCH DIAMETER PIN (9).

STEP 2

PLACE THE HOOK LOCK LUG (5) IN THE UNLOCKED POSITION, WITH THE CATCH (10) HOLDING IT IN PLACE.

STEP 3

PULL THE LEVER (6) FORWARD 0.60 TO 0.90 INCH. RESULT: THE CATCH (10) SHOULD RELEASE THE LUG (5), ALLOWING THE LOCK TO SHIFT TO THE LOCKED POSITION, LOCKING THE HOOK (8) CLOSED.

STEP 4

PULL THE LEVER (6) THROUGH THE REMAINDER OF ITS TRAVEL FORWARD. RESULT: THE LUG (5) SHOULD SHIFT BACK TO THE UNLOCK POSITION, ALLOWING THE HOOK (8) TO OPEN.

STEP 5

RETURN THE HOOK (8) TO THE POSITION DESCRIBED IN STEP 1. THIS IS THE NORMAL STOWED POSITION.

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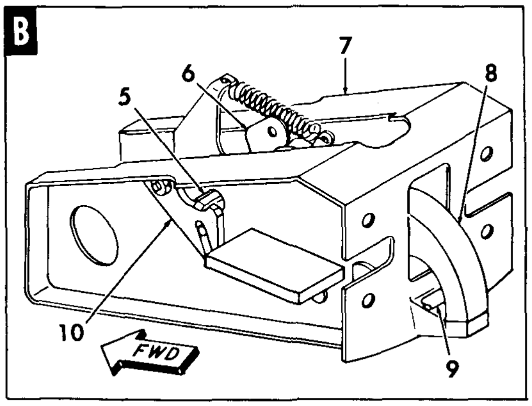
Figure 39-4. Drag Chute Release System Rigging Instructions (Sheet 1 of 2)



WARNING

DO NOT MOVE THE LEVER (6) THROUGHOUT THE REMAINDER OF RIGGING PROCEDURE. THIS COULD CAUSE THE CHUTE TO BE JETTISONED PREMATURELY.

STEP 6

PLACE THE PULL HANDLE (1) IN THE FORWARD POSITION.

STEP 7

TIGHTEN THE TURNBUCKLE (14) UNTIL A 1/4-INCH RIG PIN CAN BE FREELY INSERTED AND REMOVED FROM THE HOLE (11) THROUGH THE BRACKET (12) AND THE LEVER (16). RESULT: THIS WILL GIVE THE CABLE APPROXIMATELY 15 POUNDS OF RIG TENSION.

STEP 8

RIG THE CABLE (2) FROM THE LEVER (16) TO THE DOOR MECHANISM (3), LEAVING 0.40 TO 0.50 INCHES OF SLACK AS SHOWN AT (4). RESULTS: THIS ASSURES THE CHUTE HOOK (8) OF BEING CLOSED BEFORE THE DOOR IS OPENED.

STEP 9

POSITION THE LEVER (17) SO THAT A 1/4-INCH RIG PIN CAN BE INSERTED IN THE HOLE (13) THROUGH THE BRACKET (12) AND THE LEVER (17).

STEP 10

WITH THE PIN IN PLACE AS IN STEP 9, TIGHTEN THE TURNBUCKLE (15) UNTIL THE SLACK BETWEEN LEVER (16) AND LEVER (6) IS REMOVED.

WARNING

DO NOT MOVE LEVER (6) AND DO NOT REMOVE THE 0.40 TO 0.50 INCHES OF SLACK AT (4). THIS COULD CAUSE THE DRAG CHUTE TO BE JETTISONED PREMATURELY.

STEP 11

REMOVE THE 1/4-INCH RIG PIN FROM THE HOLE (13).

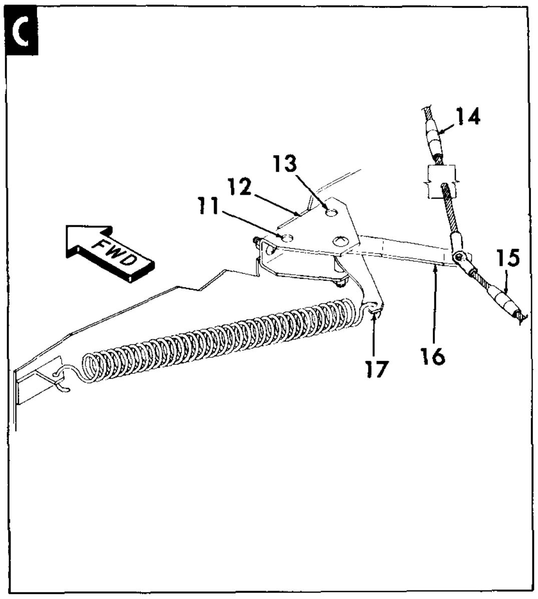


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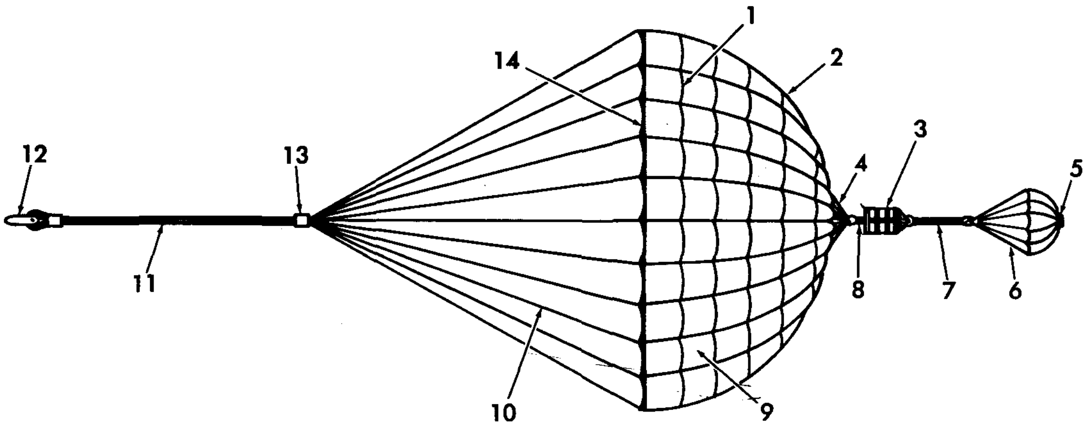
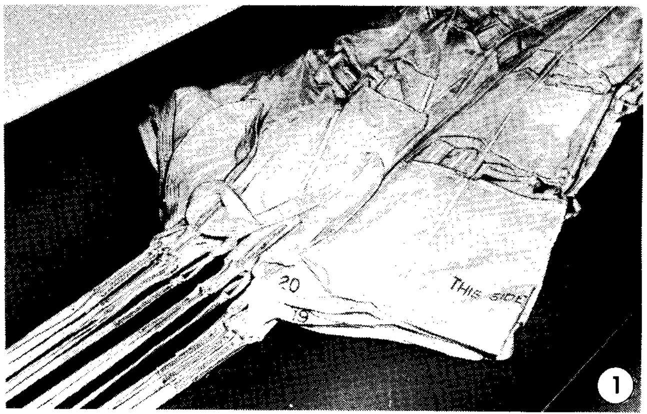
Figure 39-4. Drag Chute Release System Rigging Instructions (Sheet 2 of 2)



1. RING SLOT VENT
2. MAIN CANOPY
3. DEPLOYMENT BAG
4. TOP VENT
5. CROWN
6. PILOT CHUTE
7. BRIDLE (PLOT CHUTE)
8. BRIDLE (DEPLOYMENT BAG)
9. GORES
10. SUSPENSION LINES
11. RISER
12. D-RING (LINK ASSEMBLY)
13. CONFLUENT POINT
14. SKIRT AND POCKET BANDS

Figure 39-5. Drag Chute Components



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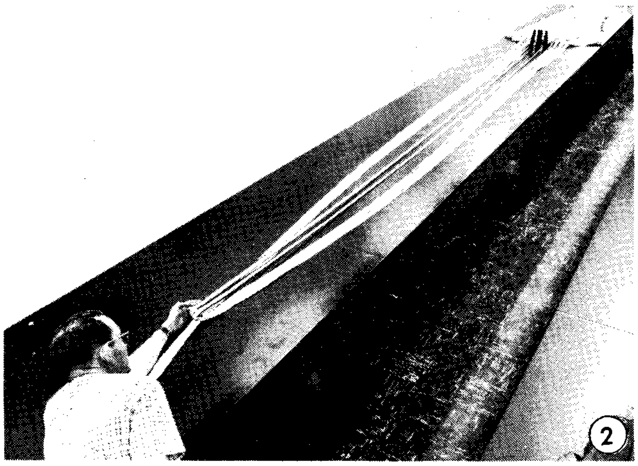
STEP 1

LAYOUT CHUTE ON A CLEAN TABLE KEEPING WORDS "THIS SIDE OUT" ON PANEL GORES UP. POCKET BANDS AT BOTTOM OF CANOPY MUST BE TURNED TOWARD THE OUTSIDE. GORE PANELS 10 AND 11 MUST BE ON BOTTOM. GORE PANELS 1 AND 20 MUST BE ON TOP.



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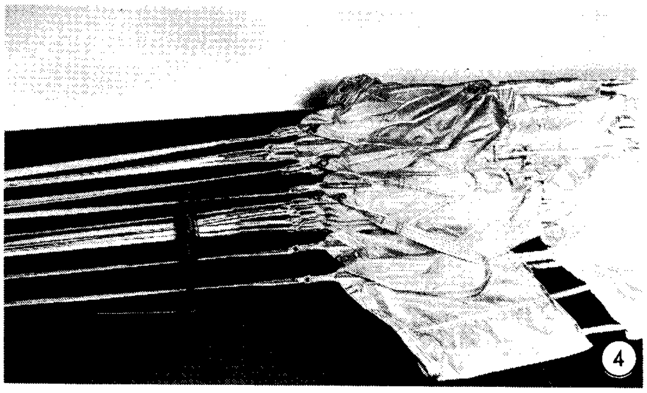
STEP 2

SEPARATE THE TWO GROUPS OF LINES, KEEPING LINES FREE FROM TANGLES. PLACE LINES 1 AND 20 ON TOP.

③

STEP 3

SECURE DEPLOYMENT BAG BRIDLE TO ANCHOR POINT (USING LENGTH OF ROPE). SEPARATE TOP VENT LINES SO LINES CORRESPONDING TO SUSPENSION LINES 1 AND 20 ARE ON TOP.
PULL SLACK OUT OF CHUTE, AND SECURE RISER TO ANCHOR POINT (USING LENGTH OF ROPE).



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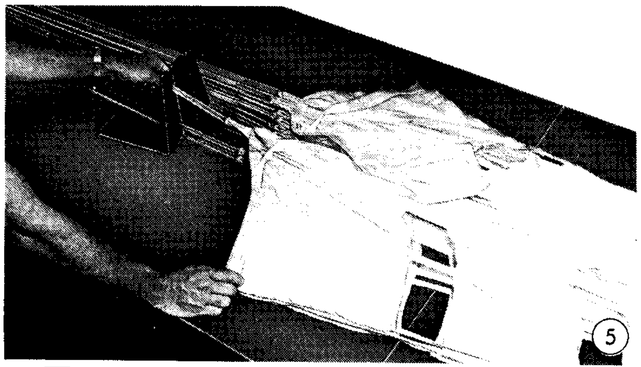
STEP 4

INSERT SUSPENSION LINE 10 IN LEFT SLOT OF SEPARATOR AND LINE 11 IN THE RIGHT SLOT.
THROW THE RIGHT-HAND GROUP OF SUSPENSION LINES AND CANOPY GORES OVER THE LEFT-HAND GROUP.



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STEP 5

INSERT SUSPENSION LINES 12 THRU 20 CONSECUTIVELY AND ONE AT A TIME IN THE RIGHT-HAND SLOT OF THE SEPARATOR, WHILE FOLDING GORES 11 THRU 20 ONE AT A TIME AND MAKING SURE THAT THE GORES ARE SMOOTH AND IN PLACE.

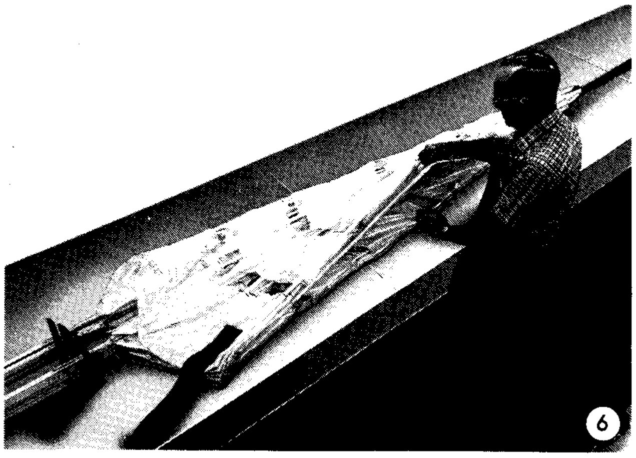

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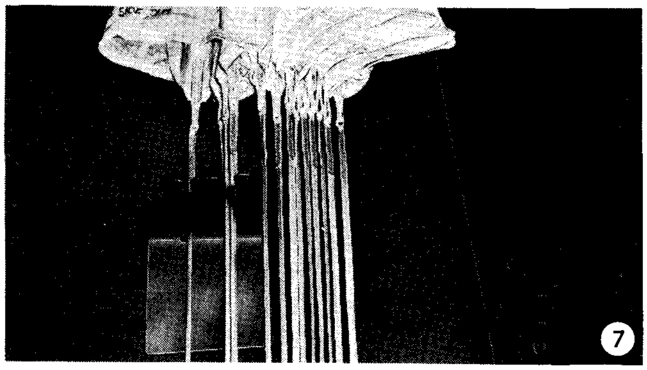
Figure 39-6. Drag Chute Packing (Sheet 1 of 5)

STEP 6

COUNT AND WEIGHT 10 PLEATS AT BOTTOM OF CANOPY. COUNT AND WEIGHT 10 PLEATS MIDWAY BETWEEN BOTTOM OF CANOPY AND TOP VENT.

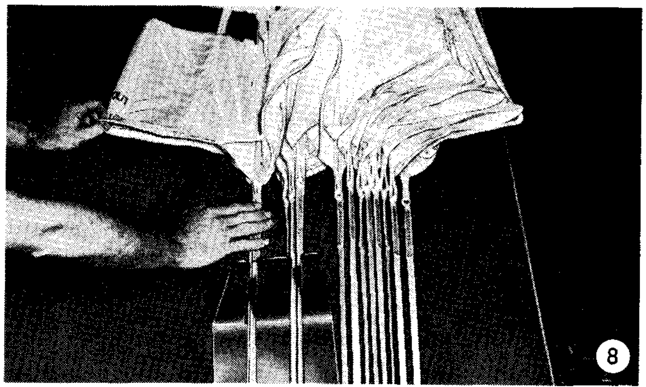
NOTE

SMOOTH AND STRAIGHTEN PLEATS WHEN COUNTING.



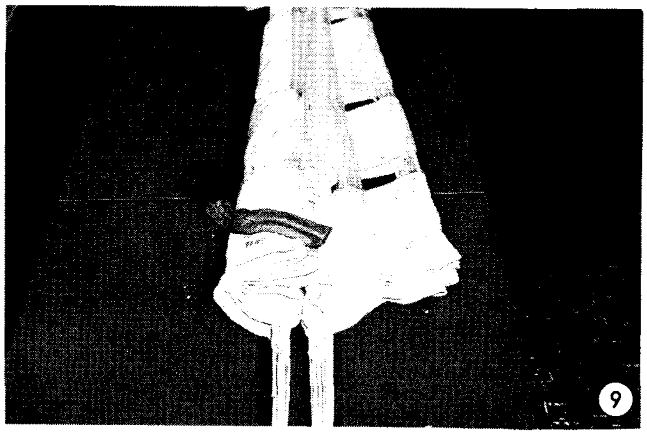
STEP 7

THROW THE LEFT-HAND GROUP OF SUSPENSION LINES AND CANOPY GORES OVER THE RIGHT-HAND GROUP.



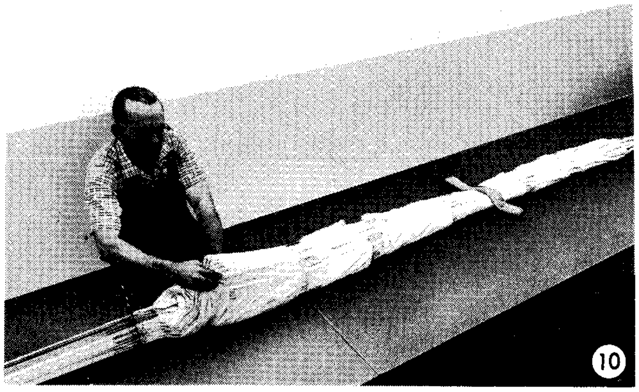
STEP 8

INSERT SUSPENSION LINES 9 THRU 1 CONSECUTIVELY AND ONE AT A TIME IN THE LEFT-HAND SLOT OF THE SEPARATOR, WHILE FOLDING GORES 10 THRU 1 ONE AT A TIME, MAKING SURE THAT GORES ARE SMOOTH AND IN PLACE.



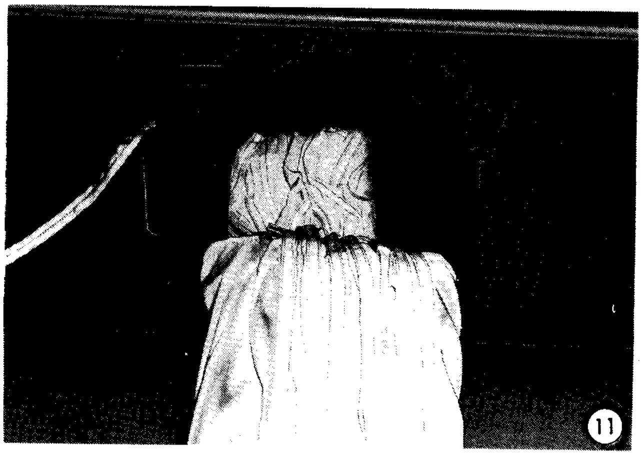
STEP 9

COUNT AND WEIGHT 10 PLEATS AT BOTTOM OF CANOPY.
COUNT AND WEIGHT 10 PLEATS MIDWAY BETWEEN BOTTOM OF CANOPY AND TOP VENT.
SMOOTH AND STRAIGHTEN PLEATS WHEN COUNTING.
FOLD LEFT-HAND PLEATS TO CENTERLINE OF CANOPY, AND WEIGHT AT BOTTOM.



STEP 10

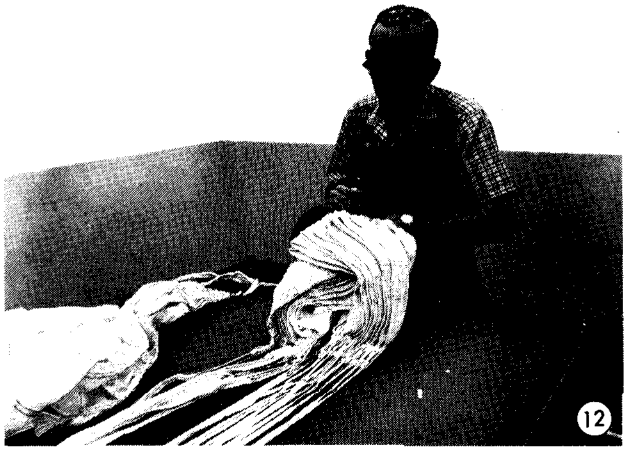
FOLD RIGHT-HAND PLEATS TO CENTERLINE OF CANOPY, FORMING A FLAT SHAPE EQUAL TO THE WIDTH OF THE DEPLOYMENT BAG.
WEIGHT AT BOTTOM OF CANOPY AND MIDWAY BETWEEN BOTTOM AND TOP VENT.
RELIEVE TENSION FROM RISER AND DEPLOYMENT BAG BRIDLE.



STEP 11

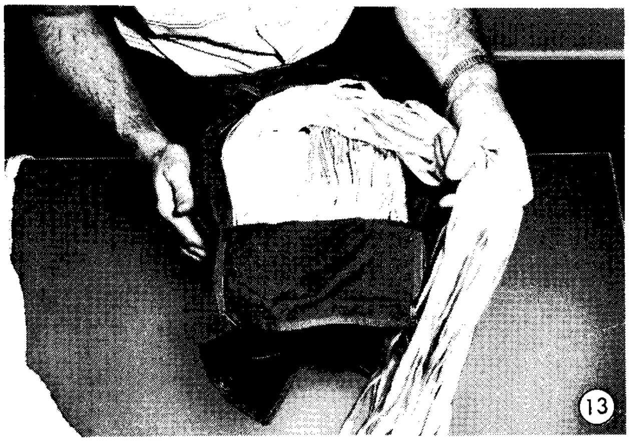
OPEN SUSPENSION LINE STORAGE FLAP ON DEPLOYMENT BAG.
PLACE DEPLOYMENT BAG IN 10X10-INCH BOX, AND COIL CANOPY BRIDLE IN BOTTOM OF BAG.
REMOVE WEIGHTS AND, USING ACCORDION FOLD, PACK CANOPY IN DEPLOYMENT BAG.

Figure 39-6. Drag Chute Packing (Sheet 2 of 5)



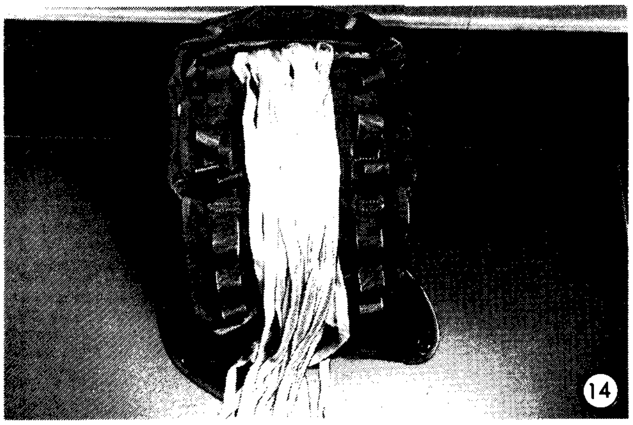
STEP 12

PACK CANOPY IN DEPLOYMENT BAG. KEEP FOLDS SAME WIDTH AS BAG.



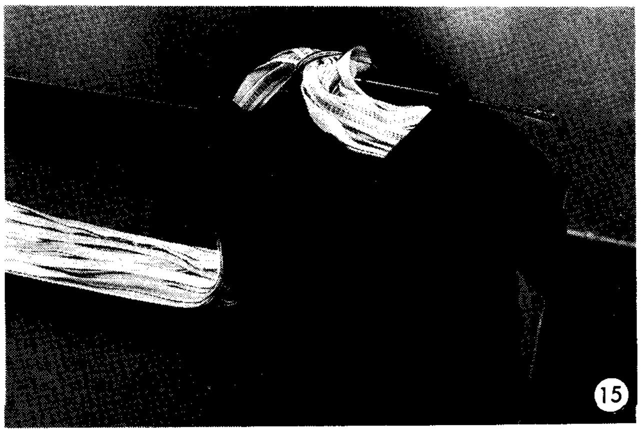
STEP 13

COIL APPROXIMATELY 18 INCHES OF SUSPENSION LINES ON TOP OF CANOPY, KEEPING CONFLUENT POINT BETWEEN LOOPS ON DEPLOYMENT BAG FLAP.



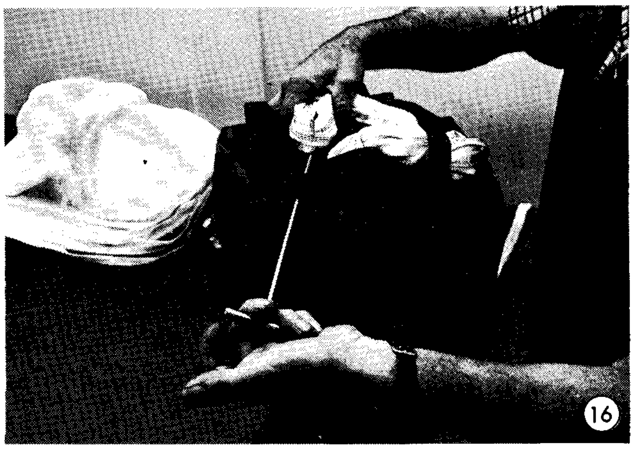
STEP 14

CLOSE DEPLOYMENT BAG INNER FLAP. ROUTE SUSPENSION LINES OVER THE CENTER OF INNER FLAP.



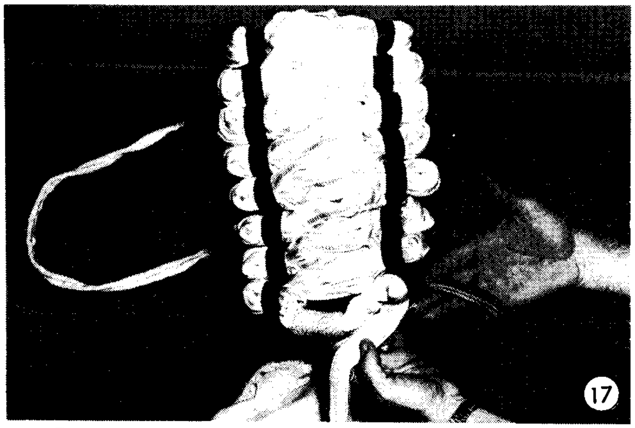
STEP 15

CLOSE DEPLOYMENT BAG SHORT FLAPS. PULL FIRST TWO LOOPS AT EACH END OF INNER FLAP THRU LOCKING SLOTS ON SHORT FLAPS.



STEP 16

USING WIRE HOOK OR PIECE OF STRING, PULL SUSPENSION LINES THRU INNER FLAP LOOPS TO SECURE SHORT FLAPS OVER INNER FLAP.



STEP 17

STARTING ON RIGHT-HAND SIDE (FACING SHORT FLAP), ROUTE SUSPENSION LINES BACK AND FORTH ACROSS INNER FLAP AND INSIDE OF OUTER FLAP.

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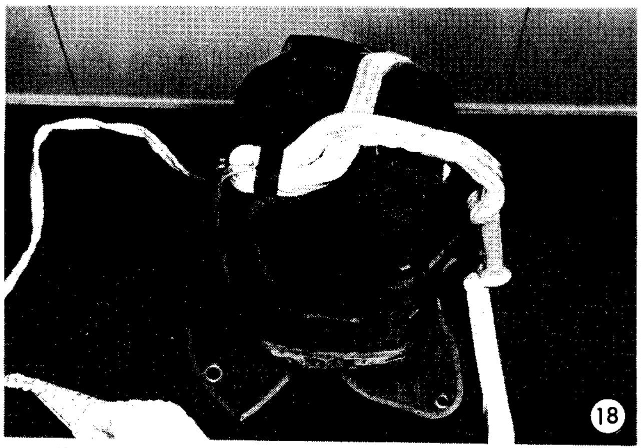
Figure 39-6. Drag Chute Packing (Sheet 3 of 5)

NOTE

THE CONFLUENT POINT SHOULD FALL BETWEEN THE LAST LOOPS ON FLAP.

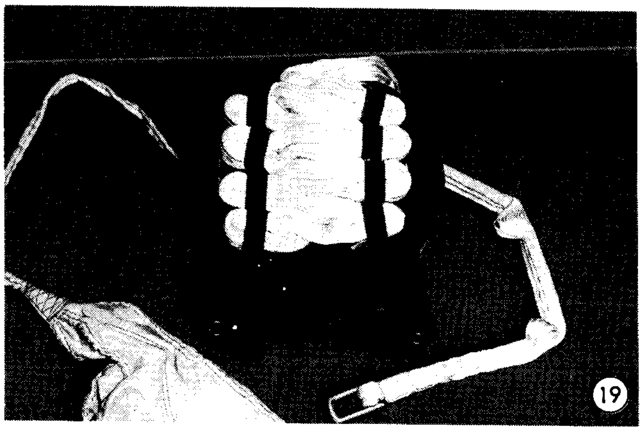


STEP 18

FOLD LOADED SIDE OF OUTER FLAP OVER INNER FLAP.

NOTE

TAKE SPECIAL NOTE OF RISER ROUTING.

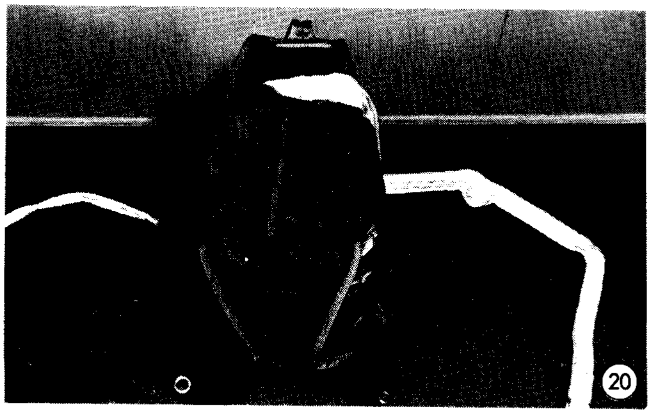


STEP 19

ROUTE RISER BACK AND FORTH ACROSS OUTER FLAP.

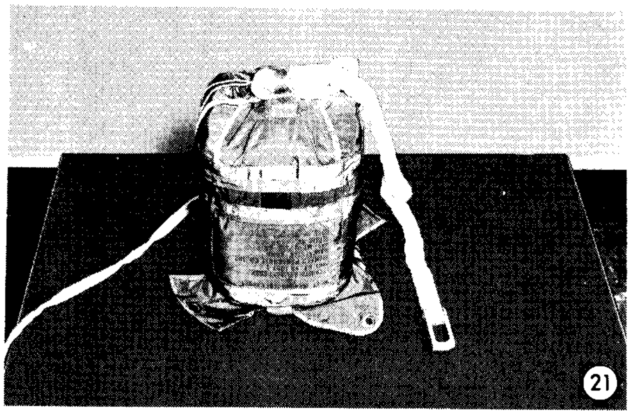
NOTE

APPROXIMATELY 24 TO 36 INCHES OF RISER SHOULD REMAIN AFTER STOWING.



STEP 20

CLOSE SIDE FLAPS. CLOSE WEBBED LOOP FLAP. CLOSE SQUARE HOLED FLAP, INSERTING WEBBED LOOP THRU SQUARE HOLE.

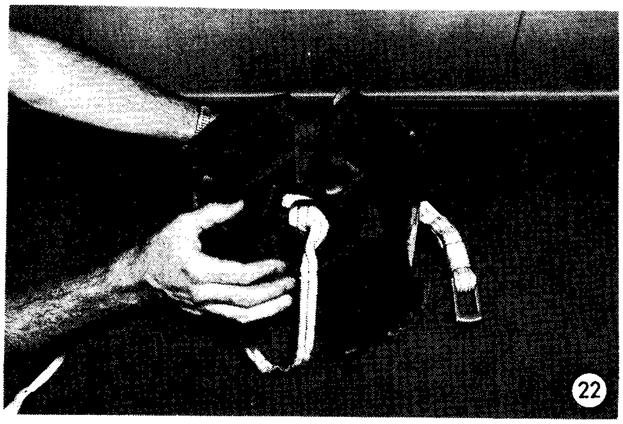


STEP 21

PULL RISER THRU WEBBED LOOP TO SECURE DEPLOYMENT BAG FLAPS; ROUTE RISER EXACTLY AS SHOWN.

NOTE

RISER SHOULD EXTEND 3/4 TO 1 INCH THROUGH WEBBED LOOP.



STEP 22

POSITION DEPLOYMENT BAG IN 10X10-INCH BOX WITH PILOT CHUTE COMPARTMENT UP.

CAUTION

NEVER LIFT OR CARRY CHUTE ASSEMBLY BY THE WEBBING. FOLD SLACK IN WEBBING TO THE CENTER OF THE PACK.

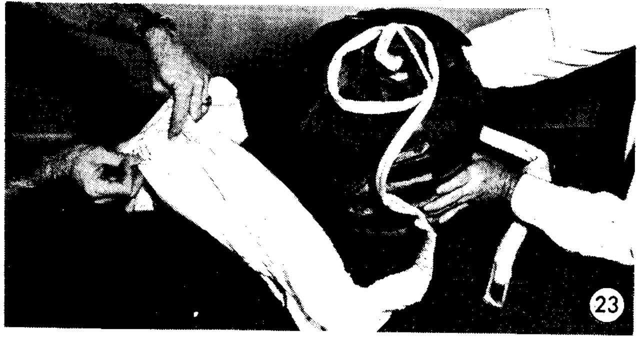


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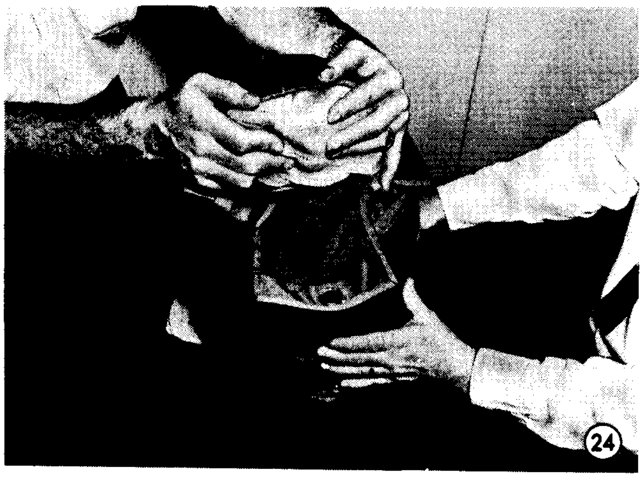
Figure 39-6. Drag Chute Packing (Sheet 4 of 5)

STEP 23

GRASP THE PILOT CHUTE AS SHOWN AND, WHILE COMPRESSING THE EJECTION SPRING, FOLD THE CHUTE USING ACCORDION FOLD.
COIL PILOT CHUTE BRIDLE LOOSELY AROUND BRIDLE ATTACHMENT TO DEPLOYMENT BAG.

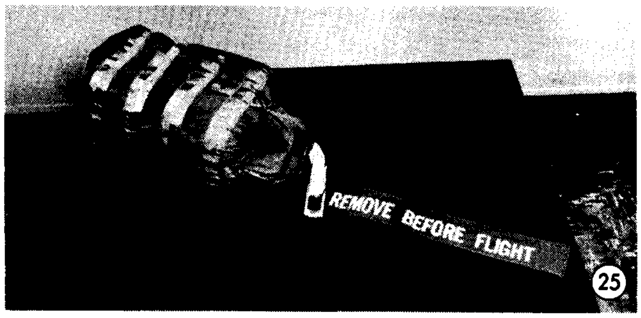
NOTE

HAVE PILOT CHUTE SAFETY PIN ACCESSIBLE BEFORE PERFORMING STEP 24.



STEP 24

TIGHTLY COMPRESS PILOT CHUTE AND SPRING ON TOP OF DEPLOYMENT BAG.
CLOSE FLAP WITH CONICAL ATTACHMENT OVER PILOT CHUTE.



STEP 25

CLOSE SIDE FLAPS AND OPPOSITE FLAP, SLIPPING THE METAL GROMMET OF EACH OVER THE CONICAL ATTACHMENT. INSERT PILOT CHUTE SAFETY PIN THRU CONICAL ATTACHMENT.

NOTE

CHECK PILOT CHUTE COMPARTMENT FOR PROTRUDING FABRIC OR WEBBING.
CHECK RISER LOOP FOR SECURITY.

JS-2-2-3906-5

Figure 39-6. Drag Chute Packing (Sheet 5 of 5)

CHAPTER 51
STRUCTURES

TABLE OF CONTENTS

Section Page

51.0 GENERAL . . . 51-3

51.1 ALIGNMENT AND SYMMETRY CHECK . . . 51-7

51.2 AIRFRAME SEALING . . . 51-13

51.3 PENETRANT INSPECTION . . . 51-19

51.4 PRESSURIZATION CHECK . . . 51-23

51.5 MASS REBALANCE OF CONTROL SURFACES . . . 51-25

51.6 CHEMICAL IDENTIFICATION OF MATERIALS . . . 51-35

51.7 CORROSION CONTROL . . . 51-39

51.8 FINISHES . . . 51-47

51.9 DEPLETED URANIUM COUNTERWEIGHTS . . . 51-57

51.10 AIRFRAME LONG TERM STORAGE . . . 51-59

SECTION 51.0
GENERAL

Contents

GENERAL 51-3

FIGURES
51-1. Standard Torque Values 51-4
51-2. Standard Torque for High Strength Applications 51-5
51-3. Wing to Fuselage Attachment Bolt Torque Values 51-6

51.0.1 GENERAL. The major structural assemblies, fuselage, wing, empennage, doors, nacelles, and windows are described in their respective chapters. This chapter covers items of a more general nature which apply to complete structure.

51.0.2 TORQ-SET RECESS SCREW SUBSTITUTION. TORQ-SET recess screws are used throughout the airplane in large quantities. When the proper TORQ-SET screws are not readily available the following short thread Phillips recess screws may be substituted:

PAN HEAD SCREWS

TORQ-SET SCREW	SCREW THREAD	PHILLIPS RECESS SCREW
NAS 1632	8-32	NAS 623-2
NAS 1633	10-32	NAS 623-3
NAS 1634	1/4-28	NAS 623-4

FLAT HEAD SCREWS

TORQ-SET SCREW	SCREW THREAD	PHILLIPS RECESS SCREW
NAS 1622	8-32	NAS 1202
NAS 1623	10-32	NAS 1203
NAS 1624	1/4-28	NAS 1204
NAS 1625	5/16-24	NAS 1205

STANDARD TORQUE VALUES. When no torque values are specified in the applicable section of the manual, use the torque values given in table 51-1 for bolts and screws with tensile strength of or over 125,000 PSI and with a nominal diameter of or over 1/4 inch except as noted. It is recommended that an attempt be made to torque all structural fasteners to the mid-range of the following tables when torquing from the nut side. When torquing from the head side use the upper limit +0, —6 percent. Replace any nuts, bolts, or screws that are tightened above the applicable torque value. For joints containing wood, plastics, rubber, or rubber-like material, use a torque value approximately 80 percent of the torque at which deformation of the materials is observed, or use the value given in the table, whichever is lower. Brass, aluminum alloy, low-carbon steel, and low-strength (under 125,000 PSI) corrosion-resistant steel bolts and screws should be tightened to produce a tight joint, but this must not exceed the applicable lower torque value of the table.

Screws and bolts should penetrate self locking nuts at least 1/32 inch. In case of round or chamfered screw ends, the full round or chamfer shall extend through the nut. Nuts shall not be bottomed out on the grip portion of a fastener.

On multiple bolt or screw installations, first finger-tighten the attachments all around. Then snug up opposite attachments all around. Then torque opposite attachments all around. Do not successively tighten adjacent attachments.

Installation torque values are applicable only if the joint has a slip fit bushing. (See figure 51-2A.) Where alignment between bolt or screw cotter pin hole and nut slot is not reached within the torque range indicated, the nut shall be replaced. Clevis type joints not having slip fit bushings shall not be tightened to the torque values indicated, but shall be tightened only to the extent necessary to eliminate longitudinal movement of the bolt or screw assembly. The fastener assembly shall be capable of being rotated by hand. If the above conditions are met and the bolt or screw cotter pin hole and nut slot do not line up, the nut shall be backed off until line-up first occurs.

INSTALLATION TORQUE, INCH-POUNDS				
	STANDARD OR TENSION NUTS △8		SHEAR NUTS	
Bolt or Screw Size	Plain, Self-Locking and Castellated Nuts. (See Note 3.)	Alternate Values for AN310 Castle Nuts. (See Note 5.)	Plain Self-Locking & Castellated Nuts. (See Note 4.)	Alternate Values For AN320 Castle Nuts. (See Note 5.)
10-32	25-40	20-40	12-15	12-19
1/4-28	70-100	50-100	30-40	30-48
5/16-24	140-225	100-225	60-85	60-106
3/8-24	190-390	160-390	95-110	95-170
7/16-20	450-500	450-660	270-300	270-390
1/2-20	480-690	480-840	290-410	290-500
9/16-18	800-1000	800-1250	480-600	480-750
5/8-18	1000-1300	1100-1800	660-780	660-1060
3/4-16	2300-2500	2300-3700	1300-1500	1300-2200
7/8-14	2500-3000	2500-4900	1500-1800	1500-2900
1-14 & 1-12	3700-5500	3700-7300	2200-3300	2200-4400
1-1/8-12	5000-7000	5000-10,000	3000-4200	3000-6300
1-1/4-12	9000-11,000	9000-17,500	5400-6600	5400-10,000

NOTE

1. The following notes give examples of nuts, bolts, and screws for which this table may be used. These are not to be considered exclusive, for they simply show examples of the types referred to.
2. Hex Head Bolts: AN3 thru AN20, AN173 thru AN186, NAS464, NAS1103 thru NAS1114, NAS1003 thru NAS1014.
   Clevis Bolts: AN21 thru AN36.
   Eyebolts: AN42 thru AN49.
   100° Head Bolts: NAS333 thru NAS340.
   Fillister Head Screws: AN502
   Round Head Screws: AN515 and AN520.
   Pan Head Screws: NAS623, NAS1630 thru NAS1634.
   Brazier Head Screws: NAS220 thru NAS227.
   100° Head Screws: NAS517, NAS1202 thru NAS1207, NAS1620 thru NAS1628, and NAS560.
3. Castle Nuts: AN310.
   Plain Hex Nuts: AN315, AN340, and AN345.
   Hex Nuts: AN363, AN365, NAS679, NAS1291, MS20500, and NAS1021.
   Anchor Nuts: AN362, NAS680 thru NAS687, NAS696, NAS697, NAS1067, and NAS1068.
   12-Point Nuts: EB.
   Barrel Nuts: 1452, 2452, 2552, 2752, and SL40.
4. Shear Nuts: AN364, AN320, AN316, NAS1022, and MS21042.
5. Use alternate values, or replace nut in those cases using AN310 and AN320 castellated nuts where alignment between bolt or screw cotter pin hole and nut slot is not reached using the normal torque values.
6. Torque values are in inch-pounds.
7. Some bolts and nuts are lubricated as received. These values assume that no added lubricant is used.

△8 Use figure 51-2 for tension applications of high strength internal wrenching bolts (MS20004 through MS20012) or 12 point bolts (NAS624 through NAS632 and MS21250).

Figure 51-1. Standard Torque Values

HIGH STRENGTH BOLT TENSION APPLICATIONS (USING LUBRICATED NUTS) INSTALLATION TORQUE, INCH-POUNDS				
THREAD SIZE	INTERNAL WRENCHING BOLTS △1		12 POINT BOLTS △2	
	NO SEALANT	WITH SEALANT △3	NO SEALANT	WITH SEALANT △3
1/4-28	75-85	120-140	95-105	140-160
5/16-24	140-160	210-240	160-180	235-265
3/8-24	250-280	340-380	280-320	375-425
7/16-20	425-475	565-635	490-560	610-690
1/2-20	635-715	800-900	750-850	915-1035
9/16-18	890-1000	1100-1240	1035-1165	1270-1430
5/8-18	1200-1350	1500-1700	1355-1525	1700-1900
3/4-16	1950-2200	2570-2900	2160-2440	2865-3235.

△1 MS20004 through MS20012.

△2 NAS624 through NAS632 and MS21250.

△3 MIL-S-8802 or MIL-S-81733.

Figure 51-2. Standard Torque for High Strength Applications

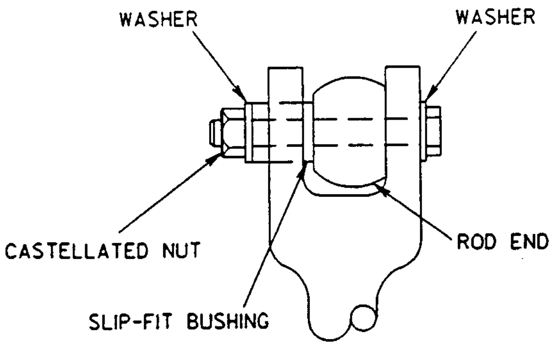
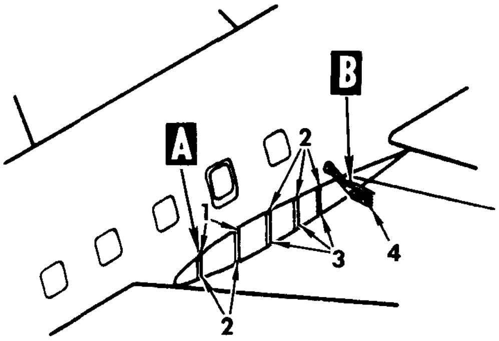
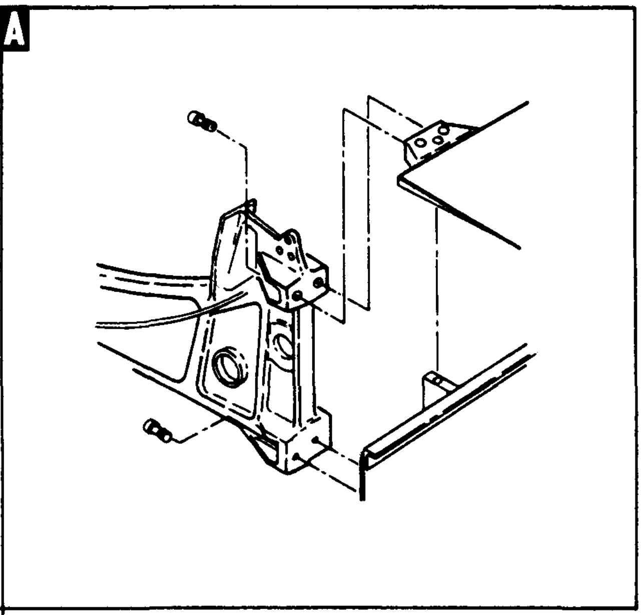


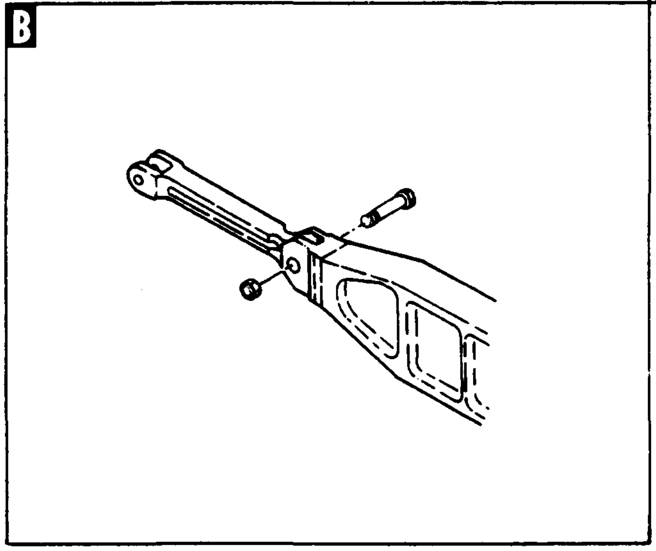
Figure 51-2A. Clevis with Slip Bushing





CAUTION

DO NOT REMOVE OR LOOSEN MORE THAN ONE BOLT ON EACH SIDE OF THE AIRCRAFT WITHOUT SUPPORTING THE WING IN A NEUTRAL POSITION. REFER TO SECTION 7.1.



TORQUE VALUES	
JOINTS	TORQUE (INCH POUNDS)
1 (1/2 IN. BOLTS)	480-691
2 (5/8 IN. BOLTS)	1000-1300
3 (3/4 IN. BOLTS)	2300-2500
4 (1/2 IN. SHEAR NUT)	290-410

NOTE

1. DETAIL "A" IS TYPICAL FOR FRAMES CONTAINING JOINTS, 1, 2 AND 3.
2. THOROUGHLY LUBRICATE THE NUTS, BOLTS AND WASHERS IN JOINTS 1, 2 AND 3 WITH LUBRIPLATE OR MIL-G-23827 BEFORE APPLYING THE TORQUE. DO NOT USE ANY LUBRICANT IN JOINT 4.

JS-51-0-003

Figure 51-3. Wing-To-Fuselage Attachment Bolt Torque Values

SECTION 51.1
ALIGNMENT AND SYMMETRY CHECK

Contents

ALIGNMENT CHECK 51-7

SYMMETRY CHECK 51-7

FIGURES
51-4. Alignment and Symmetry 51-10
51-5. Alignment Data Sheet 51-12

51.1.1 ALIGNMENT CHECK. Perform an alignment check if it is suspected that the airplane is distorted, and after major structural repairs have been made. This check consists of measuring the distances between reference points projected from the leveled airplane to the floor plane. These distances are checked against established dimensions. (See figure 51-4 for the locations of the reference points used in this test.) Perform this check on a defueled airplane, in an enclosed level area, in still air, and at a constant temperature.

a. Level the airplane as outlined in Chapter 8. Use the external leveling provisions.

b. Prepare an "Airplane Alignment" data sheet and a "Nacelle Alignment" data sheet similar to the tables shown in figure 51-5.

c. Project the following points to the floor with a plumb bob, and identify each point: 1, 2, 3L, 3R, 5L, 5R, 9L, 9R, 10L, 10R, 11L, 11R, 8L, and 8R.

d. Snap a chalk line through the projected points 1 and 2, 3L and 3R, 9L and 9R, 10L and 10R, 11L and 11R, and 8L and 8R.

e. Identify the intersections of the snapped lines as follows: 1-2 and 3L-3R as point A, 1-2 and 10L-10R as point B, 1-2 and 9L-9R as point C, 1-2 and 11L-11R as point D, and 1-2 and 8L-8R as point E.

f. With a steel tape or other accurate means, measure the distances (1-3L, 1-3R, 1-9L, 1-9R, 2-3L, 2-3R, A-5L, and A-5R), and record the values in the appropriate blocks of the data sheet. Subtract the smaller value from the larger (right from left or left from right), and record the differences in the proper blocks on the data sheet.
RESULT: The differences should be within the allowable limits indicated on the data sheet.

g. Measure the distances C-10L, C-10R, B-9L, and B-9R.
RESULT: These distances should all be equal to each other within ±0.13 inch.

h. Measure the distances D-11L, D-11R, E-8L, and E-8R. Record the distances in the proper spaces of the nacelle alignment data sheet. Subtract the values for 8L and 8R from 11L and 11R, and record the differences.
RESULT: The differences should be within the limits given in figure 51-4.

i. Fair the rudder with the vertical stabilizer, suspend a plumb bob from the upper aft trailing edge of the rudder, and allow the bob to swing free just above the fairing for the fuel jettison outlet. (See figure 51-4.)

j. Adjust the trim of the empennage section until the bob is aligned on the transverse centerline of point 7.
RESULT: The plumb bob should not vary more than 0.25 inch to the left or right of the longitudinal centerline.

51.1.2 SYMMETRY CHECK. Perform a symmetry check if it is suspected the airplane is distorted and after major structural repairs have been made on the airplane. This check consists of measuring the elevation of reference points on the leveled airplane relative to a horizontal reference plane and checking these measurements against established dimensions. (See figure 51-4.) Perform this check on a defueled airplane, in an enclosed level area, in still air, and at a constant temperature.

a. Level the airplane using the external leveling provisions. (See instructions in Chapter 8.)

b. Use a transit to make the elevation measurements.

51.1.2 (Cont).

Note

> Position the transit opposite the entrance door, approximately in line with the external fuel tank, and use a stand short enough to permit an unobstructed view beneath the airplane. This will permit all of the required shots to be taken without moving the transit.

c. Level the transit.

d. Prepare a "Dihedral and Fuselage Symmetry Check Data" sheet similar to the table shown in figure 51-5.

e. Make readings at points 1 through 6, left and right, and 9 and 10, left and right. Subtract the reading taken at point 2 from each of the other readings, and record the difference in the proper block in line (a) on the data sheet.

Note

> To make elevation readings, hold a metal scale (or equivalent) vertically with the zero end against the point at which the reading is being taken. Take the readings through the leveled transit.

f. In line (a) of each of the columns for points 3 through 6, 9 and 10, subtract the smaller reading (right from left or left from right) and record the differences.
RESULT: The differences should be within the limits given.

g. Prepare an "Incidence Checks" sheet similar to the table shown in figure 51-5.

h. Insert in the proper spaces the height of the different points above point 2 as obtained in step (e).

i. Subtract the values, both left and right, as follows: 10 from 9, 3 from 4, and 5 from 6. Record the differences in the proper column in the difference line of the sheet.
RESULT: The differences should be within the limits given.

j. Prepare a "Nacelle Incidence Check" data sheet similar to the table in figure 51-5.

k. Make readings of the elevations at points 11 and 8, left and right. Subtract the reading previously taken at point 2 from each of these readings, and record the differences in the proper spaces of the data sheet as the height above point 2.

l. Subtract the values for point 8, left and right, from those for point 11, left and right, and record the differences in the proper spaces on the data sheet.
RESULT: The differences should be within the limits given.

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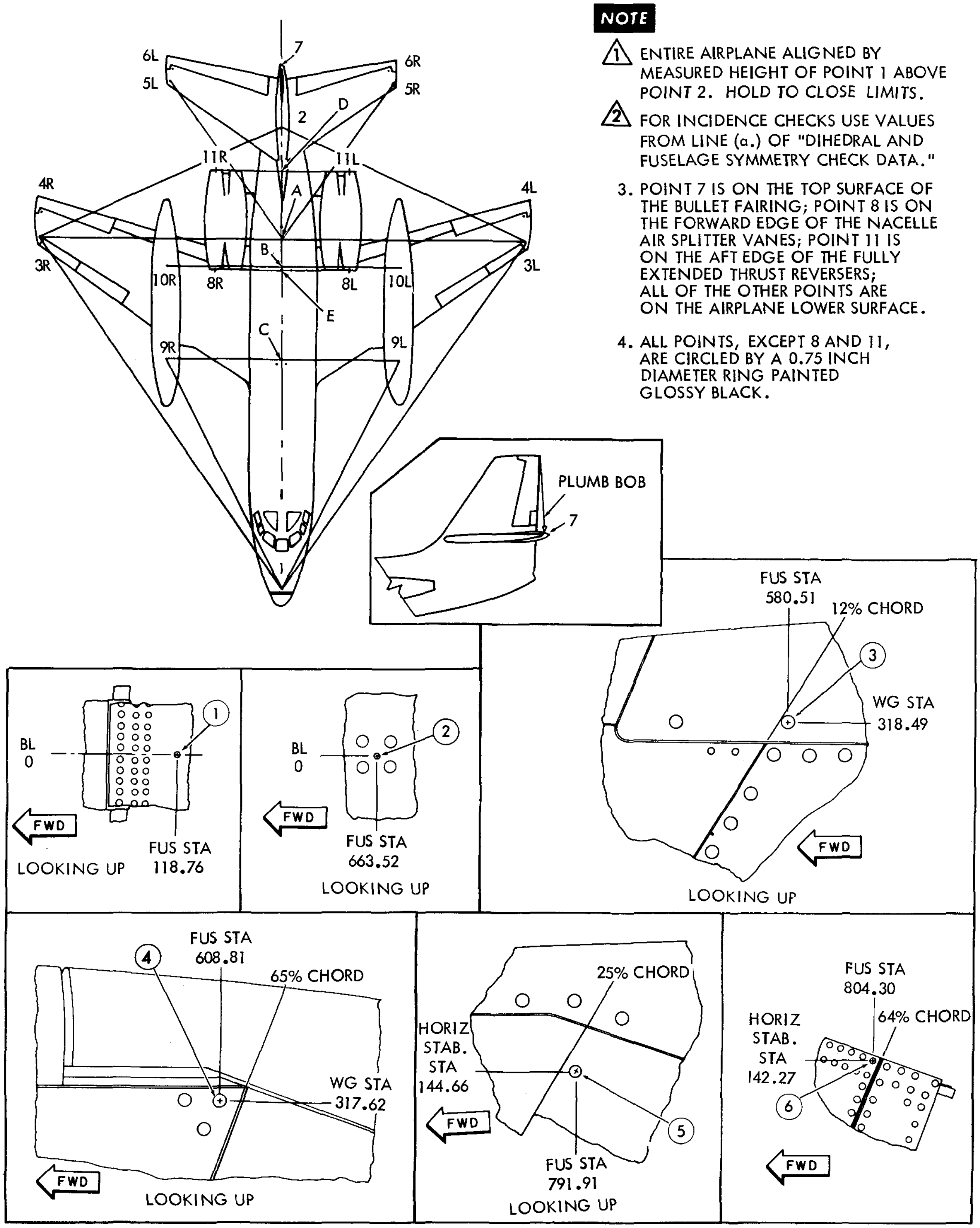


NOTE

△1 ENTIRE AIRPLANE ALIGNED BY MEASURED HEIGHT OF POINT 1 ABOVE POINT 2. HOLD TO CLOSE LIMITS.

△2 FOR INCIDENCE CHECKS USE VALUES FROM LINE (a.) OF "DIHEDRAL AND FUSELAGE SYMMETRY CHECK DATA."

3. POINT 7 IS ON THE TOP SURFACE OF THE BULLET FAIRING; POINT 8 IS ON THE FORWARD EDGE OF THE NACELLE AIR SPLITTER VANES; POINT 11 IS ON THE AFT EDGE OF THE FULLY EXTENDED THRUST REVERSERS; ALL OF THE OTHER POINTS ARE ON THE AIRPLANE LOWER SURFACE.
4. ALL POINTS, EXCEPT 8 AND 11, ARE CIRCLED BY A 0.75 INCH DIAMETER RING PAINTED GLOSSY BLACK.

Figure 51-4. Alignment and Symmetry (Sheet 1 of 2)

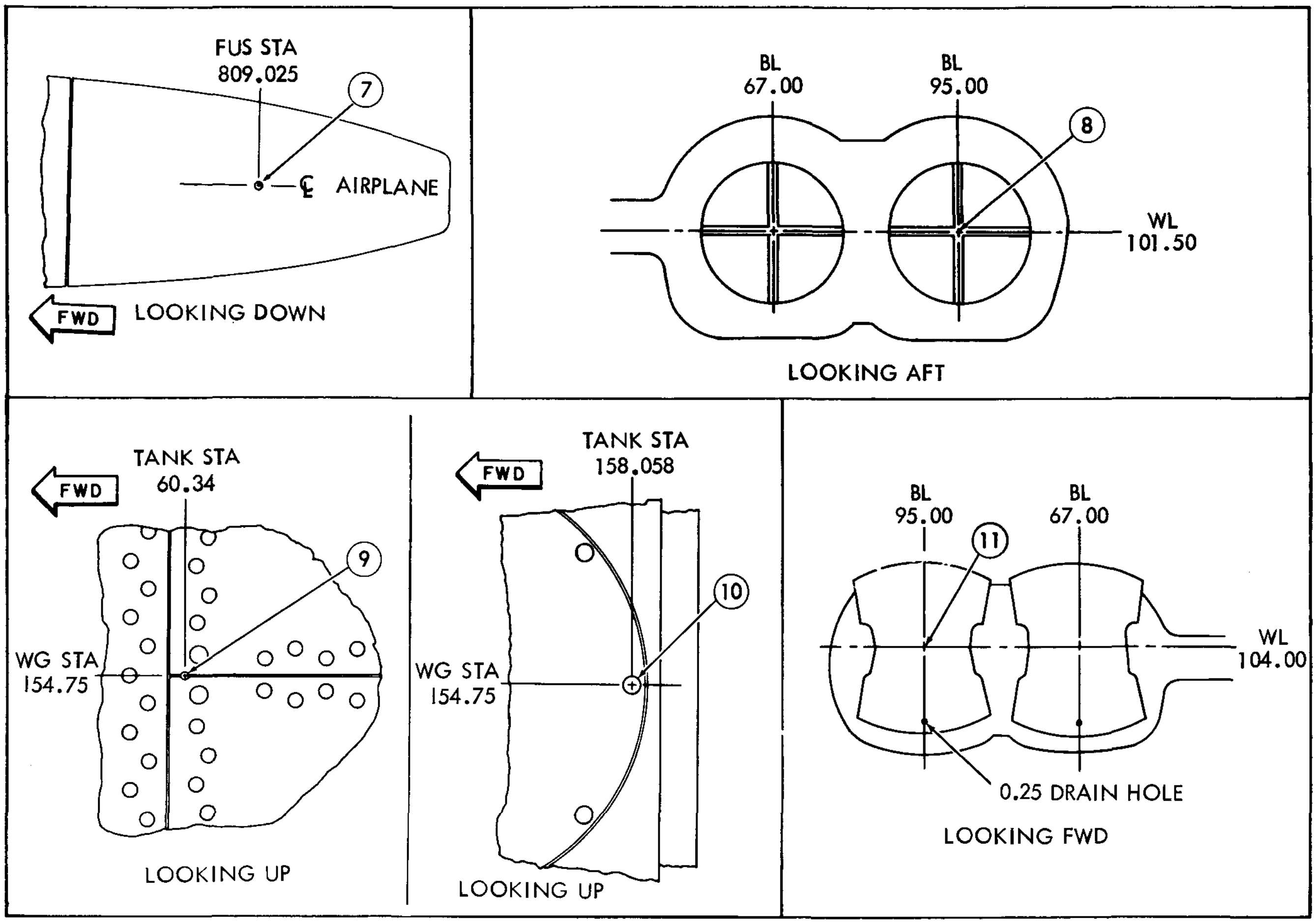


Figure 51-4. Alignment and Symmetry (Sheet 2 of 2)

DIHEDRAL AND FUSELAGE SYMMETRY CHECK DATA

	POINT 1	POINT 3		POINT 4		POINT 5		POINT 6		POINT 9		POINT 10	
		LEFT	RIGHT	LEFT	RIGHT	LEFT	RIGHT	LEFT	RIGHT	LEFT	RIGHT	LEFT	RIGHT
a. MEASURED HEIGHT ABOVE POINT 2	3.70 [1]									(-)	(-)	(-)	(-)
b. CALCULATED HGT ABOVE POINT 2	3.70	9.51	9.51	10.61	10.61	97.83	97.83	98.52	98.52	-13.42	-13.42	-13.42	-13.42
c. DIFFERENCE BETWEEN LEFT AND RIGHT													
d. ALLOWABLE DIFFERENCE LEFT AND RIGHT		0.10		0.10		0.60		0.60		0.20		0.30	

NACELLE INCIDENCE CHECK

	LEFT	RIGHT
POINT 11 ABOVE POINT 2	37.73	37.73
POINT 8 ABOVE POINT 2	35.23	35.23
CALCULATED DIFFERENCE	2.50	2.50
ALLOWABLE DIFFERENCE	2.00 TO 3.00	2.00 TO 3.00

NACELLE ALIGNMENT

DISTANCE	LEFT	RIGHT
D-11	95.00	95.00
E-8	95.00	95.00
CALCULATED DIFFERENCE	0.00	0.00
ALLOWABLE DIFFERENCE	+0.25 TO -0.25	+0.25 TO -0.25

AIRPLANE ALIGNMENT

DISTANCE	MEASURED DISTANCE	CALCULATED DISTANCE
1-3L		560.86
1-3R		560.86
DIFFERENCE		
ALLOWABLE DIFFERENCE	0.20	
2-3L		329.13
2-3R		329.13
DIFFERENCE		
ALLOWABLE DIFFERENCE	0.25	
A-5L		256.16
A-5R		256.16
DIFFERENCE		
ALLOWABLE DIFFERENCE	0.60	
1-9L		358.31
1-9R		358.31
DIFFERENCE		
ALLOWABLE DIFFERENCE	0.10	

INCIDENCE CHECKS [2]

EXTERNAL WING TANKS	LEFT	RIGHT	WING TIP	LEFT	RIGHT	HORIZ STAB. TIP	LEFT	RIGHT
POINT 9			POINT 4			POINT 6		
POINT 10			POINT 3			POINT 5		
DIFFERENCE			DIFFERENCE			DIFFERENCE		
CALCULATED DIFFERENCE	0.00	0.00	CALCULATED DIFFERENCE	1.10	1.10	CALCULATED DIFFERENCE	0.69	0.69
ALLOWABLE DIFFERENCE	0.10	0.10	ALLOWABLE DIFFERENCE	1.00 TO 1.20	1.00 TO 1.20	ALLOWABLE DIFFERENCE	0.59 TO 0.79	0.59 TO 0.79

JS-2-2-5103

Figure 51-5. Alignment Data Sheet

SECTION 51.2
AIRFRAME SEALING

Contents Page

GENERAL 51-13
  Pressure Seal 51-13
  Weather Seal 51-14
  Acid Seal 51-14
  Aerodynamic Seal 51-14
  Vapor Seal 51-14
  Firewall Seal 51-14

REPAIRS IN SEALED AREAS 51-14

SEALANT APPLICATION METHODS 51-14
  Filleting 51-14
  Faying 51-14
  Fastener Sealing 51-14
  Hole and Slot Filling 51-14

SEALANTS 51-14

APPLICATION LIFE OF SEALANTS 51-15

TACK FREE TIME 51-15

VARIATIONS IN APPLICATION LIFE AND TACK FREE TIME 51-15

CALCULATION OF TACK FREE TIME CONSIDERING HUMIDITY AND TEMPERATURE VARIATIONS 51-15

STORAGE LIMITATIONS FOR SEALANTS 51-16

FIGURES
  51-6 Sealing Compounds 51-16
  51-7 Sealant Tack Free Time 51-17
  51-8 Sealing Hand Tools 51-18

51.2.1 GENERAL. Several different types of seals are used in the JetStar airplane to meet the requirements of different areas. The type of seals and areas in which they are used are as follows:

51.2.1.1 PRESSURE SEAL. A pressure seal is a complete seal of all joints, seams, holes, and fasteners as necessary to provide a pressure-tight vessel. The fuselage is pressure-sealed between fuselage stations 97 and 570. The following materials (or equivalents) are used in the seal:

a. HOLE FILLING COMPOUND. Use sealant F, figure 51-6. Sealants C or D may be used as an alternate for sealant F to fill holes by applying successive layers of sealant as required to close holes.

b. FAY SURFACE AND WET FASTENER SEALANTS. Use sealant E, figure 51-6.

c. FILLET AND INJECTION SEALANTS (Not for use in fuel tanks.) Use sealant C or D, figure 51-6. Sealant H may be used in areas where resistance to fuel is not required.

d. SPRAYABLE SEALANT. Use sealant G, figure 51-6.

e. BRUSH SEALANTS. Use sealant A or B, figure 51-6. Sealant I may be used in areas where resistance to fuel is not required.

51.2.1.2 WEATHER SEAL. A weather seal is a seal only to the extent necessary to exclude water and debris from the sealed area. The fuselage is weather-sealed between fuselage stations 570 and 800, from the left 30° longeron over the top to the right 30° longeron. The same materials (or equivalents) that are used for pressure sealing are used for weather sealing.

51.2.1.3 ACID SEAL. An acid seal is used on the lower inside surface of the fuselage. It is a complete seal of all joints, seams, holes, and fasteners as necessary to confine spillage of corrosive fluids. It is applied as shown in note 1 of figure 51-22. Use sealant H, figure 51-6.

51.2.1.4 AERODYNAMIC SEAL. Aerodynamic sealing is the application of aerodynamic smoothers to exterior surfaces of the airplane to provide aerodynamic flushness and to fair joggles and other irregularities of the exterior skin surfaces. Use sealant K, figure 51-6.

51.2.1.5 VAPOR SEAL. A vapor seal is a seal of all splice joints, holes, access doors, and cover as necessary to contain fuel vapors. Fasteners (such as rivets, bolts, and screws) are not sealed. The fuel valve compartment in the aft equipment area is vapor-sealed as shown in figure 51-22, note 3. Fuel resistant sealants A, B, C, D, E, F, and G (figure 51-6) are all suitable for vapor sealing.

51.2.1.6 FIREWALL SEAL. All seams and holes in the nacelle firewalls are sealed with Pro-Seal #700 (Essex Chemical Corp.) firewall sealant (or equivalent) in accordance with MIL-S-38249 Type I. Fasteners (such as rivets, bolts, and screws) are not sealed. Use sealant J, figure 51-6.

51.2.2 REPAIRS IN SEALED AREAS. When making structural repairs in sealed areas, all damaged sealant must be replaced, and the repair must be adequately sealed. Use the following procedure when making repairs in any of the sealed areas.

Note

Coordinate these instructions with the applicable repair instructions elsewhere in this manual.

a. Remove all damaged sealant and all other sealant as necessary for positioning the repair members. Use the hand tools shown in figure 51-8 for removing the sealant.

b. Taper the edges of the remaining sealant.

c. Thoroughly clean the area of all dirt, grease, fingerprints, metal chips, bits of sealant, and any other foreign matter. Use clean cloths dampened with Specification TT-N-95A, Type II aliphatic naphtha, or O-T-620 trichloroethane. Wipe dry with clean cloth before solvent is allowed to evaporate.

d. After the repair members have been fabricated and given the proper anticorrosion treatment, apply a faying surface seal between the mating surfaces of the repair and the existing structure, and install the repair members during minimum application life of sealant. Use inhibited fay sealant (E).

e. Finish sealing the repair with fillet seals, fastener sealing, brush sealants, or spray sealants as applicable, depending on the location of the repair.

f. Apply interior finish as applicable.

51.2.3 SEALANT APPLICATION METHODS. Various methods of applying sealant are needed to seal structural repairs properly inside the sealed fuselage areas. They are as follows:

51.2.3.1 FILLETING. Apply a bead of filleting compound (C) or (D) with a filleting spatula or gun at the parting line between two attached parts on the sealed side of the structure (see figure 51-6). Apply filleting sealant with a gun when available. Work the sealant carefully into place with a spatula to force out any trapped air. Fair the sealant in smoothly with the original material and structure. Within one hour after application of fillets, smooth and dress all fillets with a spatula or a brush dipped in sealant (A) or (B).

51.2.3.2 FAYING. Faying surface sealing consists of applying sealant between mating surfaces of attached parts. To accomplish this, coat the mating surfaces of the repair members with sealant before assembly. Apply enough sealant to form a bead of excess sealant along the joint after assembly. Assemble the repair members before the sealant sets. Use sealant (E), as given in figure 51-6.

51.2.3.3 FASTENER SEALING. Seal the fasteners after assembly on the sealed side of the structure with two coats of brush sealant (A) or (B) (see figure 51-6). Allow first coat to become tack free before applying second coat. Apply brush sealant with a stiff bristle brush. To insure good adhesion, rub the material on the structure with a rotary scrubbing motion.

51.2.3.4 HOLE AND SLOT FILLING. Pack sealant into those holes and slots in the structure that have a width not exceeding 0.5 inch. Use sealant (F) to fill holes and slots (see figure 51-6). Seal holes and slots in the structure smaller than 0.05 inch with a pressure-sealing gun and sealants (C) or (E) (see figure 51-6).

51.2.4 SEALANTS. Sealants are supplied by the manufacturer in two parts: a base material and an accelerator. The label of each container of sealant has a correct mix ration, which is usually parts by weight of accelerator to be added to parts by weight of sealant. Follow the mix ratios on the labels, since these ratios may vary with products from different manufacturers and with different batches of the same product. Kits in the ½-pint size should be sufficient for sealing most repairs. The ½-pint kits are supplied with correct proportions of base and accelerator, which eliminates

51.2.4 (Cont).

the problem of weighing. Mix two-part sealants according to the following instructions:

CAUTION

Be careful to mix two-part sealants correctly. Incorrect mixing will result in improper set and poor flexibility.

a. Do not attempt to mix two-part sealants below 18.3°C (65°F) or above 37.8°C (100°F). Heating or cooling can be accomplished before mixing by placing the two containers in hot or cold water as necessary. Do not allow water to contact the sealant.

b. Mix the accelerator thoroughly to a smooth paste before adding to the base. Discard lumpy or caked accelerator.

c. Mix only in the proportions specified on the label, or if ½-pint kits are used, mix all of the accelerator with all of the base, regardless of how little sealant is required. With ½-pint kits it is important that all of the accelerator be scraped into the can of base compound, and that the base compound remaining on the lid also be scraped into the can.

d. Mix ½-pint kits in the base can. Mix weighed quantities from larger size kits in a clean container.

e. Stir the mixture by hand. Use a flat paddle or spatula similar to the one shown in figure 51-8. Stir slowly until a uniform color, free from marbled appearance, results. Stir for at least 4 minutes. Stirring the mixture rapidly will introduce air bubbles which will later burst and cause pinhole leaks. At least once during stirring, remove the paddle and scrape the sealant from the paddle into the container.

f. Do not try to economize with ½-pint kits by mixing only a part of the accelerator with only a part of the base compound. This practice will result in poorly sealed repairs.

g. Do not mix together different brands of sealing compound.

h. Do not mix together different types or different batches of the same brand of sealing compound.

CAUTION

Do not attempt to thin any two-part sealing compounds.

51.2.5 APPLICATION LIFE OF SEALANTS. This time after mixing during which a two-part sealing compound can be applied effectively is called the application life. (See figure 51-6.) When the application life has expired, the compound will be too thick for satisfactory application. Use the following rules to determine the usability of mixed sealant.

CAUTION

Discard still or partially set sealant, since it will not adhere properly when applied.

a. Class A brush material is usable until brush marks no longer flow out when the material is applied.

b. Class B fillet and injection materials are usable until the material sticks to the application gun or spatula instead of to the structure.

51.2.6 TACK FREE TIME. (See figure 51-7.) Tack free time is the time required after mixing for a sealing compound to lose stickiness and become firm and rubbery. The tack free time can be checked by touching a knuckle to the applied sealant. When the applied sealant is not sticky to the touch, the sealant is tack free.

51.2.7 VARIATIONS IN APPLICATION LIFE AND TACK FREE TIME. The following conditions will cause variations in application life and tack free time of sealants:

a. Tack free time decreases with increse in humidity, and increases with decrease in humidity.

b. Tack free time decreases with increase in temperature, and increases with decrease in temperature.

Note

Cure of sealants is difficult to obtain below 20 percent relative humidity or below 18.3°C (65°F).

51.2.8 CALCULATION OF TACK FREE TIME CONSIDERING HUMIDITY AND TEMPERATURE VARIATIONS. (See figure 51-7.) Calculate the approximate tack free time according to the following instructions.

a. Determine the relative humidity of the air. Determine the temperature of the structure to be sealed.

b. Determine the class of sealant to be used. Select the applicable graph on figure 51-7.

c. On the proper graph, select a line for the amount of relative humidity of the air.

d. Mark the temperature read on the temperature scale. Draw a straight line until it crosses the relative humidity line that was drawn or selected in step c.

e. At the point where the temperature line crosses the relative humidity line, draw a vertical line down to the tack free time scale. Read the tack free time where the vertical line crosses the tack free time scale.

51.2.8 (Cont).

f. The tack free time read in e will be approximately the time required for the sealing compound to cure after it has been applied to the structure.

51.2.9 STORAGE LIMITATIONS FOR SEALANTS. The storage limits or shelf life of unmixed sealing compounds should be in accordance with the manufacturer's recommendations. If these recommendations are not available, limits of one year at temperatures below 92°F may be established.

Sealant	Material Description	Minimum Application Life at 77°F and 50 PCT Relative Humidity	Maximum Tack Free Time at 77°F and 50 PCT Relative Humidity	Manufacturer's Identification	Government Specification
A	Quick Repair Brush Sealant: used in service for all minor repairs of rivets and fasteners.	30 minutes	10 hours	[2] CS-3204,A-1/2 [3] Pro-Seal 890,A-1/2 [4] GC-408, A-1/2 [5] PR-1422, A-1/2	MIL-S-8802 Class A-1/2
B	Standard Brush Sealant: used for sealing rivets, bolts, and all fasteners.	2 hours	40 hours	[2] CS-3204, A-2 [3] Pro-Seal 890,A-2 [4] GC-408,A-2 [5] PR-1422,A-2	MIL-S-8802 Class A-2
C	Quick Repair Fillet & Injection Sealant: used in service repairs for filling gaps, voids, or for filleting along seams and joints.	30 minutes	10 hours	[2] CS-3204,B-1/2 [3] Pro-Seal 890,B-1/2 [5] PR-1422,B-1/2	MIL-S-8802 Class B-1/2
D	Standard Fillet & Injection Sealant; used for filling gaps, voids, and for filleting along seams and joints where longer work life is desirable.	2 hours	40 hours	[2] CS-3204,B-2 [3] Pro-Seal 890,B-2 [4] GC-408,B-2 [5] PR-1422, B-2	MIL-S-8802 Class B-2
E	Corrosion inhibited sealant for fay-sealing and wet installing fasteners.	30 minutes	16 hours	[3] Pro-Seal 870,B-1/2 [5] PR-1422G, B-1/2	MIL-S-81733 Type II, B-1/2
		2 hours	24 hours	[3] Pro-Seal 870, B-2 [5] PR-1422G, B-2	MIL-S-81733 Type II, B-2
		4 hours	32 hours	[3] Pro-Seal 870, B-4 [5] PR-1422G,B4	MIL-S-81733 Type II, B-4
		12 hours	144 hours	[3] Pro-Seal 870, B-12 [5] PR-1431G, Type I	MIL-S-81733 Type IV, B-12
		24 hours	240 hours	[3] Pro-Seal 870, B-24 [5] PR-1431G, Type II	MIL-S-81733 Type IV, B-24
		48 hours	504 hours	[3] Pro-Seal 870, B-48 [5] PR-1431G, Type IV	MIL-S-81733 Type IV, B-48

Figure 51-6. Sealing Compounds (Sheet 1 of 2)

Sealant	Material Description	Minimum Application Life at 77°F and 50 PCT Relative Humidity	Maximum Tack Free Time at 77°F and 50 PCT Relative Humidity	Manufacturer's Identification	Government Specification
F	Two-part Hole Filling compound: used for filling holes and open slots. See note 1.	2 hours	24 hours	△3 Pro-Seal 567, B-2 △4 GC-1406,B-2 △5 PR-205,B-2	
G	Sprayable sealant	2 hours	24 hours	△4 GC-411	MIL-C-38713
H	Pressure Sealant, fillet and injection sealing, low density. (Not to be used in fuel tanks.)	15 minutes	8 hours	△5 PR-1223, B-1/4	
		30 minutes	10 hours	△4 GC-401, B-1/2 △5 PR-1223,B-1/2	
		2 hours	24 hours	△4 GC-401,B-2 △5 PR-1223, B-2	
		4 hours	30 hours	△4 GC-401, B-4 △5 PR-1223, B-4	
I	Pressure Sealant, brush sealing, low density. (Not to be used in fuel tanks.)	30 minutes	10 hours	△4 GC-401, A-1/2	
		2 hours	24 hours	△4 GC-401, A-2	
		4 hours	36 hours	△4 GC-401, A-4	
J	Firewall Sealant	1.5 hours	24 hours	△3 Pro-Seal 700	MIL-S-38249, Type I
K	Aerodynamic Sealant, smoothing and fairing	30 minutes	10 hours	△3 Pro-Seal 895, B-1/2	
		1 hour	24 hours	△3 Pro-Seal 895, B-1	
		1 hour	12 hours	△6 93-006-1	
		2 hours	24 hours	△6 93-006-6	

Note

1. Alternate material-use sealant C or D. Build up to fill hole by successive applications.

△2 Chem Seal Corp., 11120 Sherman Way, Sun Valley, CA 91352.

△3 Pro-Seal products are available from Products Research Corp., See Note 5.

△4 Goal Chem Corp., 1337 East 26th Street, Los Angeles, CA 90023.

△5 Products Research Corp., 2919 Empire Avenue, Burbank, CA 91504.

△6 Dow-Corning, 4825 N. Scott St., Schiller Park, IL 60176.

Figure 51-6. Sealing Compounds (Sheet 2 of 2)

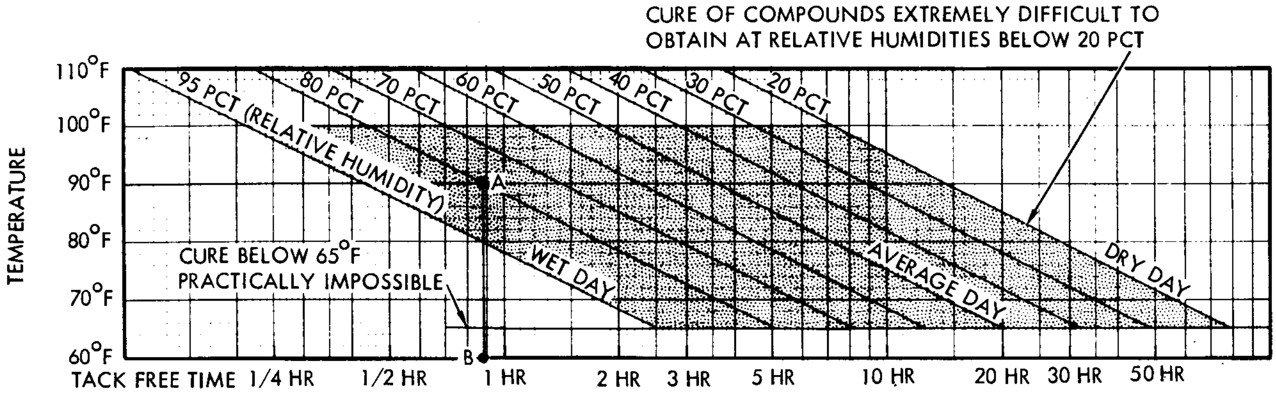


CHART A-SEALANT: CLASSES A-1/2 AND B-1/2 QUICK REPAIR SEALANTS

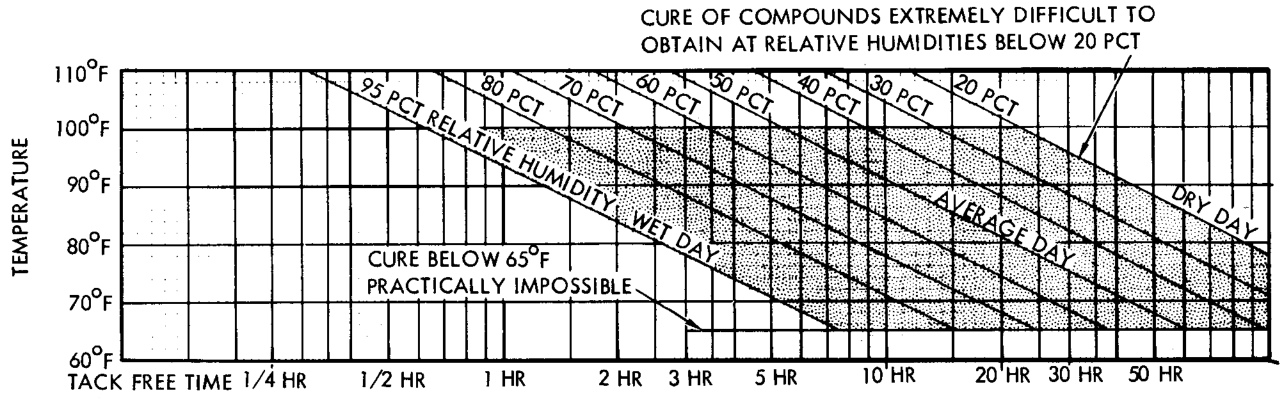


CHART B-SEALANT: CLASS B-4 SEALANTS

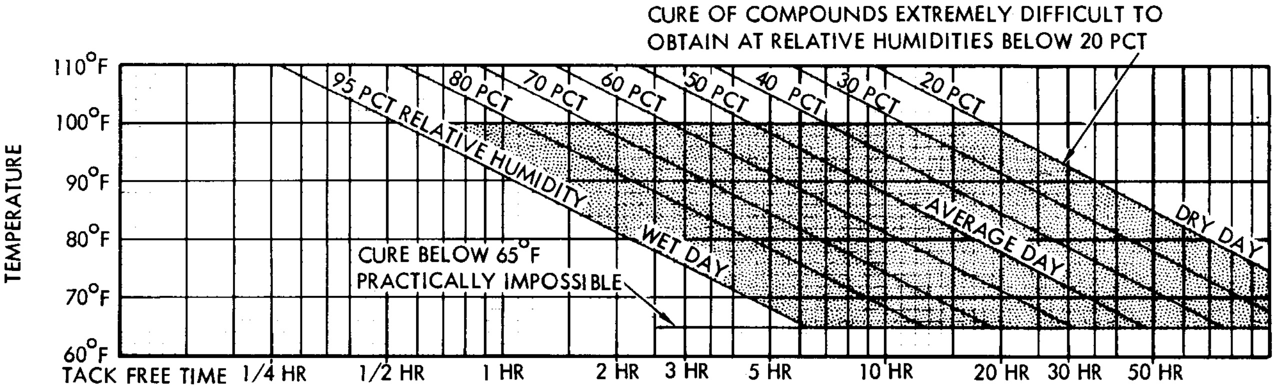


CHART C-SEALANT: CLASSES B-2 AND A-2 SEALANTS

EXAMPLE PROBLEM:

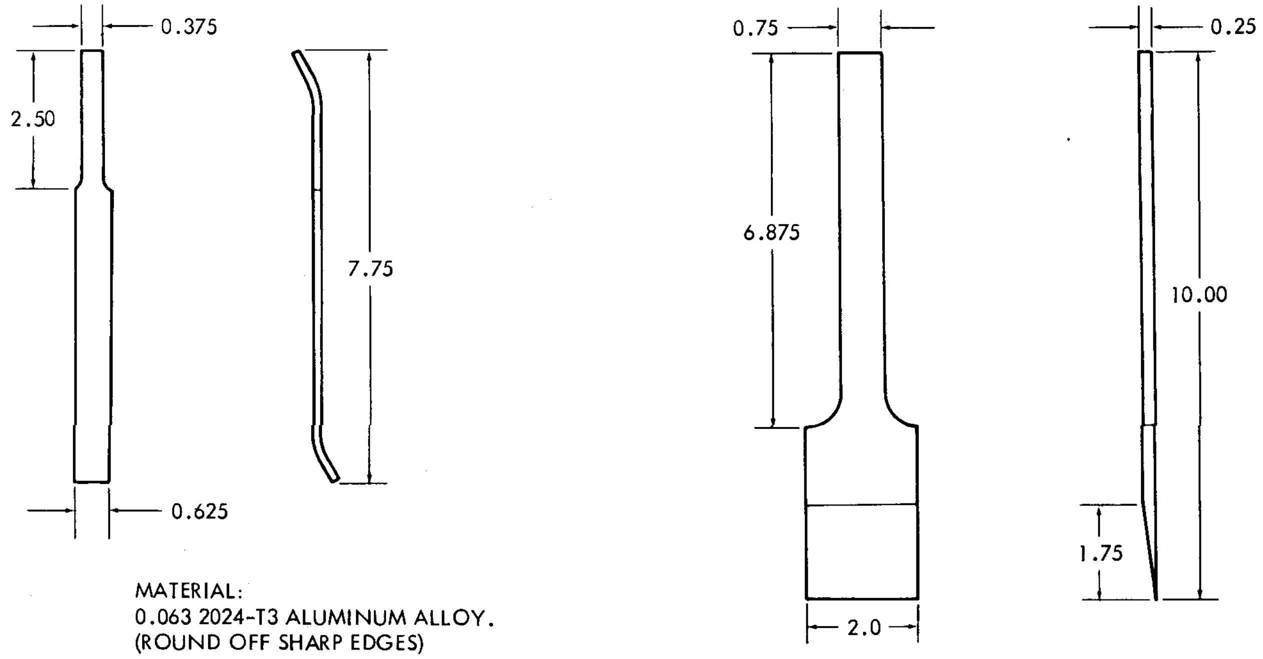
SEALANT IS CLASS A-1/2.
WHAT IS TACK FREE TIME AT TEMPERATURE OF 90°F AND RELATIVE HUMIDITY OF 80 PCT?

SOLUTION:

FIND THE INTERSECTING POINT OF THE 90°F TEMPERATURE LINE AND THE INCLINED 80 PCT RELATIVE HUMIDITY LINE. THIS IS POINT A ON CHART A. THEN FOLLOW VERTICALLY DOWN TO THE TACK FREE TIME LINE (POINT B) AND READ ANSWER: SLIGHTLY OVER 3/4 HOURS.

JS-1-2-X0/2-5107

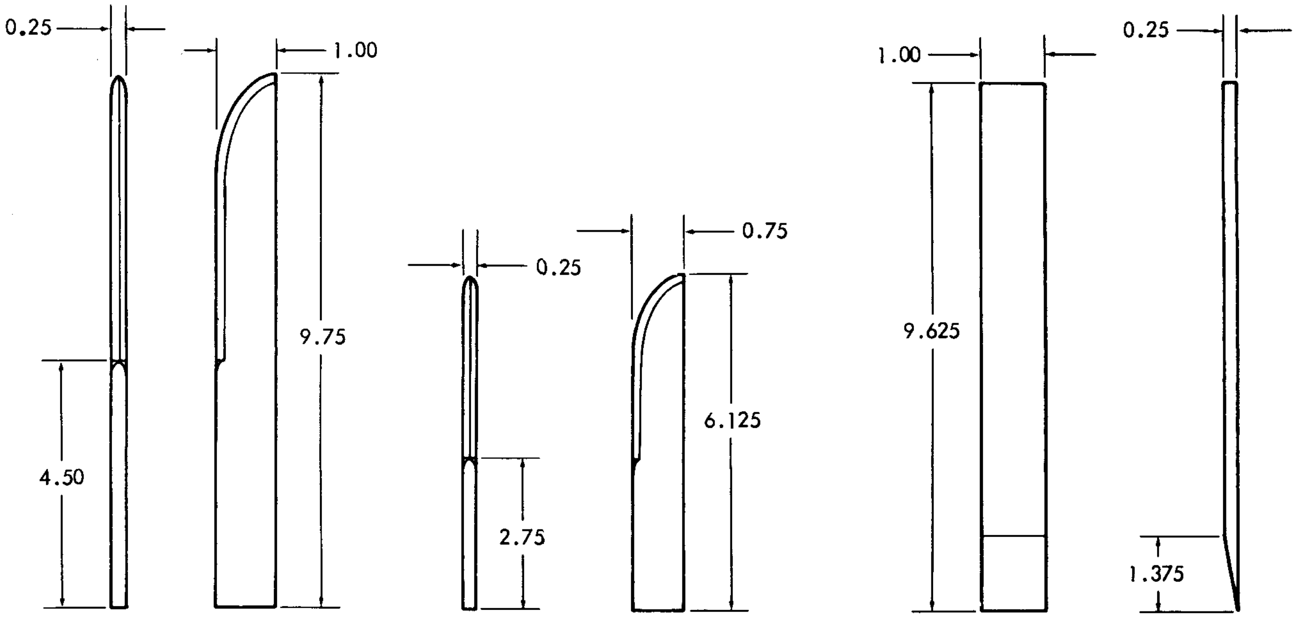
Figure 51-7. Sealant Tack Free Times



MATERIAL:
0.063 2024-T3 ALUMINUM ALLOY.
(ROUND OFF SHARP EDGES)

FILLETING SPATULA

MIXING PADDLE AND BROAD SCRAPER



CLEANING KNIVES

CHISEL AND SCRAPER

NOTE

1. MAKE TOOLS FROM ANY OF THE FOLLOWING MATERIALS, EXCEPT AS NOTED:
   (a) HARD, VULCANIZED FIBER
   (b) HARD MAPLE LUMBER
   (c) OAK LUMBER
2. ALL DIMENSIONS GIVEN IN INCHES.
3. BRUSHES:
   (a) 0.50 INCH ROUND
   (b) 1.00 INCH FLAT
   (c) 2.00 INCH FLAT

JS-1-2-X0/2-5108

Figure 51-8. Sealing Hand Tools

SECTION 51.3

PENETRANT INSPECTION

For penetrant inspection methods see Section 58.1

All Data on page 51-20A/51-20B deleted.
Figures 51-9, 51-10, and 51-11 on page 51-21/51-22 deleted.

MATERIAL	PENETRATION (MINUTES)	EMULSIFICATION (MINUTES)	DEVELOPING (MINUTES)
SAND CASTINGS	1 - 2	3 - 5	1/2 - 1
PERMANENT MOLD CASTINGS	4 - 6	3 - 5	2 - 3
DIE CASTINGS	4 - 6	3 - 5	2 - 3
INVESTMENT CASTINGS	8 - 12	1/2 - 2	4 - 6
FORGINGS	15 - 20	1 - 2	8 - 10
EXTRUSIONS	20 - 30	1 - 2	10 - 15
ROLLED PRODUCTS	10 - 20	1 - 2	5 - 10
WELDS	4 - 6	2 - 5	2 - 3
PLASTICS	1 - 2	1 - 2	1/2 - 1
CERAMICS	1 - 2	1 - 2	1/2 - 1

Figure 51-9. Fluorescent Inspection with ZL-2 and ZL-22 Penetrant

MATERIAL	PENETRATION (MINUTES)	DEVELOPING (MINUTES)
CASTINGS	30	15
EXTRUSIONS	45	23
ROLLED PRODUCTS	45	23
WELDS	30	15
PLASTICS	5	3
CERAMICS	5	3
GLASS	30	15

Figure 51-10. Fluorescent Inspection with ZL-1B Penetrant

MATERIAL	TYPE DEFECT	PENETRATION (MINUTES)
NONFERROUS METALS	HEAT TREAT CRACKS	2
	GRINDING CRACKS	10
	FATIGUE CRACKS	10
PLASTICS	CRACKS	1 - 5
CERAMICS	CRACKS	1 - 5
	POROSITY	1 - 5
CASTINGS	POROSITY	3 - 10
DIE CASTINGS	COLD SHUTS	10 - 20
FORGINGS	CRACKS	20
	LAPS	20
METAL ROLLINGS	SEAMS	10 - 20
WELDS	CRACKS	10 - 20
	PORES	10 - 20

Figure 51-11. Dye Penetrant Inspection

SECTION 51.4
PRESSURIZATION CHECK

Contents

GENERAL 51-23

NORMAL LEAKAGE CHECK 51-23

CABIN UNCONTROLLED LEAKAGE CHECK 51-23

STRUCTURAL PROOF TEST 51-23

51.4.1 GENERAL. Three standard pressure tests are used on the JetStar. The first, a normal pressure check to 4.0 PSI, may be made with a ground pressure source or with the normal pressurization system installed in the airplane. The second, an uncontrolled leakage check, may be made if cabin leakage is found to be excessive during a pressurization system checkout (refer to Chapter 21). The third, a structural proof check to 12.7 PSIG must be performed using a ground pressure source, and follows major structural repair to the fuselage pressurized section or replacement of entrance door, nose cone, or emergency exits.

51.4.2 NORMAL LEAKAGE CHECK. This test should be run after extensive repairs to the fuselage pressure sealant have been made or when it is suspected that the leakage rate is excessive. The airplane should be pressurized according to the procedure given in section 21.1. When it is necessary to check for local leaks; use the following procedure:

a. Open and close the nose cone, the entrance door, emergency exits and windows, and make sure that they are closed and latched.

WARNING

All personnel must be evacuated from the airplane during this test.

b. Make sure that the airplane is tied down securely.

c. Make sure that the static ground lines are attached.

d. Brush leak detector fluid, Purex Corp., Turco Products Div., 24600 South Main St., Carson, CA 90749 (or equivalent) over the repaired area.

e. Slowly raise the pressure in the fuselage to 4.0 PSI, using the procedure given in Section 21.1.

CAUTION

Do not increase pressure at a rate higher than 0.5 PSI per minute, and do not exceed 6.0 PSI.

f. Maintain the pressure at 4.0 PSI for 3 minutes.
RESULT: There should be no bubbling of the leak detector fluid at the repaired area.

g. Slowly reduce the pressure to zero.

h. Remove the pressure source, and return the fuselage to normal.

51.4.3 CABIN UNCONTROLLED LEAKAGE CHECK. Refer to Chapter 21.

51.4.4 STRUCTURAL PROOF TEST. A fuselage pressure test is required following the installation of a new entrance door, emergency exit, or nose section, and following any major structural repair to the fuselage pressurized section. Because of the large volume of the fuselage, a potentially dangerous condition is created when the fuselage is pressurized. When pressurized, the fuselage can suffer major damage, with possible injury to personnel, if struck by any object, such as ground equipment. Use the following procedure when pressure-testing the fuselage:

a. Isolate the airplane, and restrict the area to personnel required to perform the test.

WARNING

All personnel must be evacuated from the airplane during this test.

51.4.4 (Cont).

b. Make sure that the airplane is tied down securely.

c. Remove the following equipment from the airplane:

  1. Air speed indicators
  2. Rate of climb indicators
  3. Altimeters
  4. Machmeter
  5. Mach trim pressure switch
  6. Autopilot pressure switch
  7. Speed warning switch
  8. Altitude control (autopilot)
  9. Dual altimeter and differential pressure gage

Note

All tubing disconnected from the above items must be plugged.

  10. Passenger oxygen mask and regulator.
  11. Passenger oxygen control panel
  12. Crew Oxygen regulator

Note

All tubing disconnected from the above items must be plugged with plugs suitable for the oxygen system.

  13. Standby compass
  14. Turn and slip indicator
  15. Course indicator
  16. Approach horizon
  17. Radar indicator
  18. DMET indicator
  19. Master compass
  20. All electronic equipment installed on the nose rack and shelf
  21. Oxygen walk-around bottle

d. Remove the cabin pressure controller, and do not plug any of the disconnected tubing.

e. Close the isobaric and differential shutoff test valves.

f. Make sure that the manual depressurization needle valve is closed.

g. On Airplanes 5001 through 5125, not modified by Service Bulletin 329-259, disconnect the tubing to the safety valves at the flush static port, fuselage station 294; plug the static port but do not plug the tubing to the safety valves.

On Airplanes 5126 and subsequent, and those modified by Service Bulletin 329-259, remove screened orifice fitting from safety valves and leave port open.

h. Remove the electronic cooling fan from the nose wheel well, and plug the opening.

i. Plug the toilet venturi.

j. Remove all loose equipment within the fuselage.

k. Make sure that the nose cone, the clear vision windshield, the emergency escape hatches, and the main entrance door are closed and properly secured.

l. Connect a pressure gage of 0-to 15-PSI capacity to the sense port in the nose wheel well. Connect a second pressure gage of the same capacity to the ground air conditioning connection in the aft equipment compartment of the fuselage. The gages should be located approximately 15 feet from the airplane.

m. Connect a source of air pressure capable of delivering approximately 60 pounds per minute at 13 PSI to the ground air conditioning connection in the aft equipment compartment of the fuselage.

n. Slowly pressurize the fuselage to 12.7 PSI. Monitor both pressure gages as the fuselage is pressurized.

CAUTION

Do not increase the pressure at a rate greater than 0.5 PSI per minute, and do not exceed 12.7 PSI.

o. Maintain the pressure at 12.7 PSI for 2 minutes.
RESULT: No failure should occur during this time.

p. Slowly reduce the pressure to 8.8 PSI, allow the pressure to stabilize, and observe the flowmeter.
RESULT: The leakage rate should not exceed 19 pounds per minute.

q. Slowly reduce the pressure to zero.

r. Remove the test equipment, and restore the fuselage to normal.

SECTION 51.5

MASS REBALANCE OF CONTROL SURFACES

Contents

Page

GENERAL 51-25

REBALANCING A CONTROL SURFACE 51 25

PAINT APPLICATIONS 51-25

REPAIR OF CONTROL SURFACES 51-27

FIGURES

51.12. Control Surface Balance Weights 51-29
51.13. Calculation Method for Control Surface Rebalance 51-29
51-14. Allowable Adjustment Weights 51-30
51-15. Control Surface Balance Limits 51-30
51-16. Rudder Balance Requirements Without Torque Tube 51-30
51-17. Elevator Balance Requirements Without Torque Tube 51-30
51-18. Aileron Rebalance 51-31
51-19. Rudder Rebalance 51-32
51-20. Elevator Rebalance 51-33

51.5.1. GENERAL. The ailerons, rudder and elevators are mass-balanced control surfaces. Rebalance is required whenever a control surface is replaced, repaired, repainted or modified in any way to change its weight or center of gravity except as specified below.

51.5.2. REBALANCING A CONTROL SURFACE. Control surfaces are rebalanced in a set-up as shown in figures 51-18 through 51-20. The surfaces to be balanced must be complete with trim tabs attached and should not contain any loose internal debris. To permit the torque tube to remain in the aircraft, the rudder and elevator may be rebalanced either with or without the torque tube attached. Measurements must be made accurately to assure good balance. The moment arm must be the measured distance through the surface theoretical center line as indicated in the illustrations and not along the outside skin of the surface. The measured weight must be adjusted with scale corrections and the weight for the wood contact block subtracted to provide a net weight. The net weight (down) of the surface in pounds multiplied by the measured moment arm in inches provides the positive (+) measured unbalance of the control surface. For aileron, rudder complete with torque tube and elevator complete with torque tube; weights should be adjusted to bring the surface within the limits specified in figure 51-15 but in no case should the maximum number or amount of adjustment weights specified in figure 51-14 be exceeded. For rudder and elevator without torque tubes attached, weights should be adjusted on the connecting torque tube as indicated in figure 51-16 or 51-17. It can be assumed that this will result in these surface assemblies being within the limits of figure 51-15 after the torque tubes are connected. Here too, the maximum number of adjustment weights specified in figure 51-14 should not be exceeded.

51.5.3. PAINT APPLICATIONS.

Note

A paint application is considered to be one coat of wash primer, one coat of zinc chromate primer or its equivalent, and two coats of finish paint.

The following applications of paint will not require rebalancing of a control surface:

a. One application of paint to an unpainted surface involving 10 percent or less of the surface skin area.

b. Removal and reapplication of paint to a surface involving 10 percent or less of the surface skin area.

c. Removal and reapplication of paint with assurance that the type of coating and area of coverage is the same as that removed.

51.5.4. REPAIR OF CONTROL SURFACES. (See figure 51-13.) Calculations may be used for rebalancing a control surface that has been repaired without removal of the surface from the aircraft provided that the surface does not already have the maximum number of adjustment weights, specified in figure 51-14, installed.

a. Find the net weight (W1) added by a repair aft of the hinge line by subtracting the weight removed from the weight of parts added.

b. Measure the distance (D1) in inches from the center of repair to the hinge line. The other distance measurement (D2) to be used in the calculations is the distance from the hinge line to where the balance weights will be added. Distance (D2) for the aileron is 4.80 inches, for the rudder it is 7.00 inches, and for the elevator it is 5.00 inches.

c. Multiply the net weight (W1) of the repair by the distance (D1), and divide this result by the distance (D2) to obtain the weight (W2).

$$W2\ (\text{pounds}) = \frac{W1\ (\text{pounds}) \times D1\ (\text{inches})}{D2\ (\text{inches})}$$

The result, W2 (pounds), is the weight that must be added to rebalance the control surface. This figure should be divided by the weight of the smallest applicable adjustment weight, see figure 51-12, to determine the number of weights to be added. Fractions of less than 0.33 should not have any additional weights added; more than 0.33 should have one or more added as required but nothing in excess of the maximum specified in figure 51-14.

WARNING

No modifications should be made in uranium counter weights - see Section 51.9.

CONTROL SURFACE	PART NUMBER	WEIGHT (POUNDS)
Aileron	JW 408-1 JW 484-1 JW 408-2 JW 484-2	0.25 0.45 0.23 0.44
Rudder	JE 208-1	0.47
Elevator	JE 206-1	0.35

JS-2-2-5121

Figure 51-12. Control Surface Balance Weights

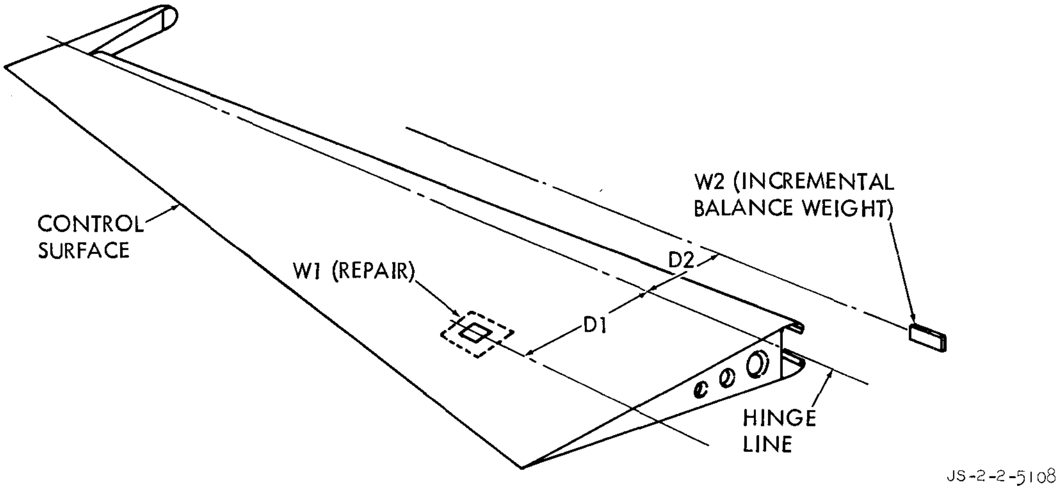


Figure 51-13. Calculation Method for Control Surface Rebalance

CONTROL SURFACE	AIRPLANE	MAXIMUM NUMBER OF ADJUSTMENT WEIGHTS
LEFT OR RIGHT AILERON	ALL	10 JW 484-2 (EQUIVALENT TO 4.4 POUNDS) IT IS PERMISSIBLE TO USE JW 408-1*, JW 484-1*, JW 408-2 AND JW 484-2 WEIGHTS BUT THEY MUST NOT EXCEED A TOTAL OF 4.4 POUNDS NOR EXCEED A MAXIMUM OF 12 ADJUSTMENT WEIGHT.
RUDDER	ALL	4 JE 208-1
LEFT OR RIGHT ELEVATOR	ALL	4 JE 206-1
* JW 408-2 AND JW 484-2 ARE PREFERRED SPARES FOR JW 408-1 AND JW 484-1, RESPECTIVELY.		

Figure 51-14. Allowable Adjustment Weights JS-1-2-X0/2-5114

CONTROL SURFACE	MEASURED UNBALANCE LIMITS (INCH-POUNDS)
LEFT AILERON	+ 10 TO + 20 (PAINTED OR UNPAINTED)
RIGHT AILERON	+ 10 TO + 20 (PAINTED OR UNPAINTED)
RUDDER (LESS TORQUE TUBE) - SEE FIGURE 51-16 TO DETERMINE THE REQUIRED NUMBER OF ADJUSTMENT BALANCE WEIGHTS.	+ 100 TO + 146 (UNPAINTED) + 100 TO + 151 (PAINTED)
RUDDER (WITH TORQUE TUBE)	+ 45 TO + 85 (PAINTED OR UNPAINTED)
ELEVATOR (LESS TORQUE TUBE) - SEE FIGURE 51-17 TO DETERMINE THE REQUIRED NUMBER OF ADJUSTMENT BALANCE WEIGHTS.	+ 80 TO + 102 (UNPAINTED) + 80 TO + 106 (PAINTED)
ELEVATOR (WITH TORQUE TUBE)	+ 7 TO + 25 (PAINTED OR UNPAINTED)

Figure 51-15. Control Surface Balance Limits JS-1-2-X0/2-5115

MEASURED UNBALANCE OF RUDDER ASSEMBLY LESS TORQUE TUBE (INCH-POUNDS)	NUMBER OF JE 208-1 ADJUSTMENT WEIGHTS TO BE INSTALLED ON TORQUE TUBE
+ 100 TO + 138	0
+ 138.1 TO + 141	1
+ 141.1 TO + 145	2
+ 145.1 TO + 148	3
+ 148.1 TO + 151	4

Figure 51-16. Rudder Balance Requirements Without Torque Tube JS-1-2-X0/2-5116

MEASURED UNBALANCE OF ELEVATOR ASSEMBLY LESS TORQUE TUBE (INCH-POUNDS)	NUMBER OF JE 206-1 ADJUSTMENT WEIGHTS TO BE INSTALLED ON TORQUE TUBE
+ 80 TO + 98	0
+ 98.1 TO + 99	1
+ 99.1 TO + 101	2
+ 101.1 TO + 103	3
+ 103.1 TO + 106	4

Figure 51-17. Elevator Balance Requirements Without Torque Tube JS-1-2-X0/2-5117

NOTE

△1 THE SUPPORT JIGS MAY BE CONSTRUCTED OF HARDWOOD FOR TEMPORARY USEAGE. USE STEEL FOR PERMANENT JIGS.

△2 ANGLE MEASURED AT AILERON STATION 68.1.

3. REFER TO FIGURE 51-15 FOR BALANCE LIMITS.

△4 REFER TO FIGURE 51-14 FOR ALLOWABLE WEIGHTS.

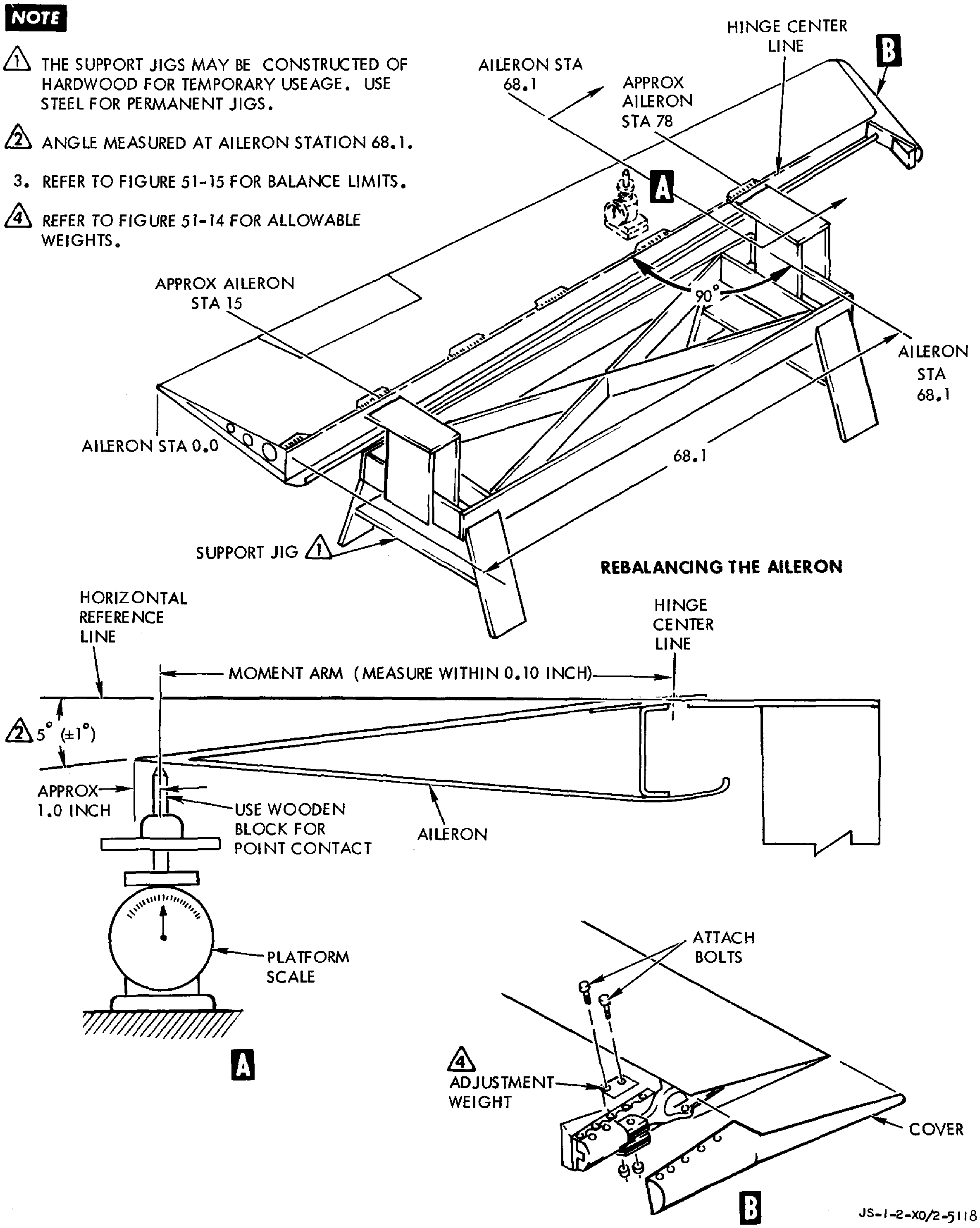
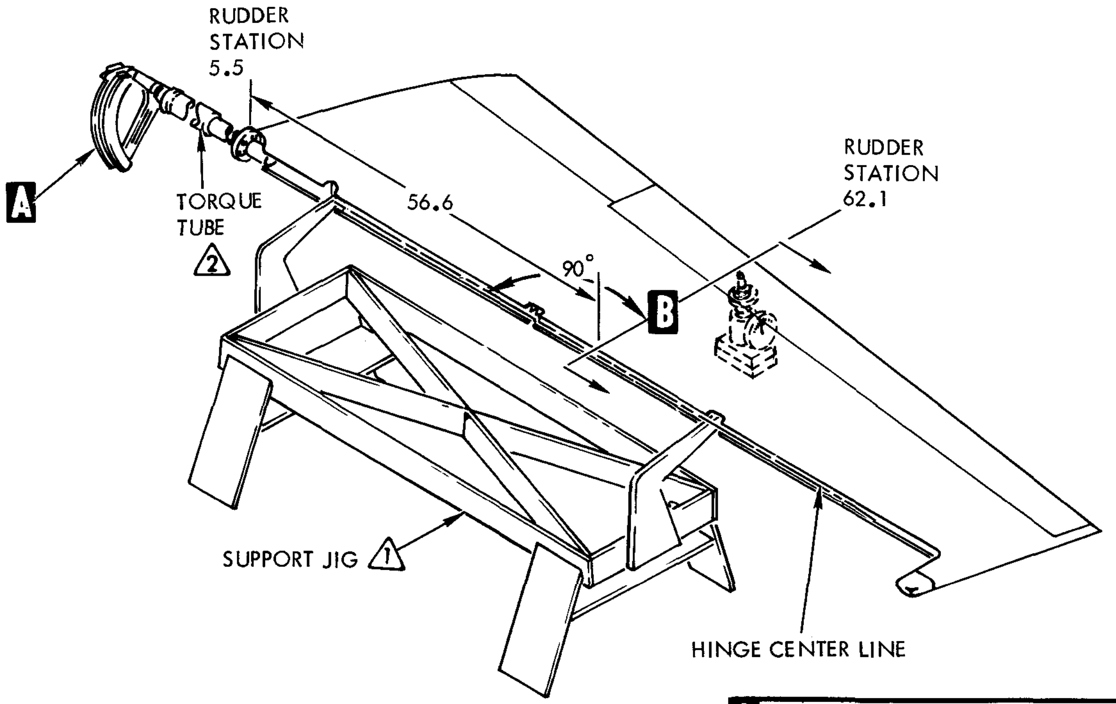


Figure 51-18. Aileron Rebalance



REBALANCING THE RUDDER

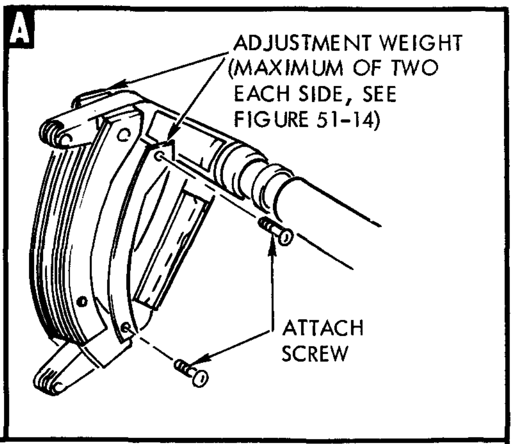
NOTE

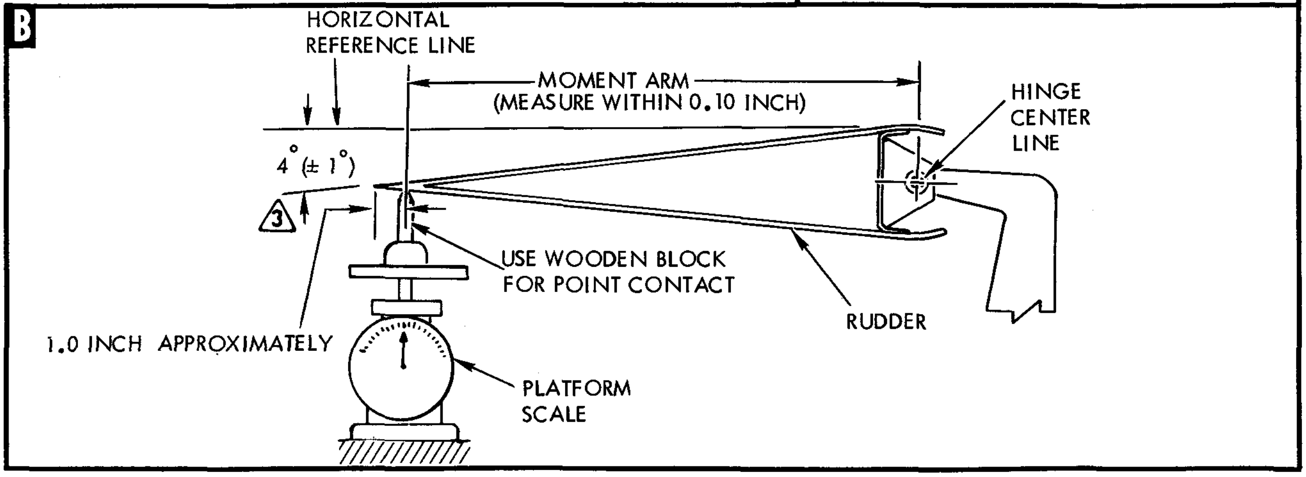
△1 THE SUPPORT JIGS MAY BE CONSTRUCTED OF HARDWOOD FOR TEMPORARY USAGE. USE STEEL FOR PERMANENT JIGS.

△2 THE RUDDER TORQUE TUBE DOES NOT HAVE TO BE REMOVED FROM THE AIRPLANE. REFER TO FIGURE 51-16 FOR ADJUSTMENT WEIGHT REQUIREMENTS WITHOUT TORQUE TUBE.

△3 ANGLE MEASURED AT RUDDER STATION 62.1.

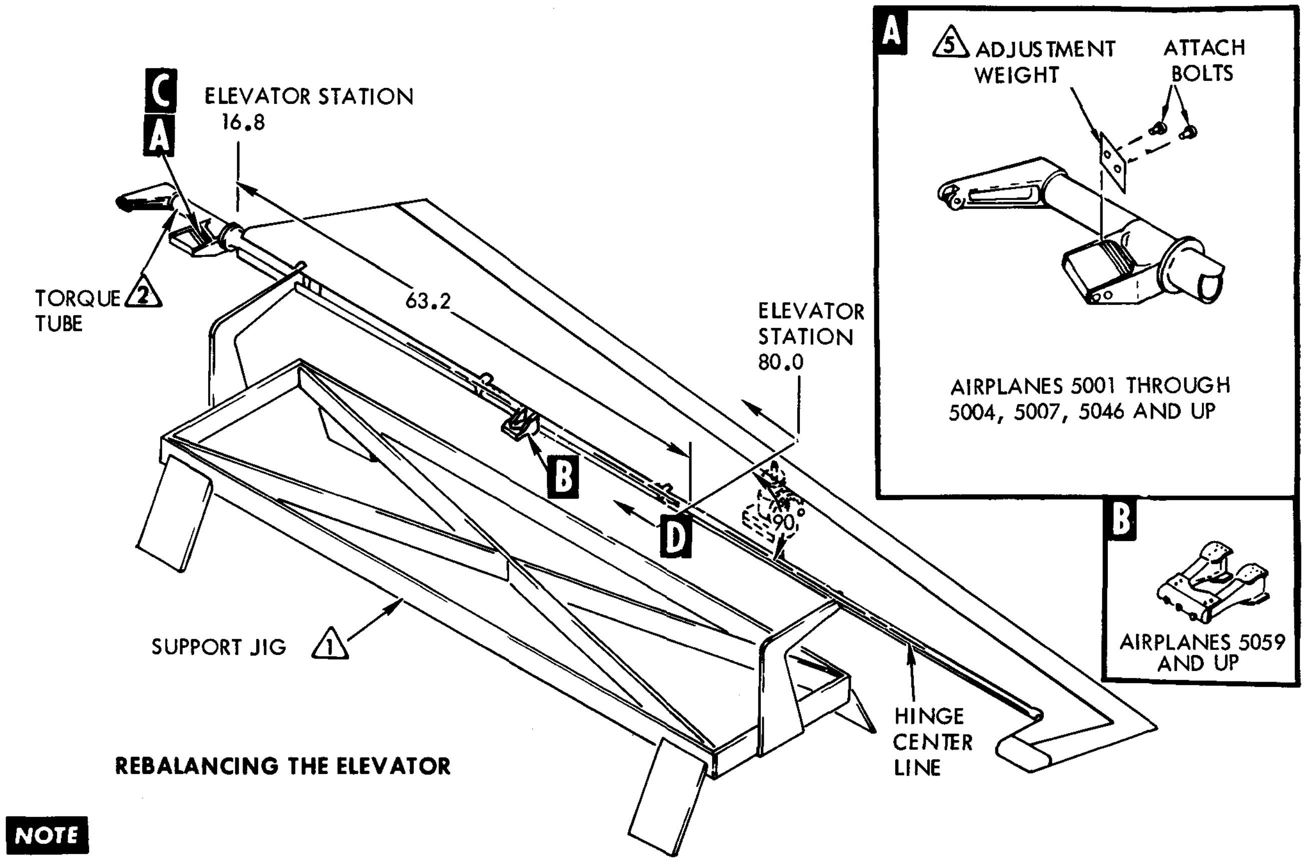
4. REFER TO FIGURE 51-15 FOR BALANCE LIMITS.


A
ADJUSTMENT WEIGHT (MAXIMUM OF TWO EACH SIDE, SEE FIGURE 51-14)
ATTACH SCREW


B
HORIZONTAL REFERENCE LINE
MOMENT ARM (MEASURE WITHIN 0.10 INCH)
HINGE CENTER LINE
4° (± 1°)
3
USE WOODEN BLOCK FOR POINT CONTACT
RUDDER
1.0 INCH APPROXIMATELY
PLATFORM SCALE

JS-1-2-X0/2-5119

Figure 51-19. Rudder Rebalance



REBALANCING THE ELEVATOR

NOTE

1. THE SUPPORT JIGS MAY BE CONSTRUCTED OF HARDWOOD FOR TEMPORARY USAGE. USE STEEL FOR PERMANENT JIGS.
2. THE ELEVATOR TORQUE TUBE DOES NOT HAVE TO BE REMOVED FROM THE AIRPLANE. REFER TO FIGURE 51-17 FOR ADJUSTMENT WEIGHT REQUIREMENTS WITHOUT TORQUE TUBE.
3. ANGLE MEASURED AT ELEVATOR STATION 80.0.
4. REFER TO FIGURE 51-15 FOR BALANCE LIMITS.
5. REFER TO FIGURE 51-14 FOR ALLOWABLE WEIGHTS.

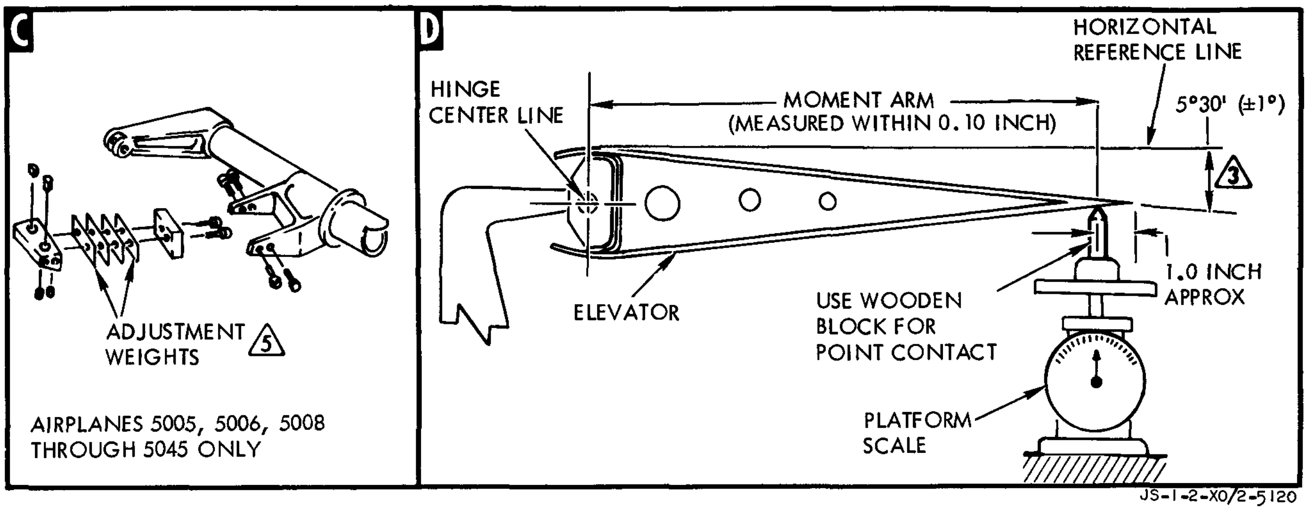

C
ADJUSTMENT WEIGHTS 5
AIRPLANES 5005, 5006, 5008 THROUGH 5045 ONLY
D
HINGE CENTER LINE
MOMENT ARM (MEASURED WITHIN 0.10 INCH)
HORIZONTAL REFERENCE LINE
5°30' (±1°)
3
ELEVATOR
USE WOODEN BLOCK FOR POINT CONTACT
1.0 INCH APPROX
PLATFORM SCALE
JS-1-2-X0/2-5120

Figure 51-20. Elevator Rebalance

SECTION 51.6
CHEMICAL IDENTIFICATION OF MATERIALS

Contents

GENERAL 51-35

PREPARATION OF TEST SOLUTION 51-35

TEST PROCEDURE 51-35

FIGURES
51-21. Chemical Identification of Materials 51-37

51.6.1 GENERAL. Before any member can be repaired its material must be positively identified. This procedure will give positive identification if the material in question is known to be in the following group:

Aluminum or Aluminum Alloy

Magnesium or Magnesium Alloy

Titanium

17-7PH Steel

Type 301 or 302 Stainless Steel

Carbon Steel

Low Alloy Steel

51.6.2 PREPARATION OF TEST SOLUTION. Prepare each of the following chemical test solutions in clean glass containers:

WARNING

Use impervious clothing, gloves, face shield and other appropriate protective clothing necessary to prevent any possibility of skin contact with any chemical or solution used in the following procedure. Avoid breathing the vapors or fumes from the materials in this procedure. Wash hands thoroughly with soap and water after handling.

CAUTION

Acid and acid solutions must always be added slowly to water.

a. Prepare a copper sulphate solution by dissolving 25 grams of copper sulphate crystals ($CuSO_4$ - $5H_2O$) in 100cc of distilled water, and add 2 or 3cc of sulfuric acid (H_2SO_4).

b. Prepare a hydrochloric acid solution by slowly adding 50cc of hydrochloric acid (HCl) to 50cc of distilled water.

c. Prepare a cadmium sulfate solution by mixing 5 grams of cadmium sulfate ($CdSO_4$). 5cc of concentrated hydrochloric acid (HCl), and 3 grams of sodium chloride (NaCl). Dissolve and dilute with distilled water to make 100cc.

d. Prepare a scratch test solution by dissolving 20 grams of potassium nitrate (KNO_3) and 10 grams of sodium hydroxide (NaOH) in 100cc of distilled water.

e. Prepare a sodium hydroxide solution by slowly adding and dissolving 25 grams of sodium hydroxide pellets (NaOH) in 100cc of distilled water. Do not allow the pellets to stick to the bottom of the container.

f. Prepare a sulfuric acid solution by slowly adding 25cc of sulfuric acid (H_2SO_4) to 75cc of distilled water.

Note

It is recommended that the test solutions be transferred to clean, individual, medicine-type bottles that have a dropper in the cap. Label the bottles, and keep them tightly capped when not in use.

51.6.3 TEST PROCEDURE. Perform the test procedure according to the following instructions (see figure 51-21):

a. Remove paint, grease, oil, and foreign matter from small, individual areas of metal surface for each application of chemical test solutions until the material is identified.

b. Abrade the cleaned areas with crocus cloth or metalographic emery paper, Grade O, to remove oxide or protective films and coatings.

51.6.3 (Cont).

Note

If the material is known to be clad aluminum alloy, it is not necessary to abrade through the cladding.

c. Follow the procedure shown on figure 51-21 until the material is identified.

d. Immediately wipe chemical test solutions from the metal surface with clean cloths, and flush thoroughly with clean water.

e. Neutralize wet areas on aluminum or aluminum alloys with 5 percent chromic acid solution; then flush again with clean water.

f. Dry metal surface and adjacent members, as applicable, with clean cloths.

g. Apply the proper anti-corrosion treatment to all airframe areas tested by this procedure.

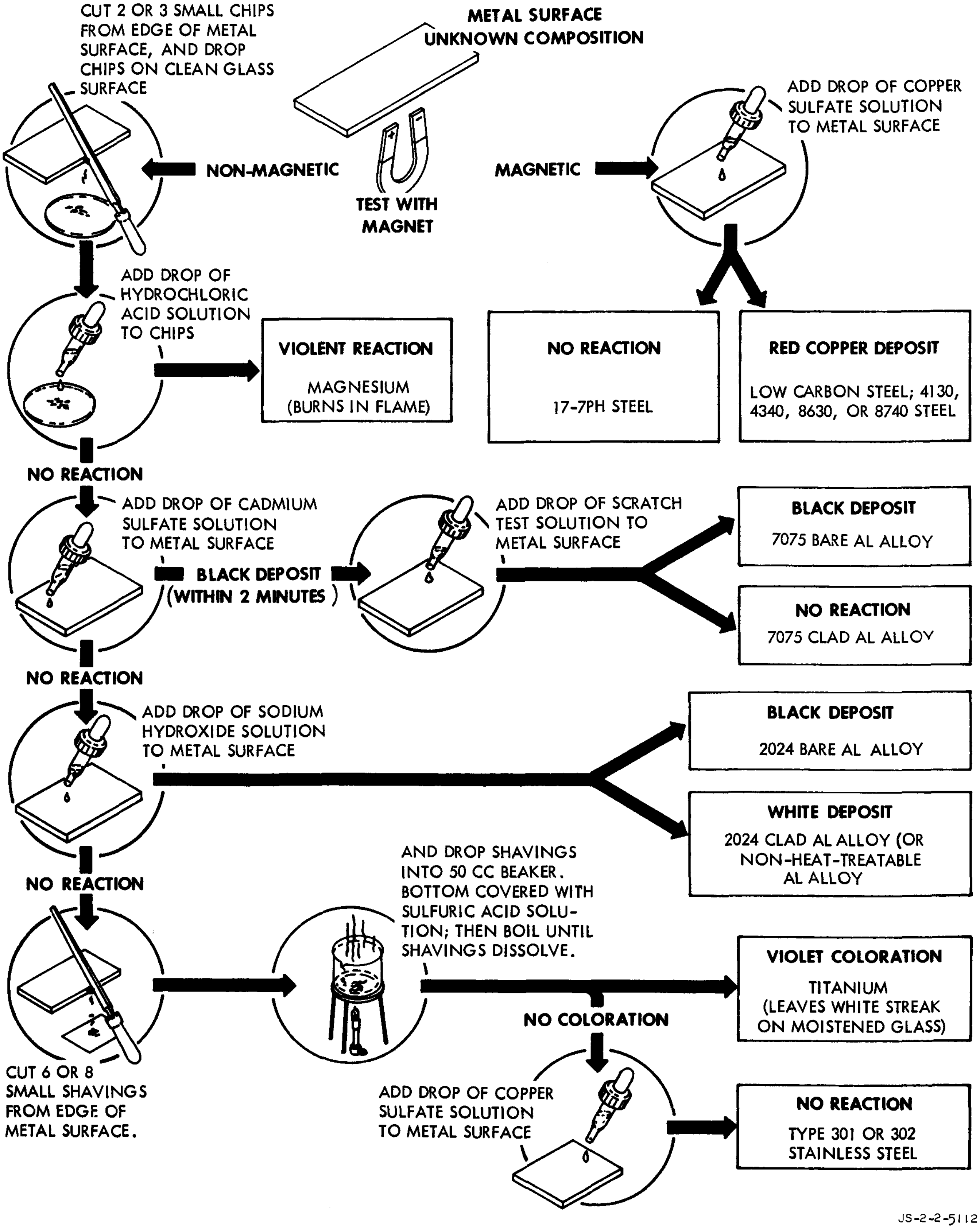

METAL SURFACE
UNKNOWN COMPOSITION
TEST WITH MAGNET
NON-MAGNETIC
MAGNETIC
CUT 2 OR 3 SMALL CHIPS FROM EDGE OF METAL SURFACE, AND DROP CHIPS ON CLEAN GLASS SURFACE
ADD DROP OF COPPER SULFATE SOLUTION TO METAL SURFACE
NO REACTION
17-7PH STEEL
RED COPPER DEPOSIT
LOW CARBON STEEL; 4130, 4340, 8630, OR 8740 STEEL
ADD DROP OF HYDROCHLORIC ACID SOLUTION TO CHIPS
VIOLENT REACTION
MAGNESIUM (BURNS IN FLAME)
NO REACTION
ADD DROP OF CADMIUM SULFATE SOLUTION TO METAL SURFACE
BLACK DEPOSIT (WITHIN 2 MINUTES)
ADD DROP OF SCRATCH TEST SOLUTION TO METAL SURFACE
BLACK DEPOSIT
7075 BARE AL ALLOY
NO REACTION
7075 CLAD AL ALLOY
NO REACTION
ADD DROP OF SODIUM HYDROXIDE SOLUTION TO METAL SURFACE
BLACK DEPOSIT
2024 BARE AL ALLOY
WHITE DEPOSIT
2024 CLAD AL ALLOY (OR NON-HEAT-TREATABLE AL ALLOY
NO REACTION
CUT 6 OR 8 SMALL SHAVINGS FROM EDGE OF METAL SURFACE.
AND DROP SHAVINGS INTO 50 CC BEAKER. BOTTOM COVERED WITH SULFURIC ACID SOLUTION; THEN BOIL UNTIL SHAVINGS DISSOLVE.
VIOLET COLORATION
TITANIUM (LEAVES WHITE STREAK ON MOISTENED GLASS)
NO COLORATION
ADD DROP OF COPPER SULFATE SOLUTION TO METAL SURFACE
NO REACTION
TYPE 301 OR 302 STAINLESS STEEL
JS-2-2-5112

Figure 51-21. Chemical Identification of Materials

SECTION 51.7
CORROSION CONTROL

Contents Page

GENERAL 51-39

CORROSION CONTROL 51-39
  Corrosion Control for Repairs 51-39
  Anti-Corrosion Treatment for Aluminum Surface not to be Painted 51-40
  Anti-Corrosion Treatment for Aluminum Surfaces to be Painted 51-40
  Alternate Method for Anti-Corrosion Treatment of Aluminum Surfaces 51-40A
  Inspection for Corrosion 51-41

TREATMENT OF AREAS SUSCEPTIBLE TO CORROSION 51-42
  Integral Fuel Tanks 51-42
  External Wing Tanks 51-43
  Area Under Fairings 51-43
  Wing 51-43
  Magnesium Speed Brake 51-44
  Uranium Counterweights 51-44
  Exterior Clad Skins 51-44
  Exterior Protection of Inactive Aircraft 51-44
  Removal of Corrosion and Treatment of Internal Aluminum Surfaces 51-44

ANTI-CORROSION SEALING 51-44A
  Removal of Fuel Tank Microbiological Contamination 51-44A

FIGURES
  51-22. Fuselage Anti-Corrosion Sealing 51-45

51.7.1 GENERAL.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

Corrosion is the deterioration of a metal by chemical or electrochemical reaction with its environment. On aircraft, there occur several types of corrosion if proper preventive care is not maintained. These types are general surface attack, galvanic (bimetallic), intergranular, stress corrosion cracking, fretting corrosion, corrosion fatigue, and filiform. Because the environment is the key to corrosive degradation, it is a well-known philosophy in the aviation industry that the amount and frequency of housekeeping, inspections, and corrosion control maintenance are directly proportional to the severity of the geographical locations where the aircraft are based and operate. It is obvious, therefore, that aircraft which operate from coastal airports, from highly industrial regions with fallout of waste products, from areas of strip mining activities or from tropical sites where there exists a combination of high humidity and abundant vegetation, require much greater attention to corrosion prevention and control than do aircraft which operate from average inland environments.

51.7.2 CORROSION CONTROL.

Keeping the aircraft clean and dry is the best approach to corrosion control. To accomplish this goal, observe the following:

a. Keep surfaces clean on a regularly scheduled basis.

b. Keep surfaces protected with a paint finish where applicable.

c. Keep drain holes open and free of debris.

d. Give particular attention to areas containing absorbent materials.

51.7.2.1 CORROSION CONTROL FOR REPAIRS. When repairs are made using repair members of attachments which are of a different material from that of the member being repaired, the repair may corrode unless proper precautions are observed. Perform the following steps to guard against corrosion when making structural repairs:

a. Clean up damaged members.

51.7.2.1 (Cont).

b. Remove damaged paint and clean existing structure in the damaged area. Remove any sealing material in the area to be repaired.

CAUTION

Do not scrape paint or sealant with steel or metal scrapers because they will scratch the aluminum surface. If scrapers are necessary they should be made from non-metallic material. (See figure 51-8.)

c. Fabricate repair members and fillers according to the repair instructions in this chapter.

d. Assemble repair members and fillers on the existing structure, and drill the attachment holes.

e. Remove repair members and fillers, and remove burrs from them. Clean and treat the fabricated repair members and fillers per paragraph 51.7.2.2 or 51.7.2.3. Remove burrs from existing structure, and clean with Specification TT-T-266 lacquer thinner, Specification P-D-680 (Type II) dry cleaning solvent, or other suitable wax-free solvent. Wipe dry with clean cloth.

f. Paint repair members, fillers, and existing structure with two coats zinc chromate primer.

g. After the zinc chromate primer is dry, assemble repair members and fillers on the existing structure.

h. Clean repair attachments with P-D-680 (Type II) solvent.

i. Dip the clean, dry attachments (pins, screws, bolts, rivets) into the zinc chromate primer, and install the attachments while the primer is still wet.

j. If a repair is located on the pressurized boundary of the fuselage, seal the repair according to the instructions in Section 51.2.

51.7.2.2 ANTI-CORROSION TREATMENT FOR ALUMINUM SURFACES NOT TO BE PAINTED. Apply the following anti-corrosion treatment to clean aluminum surfaces for protection against corrosion. This treatment does not leave a stain on the surface; therefore, where a bright clear surface is desirable, it can be used on surfaces which are not to be painted.

a. Mix a solution of conversion coating agent per MIL-C-81706 such as "Alodine 1000" powder (American Chemical Paint Co.) in the proportion of 0.4 to 0.5 ounce of chemical to one gallon of water. Adjust the acidity of the solution to a pH of 2.6 to 3.5 by adding ammonium hydroxide or nitric acid as necessary. Use a plastic, stainless steel, or rubber container. Do not use a glass container, for "Alodine 1000" reacts with glass.

WARNING

Wear rubber gloves and a respirator when working with "Alodine 1000." Do not breathe dust from the powder or vapor from the solution. If the powder or solution comes in contact with the skin, immediately wash the affected area with water.

b. Sponge or spray the solution onto the part. Be sure the sponges and the part are clean. When applying the solution by the spray method, use low-pressure, acid-resistant spray equipment with a fine (but not atomizing) spray, or use conventional pressure-pot spray equipment.

c. Allow the solution to remain on the part according to ambient temperature as follows:

10°C to 21°C (50°F to 70°F) - 6 minutes

21°C to 32°C (70°F to 90°F) - 5 minutes

32°C to 38°C (90°F to 100°F) - 4 minutes

Note

If the temperature is lower than 10°C (50°F) or above 38°C (100°F), do not treat the parts. Wait or proper temperature conditions.

d. Upon expiration of the time limits in step (c), rinse the parts thoroughly with cold water, and dry with a soft, clean cloth.

51.7.2.3 ANTI-CORROSION TREATMENT FOR ALUMINUM SURFACES TO BE PAINTED. Apply the following anti-corrosion treatment to clean aluminum surfaces for protection against corrosion. This treatment will leave a stain and is therefore better adapted to use on surfaces which are to be painted. However, if a bright clear metal surface is not necessary, this treatment may also be used for surfaces not to be painted.

a. Mix a solution of "Alodine 1200" (American Chemical Paint Co.) in the proportion of 3 ounces of chemical to 1 gallon of water. Adjust the acidity of the solution to a pH of 1.5 to 1.6 by adding ammonium hydroxide or nitric acid as necessary. Stir the mixture thoroughly. Do not mix the solution in a glass container, for "Alodine 1200" reacts with glass. Use a plastic, stainless steel, or rubber container.

WARNING

Wear rubber gloves and a respirator when working with "Alodine 1200." Do not breathe fumes or dust from the powder, or vapor from the solution. If the powder or solution comes in contact with your skin, wash the affected area immediately with water.

51.7.2.3 (Cont).

b. Sponge or spray the solution onto the part. Be sure the sponges and the part are clean. When applying the solution by the spray method, use low-pressure, acid-resistant spray equipment with a fine (but not atomizing) spray.

c. The activity of the "Alodine 1200" solution may be controlled by diluting the solution with fresh water. Mix ratios of 1:1, 1:2, 1:3, and even as diluted as one part of solution to 10 parts of water may be used.

d. After the chemical treatment solution has been allowed to stand for the required time, rinse thoroughly and inspect the surface for a uniform golden color. If the golden color is not uniform, the surface is not uniformly or sufficiently clean. Reclean the surface, and reapply the anti-corrosion treatment. When the golden color is uniform and the solution has remained on the required time, rinse thoroughly and wipe dry with clean, dry cloths.

e. Direct a stream of dry compressed air into all cracks and crevices to insure removal of all entrapped water. Allow to dry for at least one hour.

Note

Non-clad skins, such as machined wing skins, should be completely dried with forced hot air, 49° to 60°C (120° to 140°F), or infrared lamps. A surface temperature of 49°C (120°F) will assure adequate drying.

Note

If neither "Alodine 1000" or "Alodine 1200" is available, provide protection by application of a 5 percent chromic acid solution, heated to between 54°C (130°F) and 60°C (140°F). If heating facilities are not available, minimum protection may be accomplished by applying the chromic acid solution cold. Prepare the solution by dissolving 2 to 4 ounces (avoirdupois) of chromic acid crystals and ¼ ounce of Nacconol NR (National Aniline Division) in one U.S. gallon of water. An alternate method is dissolve chromic acid crstals in water until a mixture having a specific gravity of from 1.035 to 1.040 is obtained; add enough Nacconol NR to give sufficient wetting ability to prevent the solution from bubbling on a piece of clean scrap aluminum alclad. A 5 percent solution of chromic acid should not be harmful to human skin, and the stain it produces will wear off after a short time. Rubber gloves may be worn if desired. When applying the solution, keep the area wet from 3 to 5 minutes. Wipe off the excess solution with a damp cloth, and let the area dry.

CAUTION

Do not allow cloths containing chromic acid to dry before being washed. They might create a spontaneous fire hazard.

51.7.2.4 ALTERNATIVE METHOD FOR ANTI-CORROSION TREATMENT OF ALUMINUM SURFACES.

Note

This procedure uses fast, active chemical solutions and, as a result, extreme care is required in their use. It is especially important to rinse all the solution away after the proper coating has been obtained.

Where the aluminum surface is clean, a protective coating can be applied as follows. This treatment will leave a stain and is therefore better adapted to use on surfaces which are to be painted. However, if a bright clear metal surface is not necessary, this treatment may also be used for surfaces not to be painted.

a. Mix a solution of conversion coating agent per MIL-C-81706 such as "Alodine 1200" (American Chemical Paint Co.) in the proportion of three ounces of chemical to one gallon of water. Adjust the acidity of the solution to a pH of 1.5 to 1.6 by adding ammonium hydroxide or nitric acid as necessary. Stir the mixture thoroughly. Do not mix the solution in a glass container, for "Alodine 1200" reacts with glass. Use a plastic, stainless steel, or rubber container.

WARNING

Wear rubber gloves and a respirator when working with "Alodine 1200." Do not breathe fumes or dust from the powder, or vapor from the solution. If the powder or solution comes in contact with your skin, wash the affected area immediately with water.

51.7.2.4 (Cont.)

b. Apply the solution with a "Scotch Brite" pad (ultra fine or very fine grade only) and lightly abrade the surface with the "Scotch Brite" pad. The solution and light abrasion will clean and form conversion coat in one operation. If the solution powders, dilute the solution with fresh water.

c. Allow the solution to remain on the part until a golden color is visible through the solution. Rinse immediately with fresh water and dry.

Note

If the temperature is lower than 10°C (50°F) or above 38°C (100°F), do not treat the parts. Wait for proper temperature conditions.

d. After the chemical treatment solution has been allowed to stand for the required time, rinse thoroughly and inspect the surface for a uniform golden color. If the golden color is not uniform, the surface is not uniformly or sufficiently clean. Reclean the surface, and reapply the anti-corrosion treatment. When the golden color is uniform and the solution has remained on the required time, rinse thoroughly and wipe dry with clean, dry cloths.

e. Direct a stream of dry, compressed air into all cracks and crevices to ensure removal of all entrapped water. Allow to dry for at least one hour.

Note

Non-clad skins, such as machined wing skins, should be completely dried with forced hot air, 49° to 60°C (120° to 140°F), or infrared lamps. A surface temperature of 49°C (120°F) will assure adequate drying.

51.7.2.5 INSPECTION FOR CORROSION. Visual inspection is the best way to detect corrosion. Radiographic and ultrasonic inspection techniques, which require the use of extensive equipment, are other methods that are available for corrosion detection.

a. The first appearance of corrosion on unpainted aluminum surfaces is in the form of deposits or spots, especially around skin seams, lap joints, areas where sand or dirt and grime collect, or in niches and crevices where traces of cleaning compound have accumulated. In looking for corrosion, give particular attention to the underside of the fuselage, wing flaps, ailerons, actuating mechanisms, and the upper and lower surfaces of the wings. Moisture in these locations does not evaporate as rapidly as in other locations. Areas subjected to battery acids, relief-tube spray, smoke, and exhaust fumes require close attention and frequent maintenance.

b. Corrosion will be less on painted and aluminum-clad surfaces; however, corrosion will attack protected surfaces if moisture permeates the protective coating. When this happens, the affected areas generally have a scaly, or blistered appearance, or sometimes discoloration of paint. Corrosion on aluminum, magnesium, and plated steel surfaces can often be recognized if the area is dulled or pitted, and sometimes by whitish, greyish, or reddish powdery deposits, respectively.

c. When inspecting interior surfaces and lap joints, give particular attention to areas where drain holes may become clogged, fittings and braces, compound parts inside the airplane, sections where foreign matter or moisture may accumulate because of improper drainage or disposal outlets, structure underneath floor boards, and faying surfaces. Make special inspections of areas where magnesium alloys are used.

d. Areas where dissimilar metals are in contact must be closely inspected, as corrosion is more

51.7.2.5 (Cont).

likely in these areas. Corrosion is accelerated when appreciable moisture is present between dissimilar metals. Leather, neoprene, soundproofing, and insulating materials can pick up moisture and hold it, causing corrosion.

51.7.3 TREATMENT OF AREAS SUSCEPTIBLE TO CORROSION.

51.7.3.1 INTEGRAL FUEL TANKS. Corrosion within integral fuel tanks is due to the presence of water or microbiological growth (bacteria and fungus) in the fuel tank. The presence of fungus growth can be determined by visually examining fuel samples from the sump drains. Such growth may appear dark red, brown, greenish brown, or light pink in color. Accomplish the following procedure for the inspection, cleaning, and corrosion control of integral fuel tanks:

WARNING

Refer to sections 12.1 and 28.2 for fuel tank safety precautions.

a. Periodically obtain samples of liquid from sump drains to determine if contamination is present; however, this may not always show the presence of contamination. The tank must be completely drained periodically and carefully examined through the boost pump access doors and fuel quantity probe openings, giving special attention to the sump areas. If any contamination is found, the wing upper surface must be opened and the tank washed out and closely inspected for corrosion. Any water with dirt or solids settled at the bottom is a strong indication that a corrosive condition exists.

b. If contamination is suspected, drain all fuel from the tanks, including the residual fuel from the sumps. Inspect the residual fuel to determine if any bacterial or fungus growth is present. Special attention should be given to the two inboard bays of each tank, and inside and outside surge box walls, since corrosion is most likely to occur in these areas.

Note

If a program of regular checks for microbiological contamination has been carried out on a given aircraft and these checks should suddenly change from negative to positive, the probability of corrosion damage is remote and does not warrant opening the tank for the extensive measures that follow. Such a case should be treated as indicated in paragraph 51.7.5.

c. Open the tanks and remove the remaining fuel by using a vacuum system or sponges and mops. For personal comfort, remove final traces of fuel and fuel vapor by circulating clean, warm air through the tanks.

d. If the condition of the tanks indicates that cleaning is necessary, flush the tanks with water mixed with an alkaline water base cleaner (MIL-C-25769, B and B 713G, Cee-Bee-A-69X, Eldorado ED-305L, Oakite Fleetline JC-6, Penwalt 791B, Turco Air Tec 20, or equal). Mix one gallon of cleaner with 50 gallons of hot water (120 to 140°F). Scrub with a soft brush to clean problem areas. Do not use stiff brushes which will damage sealants or sealant topcoats. Flush the tanks with clean water to remove all traces of cleaner, dirt, contamination or residual fuel. Repeat this procedure until the tanks are clean.

e. Remove all excess water from the tanks with clean cloths or sponges. Then dry completely with clean, warm air (140°F maximum). Make sure the entire cleaning operation, through drying, can be accomplished in an uninterrupted sequence.

f. If bacteria or fungi were present in the tank, the tank should be sprayed with a solution of isopropyl alcohol (TT-I-735) mixed with equal parts of water. Keep all areas of tank wet for 15 minutes to effect a kill off microorganisms. Remove all puddles of alcohol/water with sponges or air powered vacuum cleaners and dry tanks.

CAUTION

Alcohol vapors are explosive.

g. As an alternate method you may spray the tanks with a solution of MIL-I-27686 icing inhibitor (Phillips Petroleum Co. PFA 55MB) and water. The additive is a mixture of approximately 99.6% ethylene glycol monomethyl ether and 0.4% glycerol. The recommended ratio for the spraying operation is a 25% solution of the Specification MIL-I-27686 material in water. The tanks should remain open for this spraying operation. After spraying with the additive solution, drain all excess from the tanks. It is not necessary to completely dry out the tank prior to refueling.

h. Inspect each square inch of the entire tank interior for signs of corrosion. Sealant is applied on joints, seams, and over fasteners. Aircraft 5001 thru 5045 have a pink-dyed Buna-N top coating applied over the entire tank interior by fill-and-drain techniques. Aircraft 5046 and up and aircraft in compliance with Service Bulletin 329-168 have a spray coat of MIL-C-27725 urethane primer protective coating on the entire tank interior. Look for dark brown or black Buna-N pinholes or breaks in the sealant, loose sealant, and pits in the metal. If corrosion is suspected and large areas must be cleaned, this may be done using phenolic scrapers

51.7.3.1 (Cont).

to remove sealant, followed by wiping with rags dampened with TT-M-268 methylisobutylkeytone (MIBK) or TT-M-261 methylethylkeytone (MEK). Use a wood or phenolic scraper to remove all loose coating. Reapply MIBK or MEK as necessary. Removal of small areas of corrosion-protection coating can best be accomplished without the use of MIBK or MEK. Use a wood or phenolic scraper to remove enough coating for cleaning and repair of the metal.

CAUTION

Use MIBK or MEK with extreme care. The faying surfaces of all parts in the JetStar tanks are sealed during assembly. This seal can be destroyed if MIBK or MEK is allowed to seep between the faying surfaces. The faying surface seal cannot be replaced without disassembly of the structural parts.

i. If corrosion is found, remove it as follows:

(1) Open any blisters by scraping off the top of the blister with a carbide-tipped scraper to expose the pocket. Do not use rotary files or chisels.

(2) Mechanically clean all pits and open blisters, using soft-rubber-bonded abrasive wheels. The abrasive in the wheel must not be any coarser than medium alumina (aluminum oxide). Blend the reworked area into the surrounding metal in accordance with the instructions in Section 59-2.

(3) Polish the reworked area with 280-grit abrasive paper, followed by 400-grit abrasive paper or a soft-rubber-bonded abrasive wheel. The abrasive in the wheel must not be coarser than fine alumina.

j. Apply Alodine 1200 conversion coating per paragraph 51.7.2.4.

k. Refer to Section 59.2.2.4 for allowable corrosion limits to determine if structural repair is required. Any structural repair should be fay surface sealed during installation, using inhibited sealing compound MIL-S-81733. The repaired area should be post-assembly sealed as applicable, using MIL-S-8802 sealant. Refer to Section 28.2 and figure 28-12.

Note

Anytime sealant is applied, adhesion promoter (see figure 28-12) must be used and allowed to dry for a minimum of 30 minutes.

l. For application of sealant and method of mixing sealants to be applied over reworked areas, refer to Section 28.2.

m. Apply a brush coat of MIL-S-81733 inhibited sealant over the cleaned and dried reworked area. If the reworked area is in the tank bottom, apply the sealant so that the center of the reworked area is slightly higher than the surrounding surfaces, to prevent the forming of a pocket that can hold water. Scrub the sealant onto the surface to assure adhesion, then add the sealant to the proper thickness. Overcoat **the inhibited sealant with MIL-C-83019 polyurethane topcoat. The overcoat should overlap adjacent surfaces by 1/2 inch maximum. This topcoat should be allowed to cure tack free (2 to 4 hours).**

51.7.3.2 EXTERNAL WING TANKS. The external wing tanks are constructed basically of clad aluminum alloy and present much less of a potential corrosion area than the integral fuel tanks. These tanks are sealed in the same manner as the wing tanks. If any corrosion occurs, it should be repaired in the manner described for the integral tanks.

51.7.3.3 AREA UNDER FAIRINGS. Fairings should be removed periodically and visually examined for corrosion. Both the inside surface of the fairing and the structure covered by the fairings should be examined. Blistered paint is often an indication of metal corrosion underneath. Where corrosion is evident, strip the paint, remove corrosion and corrosion products by grinding and sanding, and hand-alodize the reworked surfaces. After thorough drying, paint the inner surfaces of the fairings with primer. Set parts aside to insure full cure of the primer before reinstallation. Paint the surfaces covered by the fairing with primer. After the paint is cured, apply grease (such as MIL-C-16173 Grade IV or Petrolatum VV-P-236) to the painted mating surface of the fairing. Check and assure that all drain holes are open. Clean with naphtha the surface of the structure that is to be covered. Apply sealing compound, Specification MIL-S-8802, Class B (Refer to Section 51.2.) to these cleaned surfaces. Reinstall the fairing, squeezing out any excess filleting sealing compound. Be careful to coat screws with grease prior to installation.

51.7.3.4 WING. The 7075-T6 chemically milled wing panel extrusions are relatively susceptible to electrochemical corrosion. Whenever this type of corrosion becomes evident, it must be removed after stripping the paint by the mechanical methods previously described for the integral fuel tanks. This removal includes removal of fasteners, opening of blisters, removal of corrosion products, and fairing or smoothing or blending the affected area into the surrounding metal surface. The reworked area is then hand-alodized. Fasteners should be installed with wet sealing compound, Specification MIL-S-81733 or MIL-S-8802 (Refer to Section 51.2.) so that all voids between the fastener and the surface of the hole will be filled to prevent the entry of corrosive liquids. If any structural repairs are found to be necessary, such parts should be installed with wet sealant in all the faying contacting surfaces. The organic finish system or paint must be reapplied to the reworked areas.

51.7.3.5 MAGNESIUM SPEED BRAKE. The magnesium speed brake is designed and fabricated with corrosion resistance as a foremost consideration. This chemically milled casting has a Dow 17 anodic finish. Following anodizing, the part is painted with one coat of acid wash primer, Specification MIL-C-8514, two coats of epoxy primer, LAC 37-722, Type I, and one coat of "Corogard 14" (an aluminum-pigmented, vinyl thiokol). The "Corogard 14" has excellent abrasion resistance. The interior surfaces of the speed brake door have two coats of walkway coating, Specification MIL-W-5044, Type II, rough texture, (Refer to figure 51-22.), in lieu of the final "Corogard 14" topcoat. The interior surface of the brake is used as a ladder to enter the aft, unpressurized portion of the fuselage above the door, and the walkway coating provides still greater abrasion resistance and water.

a. If the organic and chemical finishes are damaged or removed, corrosion can occur and will be evident as a greyish powder. Any corrosion under the paint will cause the paint to blister or flake off.

b. If corrosion occurs, clean the affected area with solvent, and remove the paint and loose corrosion with aluminum wool or a stiff fiber brush. Mask off all areas not requiring treatment, as well as all rubber parts and pressed-in inserts. Dissolve the remaining corrosion with chromic acid (5 to 6 ounces in 1 quart of water). Stubborn areas may require the addition of about 1 ounce of sodium nitrate to each quart of pickling solution. Rinse well with water, and treat the area with Dow 19 solution.

c. This solution is made up of 1-1/3 ounces chromic acid, 1 ounce of calcium sulfate, and water to make 1 gallon. The solution must be stirred vigorously for 15 minutes before use. The magnesium surface must be kept wet with solution until a brassy film is produced (1 to 3 minutes). Following application, rinse with cold water, and dry. Acid wash primer and epoxy primer must be applied to the reworked area. If any voids, pits, or low areas result from the corrosion and its subsequent removal, fill these areas with sealing compound or aerodynamic sealer. Apply the appropriate topcoats over the primed and sealed areas to match the surrounding area.

51.7.3.6 URANIUM COUNTERWEIGHTS. Refer to Section 51.9 for corrosion control procedures, precautions, handling, and disposal instructions for depleted uranium counterweights.

51.7.3.7 EXTERIOR CLAD SKINS. Most of the aircraft exterior surfaces, with the major exception of the wing box machined surfaces, are alclad aluminum alloy. It is very important to keep the aircraft skin clean and in a passive condition. If no corrosion is present, keep unpainted aircraft skins clean and passive as follows:

a. Clean the skin area with alkaline, water-based, emulsion cleaner (Specification MIL-C-25769, B and B 713G, Cee Bee A-69X, Eldorado ED-30SL, Oakite Fleetline JC-6, Penwalt 791B, Turco Air Tec 20, or equal.) mixed in accordance with manufacturer's recommendation.

b. Rinse thoroughly with fresh water, concentrating on seams and joints.

c. Apply "Alodine 1000" clear conversion coating to passivate the surface. (Refer to paragraph 51.7.2.2 for preparation and application of "Alodine 1000" solution.)

d. Rinse completely. Pressure-rinse all seams and joints. Remove excess water with squeegees, sponges, chamois skins, or clean, damp cloths.

e. If corrosion is present, treat unpainted aircraft skin as follows:

(1) Clean the area with mixed alkaline, water-based emulsion cleaner (Specification MIL-C-25769).

(2) Rinse thoroughly with fresh water.

(3) Brighten the area with metal conditioner and brightener (MIL-C-38334, Type I, B and B M34H, Intex 8019, Pennwalt 715K, Turco Metal Glo No. 6 mixed in accordance with manufacturer's recommendations. A light-duty rubbing compound for alclad skins such as Turco "Red-X", used in conjunction with the brightener, will aid in removing stubborn stains and obtaining a uniformly brightened surface.

(4) Rinse the surface, and pressure-rinse seams and joints.

(5) Reclean and neutralize the surface with Specification MIL-C-25769 alkaline emulsion cleaner.

(6) Rinse the surface, and pressure-rinse seams and joints.

(7) Apply "Alodine 1000" to passivate the surface. (Refer to paragraph 51.7.2.2 for preparation and application of "Alodine 1000.")

(8) Rinse thoroughly. Pressure-rinse all seams and joints. Remove excess water with squeegees, sponges, chamois skins, or clean, damp cloths.

f. Much better and more permanent protection can be obtained by painting the entire aircraft exterior. Before painting, the above cleaning and passivation procedure should be followed except that "Alodine 1200" (or 1200S) should be used instead of "Alodine 1000." (Refer to paragraph 51.7.2.3 for preparation and application of "Alodine 1200" solution, and Section 51.8 for the paint system.)

51.7.3.8 EXTERIOR PROTECTION OF INACTIVE AIRCRAFT. (Refer to Section 51.10.)

51.7.3.9 REMOVAL OF CORROSION AND TREATMENT OF INTERNAL ALUMINUM SURFACES. This procedure cleans, removes corrosion, and applies the chemical conversion coating to aluminum in one operation. As a result, it is a fast method and the chemicals involved are very

51.7.3.9 (Cont).

active, requiring additional care in their use. It is especially important to rinse all the solution away after the proper coating has been obtained. Where corrosion is present on the surface of the aluminum, correction of this condition can be made in the following manner. Cleaning of surface and application of protective coating are accomplished in one operation.

a. A colored conversion coating solution should be made by adding 3 ounces of "Alodine 1200S" powder (Amchem Products, Inc., Ambler, Pennsylvania) to each gallon of water in a stainless steel or polyethylene container. Stir the solution with a wooden paddle until the powder is dissolved. A small amount of insoluble material may settle out and may be disregarded. Heating the solution is not required. Add 20 milliliters of concentrated nitric acid (approximately 42 degrees Baume) to each gallon of solution, mix, and allow the solution to stand for at least one hour before use. Conversion coating solutions are sensitive to contamination from dirt and old solution; therefore, care should be taken to ensure mixing vessels and area are clean. Fresh solutions give the best results, and for this reason it is best to mix only enough coating material for the particular job in hand. Should it be necessary to store the fluid, plastic bottles, not glass, must be used. In no circumstances should the solution be retained for more than 30 days.

WARNING

Wear rubber gloves and respirator when working with "Alodine 1200S." Do not breathe fumes or dust from the powder, or vapor from the solution. If the powder or solution comes in contact with your skin, wash the affected area immediately with water.

b. The surface may be cleaned, the corrosion removed, and the conversion coating applied to the surface with the one solution, prepared as above, by applying the solution and abrading the surface while it is wet with the solution. Apply the solution evenly and liberally with a "Scotch Brite" pad to the surface of the sheet to be treated. Thoroughly wet the entire surface. Remove corrosion products, stains, and soil by lightly abrading the metal while it is completely covered by the solution with the "Scotch Brite" coated scouring pad. (Use "fine" or "ultra-fine" grade only.) Supplemental abrasive cleaning will be necessary in a very few instances. This cleaning must be followed by the Alodine 1200 conversion coating applied per paragraph 51.7.2.4.

CAUTION

Take care to keep the solution out of faying surfaces, inaccessible areas, and "hard-to-rinse" areas. Such areas may be masked off with zinc chromate putty. The solution should not be allowed to come into contact with cadmium plated parts. It is suggested that such parts should be temporarily removed.

c. Organic soils and dirt will reduce the conversion coating and cause it to turn green. The green solution may be used to continue the cleaning operation but must be removed and replaced with fresh solution (golden yellow) in order to get an adequate "take" on the metal. A cellulose sponge is useful for removing spent solution and for reapplying fresh solution. Due to the different degrees of cleanliness, and consequently different activity, of the metal being treated, the coating will not take uniformly. Some areas will deposit a heavy brown coating while other areas will receive no coating at all. Continue to treat those areas where no coating is forming, and at the same time continue to remove the heavy deposits by abrading the areas while covered with solution. When the entire panel is properly conditioned and a conversion coating is beginning to form on the entire panel, interrupt the abrading for about 30 to 45 seconds and allow the coating to form. This formation of the coating can be readily observed through the solution. Rinse off all excess solution immediately. Heavy brown deposits should be removed by applying fresh solution, abrading, and rinsing before the heavy deposit is reformed. Care should be taken to be sure that all excess coating solution is completely rinsed away. A water spray should be directed into faying surfaces, inaccessible areas, and "hard-to-rinse" areas. The activity of the Alodine 1200S may be slowed down to prevent powdering by diluting the solution with fresh water. This dilution may be done on the work itself.

d. Thoroughly dry all treated areas and apply one coat of TT-P-1757 zinc chromate primer.

e. Reinstall any cadmium plated parts that may have been removed and replace any parts that may have been removed for accessibility.

51.7.4 ANTI-CORROSION SEALING.

The fuselage is sealed in certain areas during manufacture to prevent corrosion. This sealing is shown in figure 51-22. If repairs to the fuselage result in removal of parts of this sealing system, they should be renewed on completion of repairs, in accordance with figure 51-22. Inhibited sealing compounds (PR-1422G and similar) should be used for this sealing.

51.7.5 REMOVAL OF FUEL TANK MICROBIOLOGICAL CONTAMINATION. This corrosion preventive procedure should be applied only to tanks that have suddenly shown an indication of microbiological contamination after regular checks over a long time period have shown them to be free of such contamination. Tanks that have developed severe or recurring contamination problems should be treated

51.7.5 (Cont).

as indicated in paragraph 51.7.3.1 Correction of minor microbiological contamination problems may be accomplished as follows:

Make a solution of kerosene and 0.5% (by volume) PFA-55MB (MIL-I-27686) additive.

With approximately 100 gallons of this mixture flush out each of the main tanks as much as possible through the probe holes, float valve access covers, and fuel pump access doors. Agitate and slosh flushing mixture in such a way as to break loose and remove all visible microbiological contamination. Drain the solution from the tank using care to see that all loose contamination is removed.

While the above is accomplished, the aircraft attitude should be adjusted to the best possible flushing and draining positions by adjustments in the landing gear strut pressure or by moving the aircraft on and off of chocks.

Observe normal fuel safety precautions. (Refer to Section 12.1.)

When the tanks are free of all visible microbiological contamination, strut pressures should be restored to normal.

After the above is accomplished, it is recommended that the tanks be filled with fuel containing 0.15% (by volume) of the PFA-55MB additive. The longer this fuel can be allowed to remain in the tank the better. Continued use of this additive is recommended until the microbiological activity disappears, perhaps 1 week more or less.

NOTE

△1 ▨ PREPARE SURFACES BY CLEANING WITH P-D-680 (TYPE II) DRY CLEANING SOLVENT, TT-N-95 (TYPE II) ALIPHATIC NAPHTHA, OR O-T-620 (TYPE I) 1-1-1 TRICHLOROETHANE SAFETY SOLVENT. (NOTE - EXCEPT FOR TRICHLOROETHANE, THESE SOLVENTS ARE FLAMMABLE.) DO NOT ALLOW THE SOLVENT TO EVAPORATE BEFORE WIPING WITH CLEAN, DRY CLOTHS. SEAL IN ACCORDANCE WITH THE APPLICABLE INSTRUCTIONS IN SECTION 51.2. FINISH WITH ONE COAT OF TT-P-1757 ZINC CHROMATE PRIMER AND TWO COATS OF STM 37-405 ACID RESISTANT VINYL LACQUER.

△2 ▩ PREPARE SURFACES BY CLEANING WITH P-D-680 (TYPE II) DRY CLEANING SOLVENT, TT-N-95 (TYPE II) ALIPHATIC NAPHTHA, OR O-T-620 (TYPE I) 1-1-1 TRICHLOROETHANE SAFETY SOLVENT. (NOTE - EXCEPT FOR TRICHLOROETHANE, THESE SOLVENTS ARE FLAMMABLE.) DO NOT ALLOW THE SOLVENT TO EVAPORATE BEFORE WIPING WITH CLEAN, DRY CLOTHS. APPLY ONE COAT OF MIL-C-8514 WASH PRIMER, ONE COAT OF TT-P-1757 ZINC CHROMATE PRIMER, AND TWO TOP COATS MIL-C-7439 (CLASS I) ELASTOMERIC COATING SYSTEM.

△3 ▒ FINISH THE AREA IN ACCORDANCE WITH THE INSTRUCTIONS IN SECTION 51.8. CLEAN THE FINISHED SURFACE WITH O-T-620 (TYPE I) 1-1-1 TRICHLOROETHANE AND APPLY ONE COAT OF SEALANT 'G' FIGURE 51-6. APPLY SEALANT TO ALL SURFACES INSIDE FUEL VALVE COMPARTMENT BELOW WL 75.5.

4. SEE FIGURE 51-24 FOR FINISH SUPPLY DATA AND FIGURE 51-6 FOR SEALING COMPOUNDS.

JS-1-2-X0/2-5122

Figure 51-22. Fuselage Anti-Corrosion Sealing

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SECTION 51.8

FINISHES

Contents Page

GENERAL 51-48
- Finish Identifications and Vendors 51-48
- Selection of Finishes for Interior and Exterior 51-48
- Finish Thicknesses 51-48
- Precautions for Magnesium Parts 51-48
- Interior Decorative Trim 51-48
- Cleaning Surfaces 51-48

BONDING 51-48
- Magnesium 51-48
- Non-Clad Aluminum Alloys 51-48
- Protection of Bonding Connections 51-48
- Bonding Tubing 51-48
- Bonding Requirements for Anti-Static Painted Plastic Surfaces 51-48

SPECIAL FINISH REQUIREMENTS 51-49
- High Temperature Areas 51-49
- Landing Gear 51-49
- Flap Wells 51-49
- Molded Plastic and Ceramics 51-49
- Other Plastic Parts 51-49
- Battery Area 51-49
- Protection from Synthetic Lubricants 51-49
- Rubber 51-49
- Thermoplastic Resin Parts 51-49
- Fairings and Fillets 51-49
- Galley and Lavatory Areas 51-49
- Protective Coating for Magnesium Components 51-49
- Fuel Vents and Jettison Outlets 51-50

EXTERIOR FINISHING 51-50
- Removal of Polyurethane Paint Coatings for Exterior Refinishing 51-50
- No-Paint Areas 51-50-1
- Repainting and Touch-Up of Existing Polyurethane Exterior Finish 51-50-1

- Radome Refinishing 51-50-1
- Radome Refurbishing 51-50-1
- Radome Renewal 51-50-1
- Radome Color Refinishing 51-50-1
- Removal of MIL-C-7439 Neoprene Coating from Radome 51-50-1
- Removal of Epoxy Anti-Static Coating from Radome 51-50-2
- Removal of Epoxy Anti-Static Coating with Decorative Topcoat from Radome 51-50-2
- Removal of MIL-C-83231, Type II Polyurethane Coating System from Radome 51-50-2
- Radome Surface Preparation 51-50-2
- Application of Epoxy Anti-Static Coating to Radome 51-50A
- Application of MIL-C-83231, Type II, Polyurethane Coating to Radome 51-50B
- Antenna Protection 51-50B
- Doppler Radome and HF Antenna 51-50B
- ADF Loop and Sense Antennas 51-50B
- Wing Tip Finishes 51-50B

FIGURES

- 51-23. Organic Finishes 51-51
- 51-24. Finish Supply Data 51-52
- 51-25. Finish Thicknesses 51-54
- 51-26. Protective Treatment for Firewalls and Interior Surfaces on Engine Side 51-54
- 51-27. Insulation Requirements for Magnesium Parts 51-55
- 51-28. Interior Decorative Trim Finishes 51-56A
- 51-29. Spring Loaded Test Prod 51-56B

51.8.1 GENERAL.

This section contains instructions for selection and application of surface finishes and for special treatment which is required for certain purposes and specific areas of the airplane.

51.8.1.1 FINISH IDENTIFICATIONS AND VENDORS. For the identifications of the recommended finishes and for information on the vendors from whom these materials can be obtained, see figures 51-23 and 51-24.

51.8.1.2 SELECTION OF FINISHES FOR INTERIOR AND EXTERIOR. The finishes recommended for use on the interior and exterior surfaces of the airplane are shown in figure 51-23.

51.8.1.3 FINISH THICKNESSES. To prevent the addition of unnecessary weight to the airplane, always hold the thicknesses of paint coatings to the minimum thicknesses specified in figure 51-25. This instruction applies to all paint coatings except those on areas where the maximum resistance to corrosion is required.

51.8.1.4 PRECAUTIONS FOR MAGNESIUM PARTS. Insulation between magnesium parts and parts made from certain other metals is essential, to preclude rapid corrosion. The various methods used to effect this insulation are shown in figure 51-27.

51.8.1.5 INTERIOR DECORATIVE TRIM. Most of the interior decorative trim in the airplane is of self-color material and does not need any additional finish. However, if a color change is required, or protection from possible abrasion, the necessary finish materials and methods are shown in figure 51-28.

51.8.1.6 CLEANING SURFACES. Before painting, clean surfaces with clean cotton cloths wet with solvents (figures 51-23 and 51-27) such as O-T-620 trichloroethane or P-D-680 standard solvent.

51.8.2 BONDING.

Items on which electrical bonding is to be accomplished must have paint and similar finishes or anodic film removed from the metal only in the area where the connection is to be made. If there is a long delay in making the final connection, normal surface oxidation can prevent an effective bond; the connection should be made within 7 days for all metals except magnesium, and within 12 hours for magnesium, otherwise it will be necessary to repeat the finish removal operation. Plating or chemical film protection need not be removed for bonding. If a clean, metal-to-metal contact is made between faying surfaces, it is unnecessary to provide an additional clean, metal-to-metal contact under the bolt or screw head. When a doubler or other riveted part is between two items to be bonded, provide a clean, metal-to-metal contact between faying surfaces so that one complete bonding path is provided.

51.8.2.1 MAGNESIUM. Where it is necessary to remove paint coatings from magnesium alloys to make a bonding connection, and the faying surface is of dissimilar metal, separate the magnesium from the dissimilar metal with a 0.016 aluminum alloy shim overlapping the bared area by ¼ inch. (See figure 51-28.)

51.8.2.2 NON-CLAD ALUMINUM ALLOYS. Where paint and other finishes are removed from non-clad aluminum alloys to make a bonding connection, treat the bare metal surfaces with chemical film protective treatment, applied by brush, before assembly, and within 12 hours after removal of the finish.

51.8.2.3 PROTECTION OF BONDING CONNECTIONS. After completing the bonding connection, except where magnesium is involved, apply the same paint finish to the bare surface as was removed to make the bond. This must be done within 7 days after removing the finish. For magnesium, treat the exposed area which has had the anodic film removed with Dow 19 touch-up compound, brushed on until a brassy appearance is obtained, then make the bonding connection. Within 12 hours after making the permanent connection, seal around the connection with Specification MIL-S-8802, Class B sealing compound, forming a fillet approximately ¼ inch wide around the connection.

51.8.2.4 BONDING TUBING. Remove paint and similar finishes or anodic film with No. 400 sandpaper at least halfway around the tube in the area where the bonding attachment is to be made. Do not do this more than 7 days before completing the connection. Plating or chemical film treatment need not be removed. When finishes are removed down to bare metal, apply chemical film protective treatment to aluminum alloy tubing before making the bonding connection, and not more than 12 hours after removing the finish in the bonding area. After the connection has been made, retouch the area around the connection with the same paint finish, if any, that is on the remainder of the tube.

51.8.2.5 BONDING REQUIREMENTS FOR ANTISTATIC PAINTED PLASTIC SURFACES. When antistatic painted plastic surfaces have been refinished per paragraph 51.8.4.3.2, 51.8.4.4, and 51.8.4.5, the electrical bonding shall meet the following requirements:

a. For the nose radome, the resistance from a point on the radome to a point on the adjacent metal skin shall be between 2.0 and 200.0 megohms.

b. For the ADF sense antennas, ADF loop antenna cover, and wing tips, the resistance from a point on the plastic part to a point on the adjacent metal skin shall be between 0.1 and 200.0 megohms.

The points selected for the above checks shall not be closer than 2.0 inches to the junction of non-metal to metal. If the anti-static paint has been painted over with a decorative finish, the probe of the measuring instrument must penetrate the decorative finish and make contact with the anti-static paint in order to obtain a valid reading.

Note

Refer to paragraph 51.8.4.3.3.2 for bonding checks on radomes after refinishing.

51.8.3 SPECIAL FINISH REQUIREMENTS.

Because of the environments to which they are exposed or the materials from which the surfaces are made, certain areas of the airplane require special finishes.

51.8.3.1 HIGH TEMPERATURE AREAS. For treatment of magnesium and non-stainless steel surfaces in areas where the maximum temperature is normally between 250° and 300°F, see figure 51-23. Use the appropriate color and gloss. For areas where temperatures are normally between 300° and 500°F, use either black, heat-resisting enamel to Specification MIL-E-5557, or aluminized, high temperature resistant paint to Specification TT-P-28 (see figure 51-24), applied over chemically surface-treated metal. For areas where temperatures are normally over 500°F, use TT-P-28 aluminized, high temperature resistant paint (see figure 51-24). Finish the exterior surfaces of the nacelles, on those airplanes having painted nacelles, with an epoxy primer /aliphatic polyurethane enamel top coat in the desired color, or the paint system used on other surfaces of the aircraft. (See figure 51-23). For treatment of firewalls and interior surfaces of the nacelles facing the engines, see figure 51-26.

51.8.3.2 LANDING GEAR. The landing gear is finished with one coat of LAC 37-722, Type I epoxy primer, one coat of LAC 37-722 Type III epoxy enamel, aluminized, and two coats of Corogard 14 aluminized. (See figure 51-24.)

51.8.3.3 FLAP WELLS. Flap well areas are finished with one coat of TT-P-1757 zinc chromate primer. (See figure 51-24.)

51.8.3.4 MOLDED PLASTIC AND CERAMICS. Do not paint transparent plastic parts. Other plastic parts, except fiberglass and similar laminates, antennas, and Doppler radomes, need not be painted except for color-matching purposes. Never paint plastic or ceramic electrical or electronic insulators.

51.8.3.5 OTHER PLASTIC PARTS. Where rain erosion-resistant and anti-static coatings are not required on plastic parts, wet-sand the area with toluene, and paint with LAC 37-722, Type II, III, or IV epoxy enamel for color-matching purposes. Other paints may be used for color coats, provided the plastic is first painted with LAC 37-722, Type I epoxy primer to assure adhesion. Finish all air conditioning supply and distribution ducts that are normally hidden from view and do not require matching of color schemes with one coat (0.6 to 0.8 mils thick) of LAC 37-722 Type IV epoxy enamel, black color #37038 (figure 51-24) after wet-sanding with Specification TT-T-548 toluene. Do not use filler-type materials on these ducts, either before or after painting.

51.8.3.6 BATTERY AREA. The area around and below the battery box is treated to protect the structure from leakage of the electrolyte. This treatment consists of one coat of MIL-C-8514 wash primer, one coat of TT-P-1757 zinc chromate primer, and two coats of cement primer and two mils of base coat conforming to MIL-C-7439 Class I. The coatings should be applied to extend at least 6 inches forward and aft of the battery box, and to either side of the box to a line at least 2 inches above the bottom surface of the box. (See figure 51-24.) (See paragraph 51.8.4.3.2.)

51.8.3.7 PROTECTION FROM SYNTHETIC LUBRICANTS. The synthetic lubricants used on the airplane, such as engine oil, are detrimental to most paint finishes. Surfaces subject to contact with these lubricants should be finished with LAC 37-722 Type I epoxy primer and/or LAC 37-722 Type II, III, or IV epoxy enamel, color as required (See figure 51-24).

51.8.3.8 RUBBER. Natural and synthetic rubber do not normally require to be painted. Refer to Chapter 12 for lubrication requirement for rubber seals subject to frequent opening. If rubber parts have to be painted, use rubber base paint of the required color (See figure 51-24).

51.8.3.9 THERMOPLASTIC RESIN PARTS. Parts made from "Duratrim", "Avtrim", "Royalite", and similar materials may be finished with vinyl lacquer (see figure 51-24).

51.8.3.10 FAIRINGS AND FILLETS. One coat of TT-P-1757 zinc chromate primer is applied to all inside surfaces of metal fairings and fillets and to the surfaces of skin and structure covered by the fairings or fillets, except for the wing surfaces covered by the external wing tanks, which are protected by two coats of TT-P-1757 zinc chromate primer. (See figure 51-24.)

51.8.3.11 GALLEY AND LAVATORY AREAS. Acid sealing is incorporated in the galley and lavatory areas between fuselage stations 470 and 570, as shown in figure 51-22. This sealing system confines spillage of corrosive fluids to the area. The sealant used is EC-1675 (Minnesota Mining and Manufacturing Co.) Class B, or PR-1472 (Products Research Co.), Class B, with a top coat of spray sealant EC-1675 (Minnesota Mining and Manufacturing Co.) over all fasteners. Except in the case of structural repairs in this area, this sealing system should not require attention. Finishing of the area, after sealing, is with one coat of TT-P-1757 zinc chromate primer and two coats of STM 37-405 acid resistant vinyl lacquer.

51.8.3.12 PROTECTIVE COATING FOR MAGNESIUM COMPONENTS.

a. REPAIR OF DRILLED HOLES. Perform the following procedure on all magnesium surfaces exposed by drilling. This procedure does not apply to abraded areas or areas exposed by machining.

   (1) Paint exposed magnesium area with a solution containing 8 ounces per gallon of potassium dichromate, ¼ ounce per gallon of Nacconal NR (or equivalent), and one gallon of water.

   (2) Wipe off excess with a damp cloth (do not rinse) and allow to dry.

b. REPAIR OF ABRADED OR MACHINED AREAS. Parts abraded in handling or on assembly or areas exposed due to machining may be repaired with DOW 19 touch-up. The touch-up shall be made by dipping, brushing, or spraying. The DOW 19 solution shall be controlled as follows:

chromic acid	1/3 ounce
calcium sulfate	one ounce
water	one gallon

51.8.3.12 (Cont).

The chemicals shall be added to the water in the order shown and stirred vigorously for a minimum of 15 minutes. The magnesium surface shall be kept wet with solution until a brassy film is produced (1 to 3 minutes). Following application, rinse with cold water and dry. Do not rinse with hot water.

51.8.3.13 FUEL VENTS AND JETTISON OUTLETS. Fuel tank vent and jettison outlets should be protected from painting and special care should be exercised to assure removal of all masking in these areas upon completion of painting prior to return of the aircraft to service.

51.8.4 EXTERIOR FINISHING.

The aircraft as delivered from the factory does not have a decorative paint system applied. Certain fiberglass areas are painted for protection or to minimize electrostatic discharges. Various acceptable decorative painting techniques are used in the field. However, the procedure that would be used if the airplane were painted at the Lockheed plant is as follows:

CAUTION

> Abrasive cleaning or sanding of metallic exterior surfaces of the airplane is prohibited. Failure to comply can cause structural damage, fatigue failure, stress corrosion or possible failure of a more serious nature. Refer to paragraph 59.8.1 for scratch repair.

a. Clean all metallic surfaces to be painted with an alkaline water base cleaner followed by a thorough water rinse.

b. Treat the surfaces with a phosphoric acid metal conditioner for prepaint conditioning followed by a thorough water rinse.

c. Apply a color chromate conversion coating treatment (i.e., Alodine 1200 or Iradite 14-2) to the surfaces followed by a thorough water rinse. Allow to dry.

d. Apply one coat of epoxy primer to a dry film thickness of approximately 0.4 to 0.7 mil.

e. Apply two coats of an aliphatic polyurethane enamel, color as desired by the customer.

51.8.4.1 REMOVAL OF POLYURETHANE PAINT COATINGS FOR EXTERIOR FINISHING.

a. Mask around all openings such as doors, escape hatches, ports, control surface wells, wheel wells, vents, speed brake, or other openings that would admit processing solutions. Cover all surfaces such as tires, acrylic plastics, glass windows, neoprene coated parts, and surfaces not being repainted that could be damaged by processing solutions. Mask metal bonded assemblies, honeycomb panels and sealants exterior to the airplane to prevent damage by processing solutions, solvents and strippers. Mask all joints, seams and fasteners in the areas being stripped to prevent paint strippers from entering or becoming entrapped in joints or faying surfaces.

CAUTION

> To prevent damage do not allow strippers to come in contact with sealant, adhesives in metal bonded assemblies, adhesives or edge sealing materials of honeycomb panels.

b. Apply Turco Products 5873 stripper to areas to be stripped. Handle and apply stripper in accordance with manufacturers instructions. At facilities where disposal of chemicals does not present a problem, Turco Products 5351 may be used. If Turco 5351 is used, all chemicals, residue and other wastes must be collected and disposed of properly. Whichever stripper is used, extreme care shall be exercised during all paint stripping operations to assure removal of all stripper, stripper residue and degraded paint from the airplane.

c. Treat all metal exterior skin surfaces to be painted with MIL-C-38334, Type I, metal conditioner reduced one-to-one by volume with water.

MIL-C-38334, compound, metal conditioner
Intex Products, Inc., Intex 8019
Turco Products, Metal Glo No. 6
Pennwalt Chemical Co., Pennwalt 715K

The mixed solution shall be applied to the surfaces and agitated. Do not allow the solution to remain on the surface more than 12 minutes, then rinse with water. It is recommended that, if possible, the airplane should be pressurized for several minutes after stripping and cleaning.

d. Apply color conversion coating material per MIL-C-81706 to all treated surfaces.

MIL-C-81706, coating, color conversion
Sealants, Adhesives & Coatings, Inc.
SAC 5541 - colored

Prepare and apply solution in accordance with manufacturers instructions. Apply color conversion coating immediately after conditioning to areas no larger than those which can be rinsed immediately upon development of the required color. Do not allow solution to dry on parts. Rinse thoroughly with water and allow to dry. The film shall be continuous and free from breaks, scratches, powdery areas, and

51.8.4.1 (Cont).

other damage which will reduce its serviceability. Any evidence of powdering shall be removed with clean rags wet with water or isopropyl alcohol per TT-I-735/MIL-F-5566 and then dried thoroughly. When the surface is rubbed with a dampened cloth, the cloth shall show no signs of powdery residue.

51.8.4.2 NO-PAINT AREAS. The following areas should not be painted:

a. Conductor strips located on wing tips.

b. Thrust reversers. (Stainless steel and titanium)

c. 1.0-inch areas adjacent to all deicer boots including the deicer boots.

d. Empennage trim operating range (Stainless steel covers) located at bottom of vertical stabilizer adjacent to empennage fairing.

e. 1.0- to 1.5-inch areas around static ports.

51.8.4.2.1 REPAINTING AND TOUCH-UP OF EXISTING POLYURETHANE EXTERIOR FINISH. It is permissible to use sandpaper to rough up old paint for better adherence of new paint; however, penetration through the paint is not permitted.

51.8.4.3 RADOME REFINISHING. The finish on the nose radome on the aircraft delivered from the factory consists of a neoprene rain erosion resistant and anti-static coating system conforming to MIL-C-7439, Class 2. (See figure 51-22.) The radome may be refinished with the MIL-C-7439 Class 2 system (paragraph 51.8.4.3.2) or completely refinished with a MIL-C-83231 Type II polyurethane rain erosion resistant and anti-static coating system in accordance with the manufacturer's instructions (paragraph 51.8.4.3.5.2) or with a decorative color paint system (paragraph 51.8.4.3.3) consisting of an epoxy anti-static coating (STM 37-510 Type I) followed by a decorative nonmetallic colored polyurethane coating.

51.8.4.3.1 RADOME REFURBISHING. To refurbish the appearance of the radome finished with the MIL-C-7439, Class 2. neoprene system, moisten a clean cotton rag with aliphatic naphtha, and wipe surface two or three times. Polish with Simonize paste wax.

51.8.4.3.2 RADOME RENEWAL. To completely renew the radome, as it was finished at the factory, refer to paragraph 51.8.4.3.3.1. Remove coating 1.0 inch in width on the adjacent metal, and clean the exposed metal surface with TT-T-548 toluene or TT-M-261 methylethylketone. Refer to paragraph 51.8.4.3.4 for surface preparation. Apply one to three coats of cement primer (United Shoe Machinery Co. Bostic 1007) to a thickness of 0.001 to 0.002 inch over entire surface of radome, and on the adjacent metal. Allow 5 minutes air-drying time between coats and 20 to 30 minutes between the last coat of cement primer and the first coat of neoprene rain erosion resistant coating.

Note

The neoprene and anti-static coatings are supplied as a Goodyear Tire & Rubber Co. kit No. 23-57S, or a Gates Engineering Co. kit N-79.

Apply the required coats of black neoprene rain erosion-resistant coating and anti-static coating in accordance with the manufacturer's instructions to obtain the following thicknesses:

a. The neoprene coating, including the cement primer, should be 0.007 to 0.01 inch thick.

b. The anti-static coating should be 0.001 to 0.002 inch thick.

Apply the rain erosion-resistant coating to the radome only. Apply the anti-static coating on the radome and on the adjacent 1.0 inch of the cement primer coated metal surface at the same time so as to provide continuity between the fiberglass radome and the metal structure.

51.8.4.3.3 RADOME COLOR REFINISHING. To refinish the radome in a decorative color, strip the neoprene and adjacent metal as described below.

Note

The colored radome coating will not provide as high a degree of anti-static and rain erosion protection as the neoprene-coated radome. The installation of neoprene or plastic rain erosion boots on the radome will reduce the transmissibility sufficiently to degrade the performance of the weather radar.

51.8.4.3.3.1 REMOVAL OF MIL-C-7439 NEOPRENE COATING FOR RADOME.

Remove the neoprene coating by covering the coating with felt pads or rags and saturate with methylethylketone or toluene or a mixture (50-50) thereof. Maintain coverings in a saturated condition until the coating has loosened to the extent that it can be peeled off or scraped off with a hand scraper made of phenolic or plastic. If a scraper is used, avoid scratching the radome surface. Remove residue with cloth dampened with toluene or methylethylketone.

Note

As alternates, there are two paint removers which may be used in lieu of the solvent removal method cited above. These products are:

Intex Products, Inc.   #8581 Remover
P.O. Box 6648
Greenville, S.C. 29606

Gates Engineering 100 South West Street Wilmington, Delaware	GACO S-14 Remover

The remover should be allowed to remain on the surface until the coating has loosened as noted above, then scraped if necessary, followed by solvent cleaning.

51.8.4.3.3.2 REMOVAL OF EPOXY ANTI-STATIC COATING FROM RADOME.

Removal of epoxy anti-static coating must be performed by abrading with 180 grit or finer abrasive paper to a point where the laminate substrate starts to show through all over the radome. **Extreme care must be exercised to ensure the fibers in the laminate are not exposed or damaged.**

The sanding dust should be removed and the surfaces solvent cleaned with clean cotton cloths wet with methylethylketone prior to refinishing.

51.8.4.3.3.3 REMOVAL OF EPOXY ANTI-STATIC COATING WITH DECORATIVE TOPCOAT FROM RADOME.

Removal of decorative topcoat from epoxy anti-static coating prior to re-decorative topcoating must be performed by abrading with 180 grit or finer abrasive paper to a point where the black epoxy anti-static coating starts to show through all over the radome. **Extreme care must be exercised to ensure the epoxy anti-static coating and the fibers in the laminate are not damaged.**

The sanding dust should be removed and the surfaces solvent cleaned with clean cotton cloths wet with methylethylketone before overcoating with the decorative finish.

51.8.4.3.3.4 REMOVAL OF MIL-C-83231, TYPE II POLYURETHANE COATING SYSTEM FROM RADOME.

Removal of the polyurethane coating system may be accomplished by applying MIL-R-25134 paint stripper to the surface. Reapplications of the stripper should be made to keep the surface wet for a maximum of 30 minutes while removing the loosened coating with phenolic or plastic scrapers. After 30 minutes, thoroughly clean the stripper away with methylethylketone and clean cotton cloths.

The cleaned surface should be then scuff sanded, cleaned, and smoothed as specified below.

51.8.4.3.4 RADOME SURFACE PREPARATION.

The cleaned surface should be scuff sanded with 180 grit or finer abrasive paper, the sanding dust removed and the surfaces solvent cleaned with methylethylkeytone and clean cotton cloths.

Surfaces of the laminates which have depressions, voids, or other surface irregularities should be smoothed by applying a smoothing putty over the surface followed by resanding back to original contour.

Note

The following epoxy type filler materials are suggested:

Andrew Brown of Koppers 801 East Lee Street Irving, TX 75060 Phone 214-445-0666	P-900 Filler
Sikkens Aerospace Finishes 20846 South Normandie Ave. Torrance, CA 90502 Phone 213-320-6800	No. 467-9 Filler
De Soto, Inc. Forest Lane and Shiloh Rd. Garland, Texas 75040 Phone 214-276-5181	No. 521x316/ 910x664

These products should be prepared and applied in accordance with the manufacturer's instructions.

51.8.4.3.4 (Cont)

The filled surfaces should be dry sanded to the original contour of the laminate surface followed by wet sanding with 400 grit abrasive paper. Alternately, the excess filler may be removed with abrasive pads. If the latter method is used, it is preferable to remove the excess filler before it is completely hardened.

The smoothed surface should be examined for voids or other surface irregularities and, if necessary, should be refilled and resanded.

The smoothed surface must be cleaned. Clean with lint-free cloth or equal.

51.8.4.3.4.1 APPLICATION OF EPOXY ANTI-STATIC COATING TO RADOME.

a. Coating Preparation - Anti-static epoxy base component must be mixed with a catalyst-thinner as supplied by the vendor. Partial mixing of kits is prohibited. The liquid material must be thoroughly agitated before, while, and after adding the catalyst-thinner, before pouring the mixed material from one container to another and before each application. After mixing, the material must be allowed to stand a minimum of one hour before use and must be used within 24 hours after being mixed.

Note

It is extremely important that all pigment is dispersed, and the material agitated before pouring from one container to another and before each application.

Note

Anti-static coating material currently approved is available from:

Andrew Brown of Koppers 801 East Lee Street Irving, TX 75060 Phone 214-445-0666	XA147A - glossy epoxy anti-static coating XA148A - lusterless epoxy anti-static coating
Sterling Lacquer Mfg. Co. 3150 Brannon Ave. St. Louis, Mo 63139 Phone 314-776-4450	#22-2686-lusterless epoxy anti-static coating

It is recommended that the Koppers XA148A or the Sterling 22-2686 material be used in all cases except where a decorative topcoat is not to be applied and a high gloss black finish is desired.

b. Coating Application

A light wet coat should be applied around all edges or masked areas. After 5 minutes dry time, two full, wet, cross coats of the anti-static coating must be applied to the required surface with a 15 to 25 minute dry time between each cross coat. The average dry film thickness must be 0.0016 to 0.0020 inch.

Note

The pigments must be thoroughly dispersed immediately prior to spraying to provide required anti-static properties in the cured coating. All necessary care must be exercised to prevent application of a coating without properly dispersed components. Use a pressure or a siphon type cup spray gun. Agitate the cup gun frequently during spray application; placing one or two marbles in the cup will help keep the pigment dispersed during the agitation of the cup.

c. Cure - The final coat must be allowed to air dry 1 to 1-1/4 hours, then force cured at 125 (±5)°F, for 24 hours (±30 minutes) or cured at any time and temperature cycle, below 160°F, that will produce a coating meeting the requirements of this specification without damaging the substrate. Usually, curing at 160°F for 8 hours is satisfactory.

d. Surface Resistivity - The applied and cured anti-static coating shall have a resistivity of 2 to 200 megohms. If low resistivity readings are encountered, the epoxy coating must be removed down to the plastic laminate surface and a new coating applied. A trace of the first epoxy coat remaining on the laminate is permissible. If high resistivity readings are obtained, the epoxy coat may be lightly abraded, and a full wet cross coat of the coating applied and cured; or the radome may be re-baked at a higher temperature for a longer period of time. If the resistivity reading is still out of specification limits after curing, the coating must be removed and replaced.

Note

Factors which affect the surface resistivity values of anti-static coatings are:

- The thicker the coating - the lower the resistivity.
- The higher the temperature cure - the lower the resistivity.
- The longer the cure time - the lower the resistivity.

Conversely: The thinner the coating - the higher the resistivity.

The lower the cure temperature - the higher the resistivity.

The shorter the cure time - the higher the resistivity.

51.8.4.3.4.1 (Cont)

e. Surface Resistivity Testing (See figure 51-27.) - Surface resistivity must be determined on all parts. Measurements must be taken when the cured parts are at a temperature of between 70 and 85°F, and must be satisfactory before application of any decorative topcoat. The surface resistivity is measured by holding two 3/8-inch diameter wire spring loaded electrodes (test prods as shown in figure 51-27, or equal) having a pressure capacity of 3 to 5 pounds, placed 6 inches apart on the coated surface. At least 6 readings should be made on the coated surface of the part being tested, and each reading should meet the surface resistivity requirements. A 500-volt megohm bridge must be used to determine the surface resistivity readings.

Note

One 500 volt megohm bridge is:

Standco Megohmer Model B-6 available from:

Herman H. Sticht Company
New York, New York

f. Surface Appearance - The cured anti-static coating, whether topcoated or not, should be continuous and be free from bubbles, cracks, blisters; and should be smooth, free from dirt, grit, or foreign matter. Rack marks on unexposed or fay surface areas are acceptable so long as the coating satisfies the other requirements.

g. Decorative Topcoats - Anti-static epoxy requiring a decorative topcoat may be overcoated with a maximum dry film thickness of 0.002 inch, and color and material as required. Pigmented coatings containing metallic particles must not be used.

The epoxy anti-static coating, when required to be topcoated, must be scuff sanded with 280 grit or finer abrasive paper after satisfactory surface resistivity readings are obtained. Sanding dust must be removed prior to topcoating by wiping with methylethylketone and clean cotton cloths.

51.8.4.3.4.2 APPLICATION OF MIL-C-83231, TYPE II, POLYURETHANE COATING TO RADOME.

a. Coating System Preparation

The coating system consists of a vinyl primer, a polyurethane base coat and a polyurethane (black) anti-static topcoat. These three coatings should be prepared as specified in the manufacturer's instructions enclosed in each coating system kit.

The coating system materials are available from:

CAAP Co., Inc. P.O. Box 2066 Huntington, Connecticut 06484 Phone 203-877-0375	(MIL-C-83231, Type 2, Class A kit)
Goodyear Aerospace Corp. P.O. Box 548 Goodyear Street Rockmart, Ga. 30153 Phone 404-684-7855	(MIL-C-83231, Type 2, Class B kit)

b. Coating System Application

The three coatings (primer, base coat, and anti-static topcoat) should be applied in accordance with the manufacturer's instructions and to the coating thicknesses specified therein. Spray application is recommended.

c. Surface Resistivity

The testing of this coating is very sensitive and is generally done only on test panels, not on the radome.

51.8.4.4 ANTENNA PROTECTION.

51.8.4.4.1 DOPPLER RADOME AND HF ANTENNA. The Doppler radar window and HF antenna, when installed, are painted with a non-anti-static top coat. These areas, when repainted, must be coated with a nonmetallic epoxy finish.

51.8.4.4.2 ADF LOOP AND SENSE ANTENNAS. The ADF loop and sense antenna housing is finished with two coats of anti-static epoxy (semiflat black), lapped around and under all edges of the fiberglass to insure a good conductive path to the metal airplane structure. After curing, a 2.0-mil top coat of fluid resistant, gray epoxy is added. If a color change or repainting is required, carefully wet-sand the color off by lightly scuff-sanding the surface with No. 320 or 400 grit sandpaper, being careful not to sand through the paint finish to the primer system. To completely refinish the surface down to the basic material, wet-sand off the existing paint system, and finish in accordance with the instructions for a colored radome (paragraph 51.8.4.3.3). To check continuity, refer to paragraph 51.8.2.5.

51.8.4.5 WING TIP FINISH. In order to control and minimize electrostatic discharges as affecting the communications systems, the wing tips are finished with an undercoat of black anti-static epoxy paint and decoratively finished with gray epoxy enamel top coat. To refinish the top coat or to make a color change, refinish the same as for the ADF loop and sense antenna housing. To completely refinish, follow the instructions listed under ADF antenna housing (paragraph 51.8.4.4.2). To check continuity, refer to paragraph 51.8.2.5.

MATERIAL	PROTECTIVE SYSTEM △2	
	EXTERIOR	INTERIOR
	FINISH	FINISH
ALUMINUM ALLOYS 1100, 3003, 5052, 6053, 6061, CLAD 2014, CLAD 2024, CLAD 7075, CLAD 7178.	NOTHING (EXCEPT WHEN TO BE PAINTED. SEE △1). △3	1 COAT TT-P-1757 ZINC CHROMATE PRIMER
BARE 2024, 7075, 7178, OTHER BARE HIGH STRENGTH ALUMINUM ALLOYS, AND METALS NOT OTHERWISE LISTED.	MUST BE PAINTED △1	1 COAT TT-P-1757 ZINC CHROMATE PRIMER
CADMIUM AND ZINC PLATED PARTS.	NOTHING (EXCEPT WHEN TO BE PAINTED. SEE △1).	1 COAT TT-P-1757 ZINC CHROMATE PRIMER
TITANIUM, CORROSION AND HEAT-RESISTANT ALLOYS AND CHROMIUM PLATE	NOTHING △3	NOTHING
MAGNESIUM ALLOYS	MIL-C-8514 WASH PRIMER. 2 COATS OF EPOXY PRIMER LAC 37-722, TYPE I AND 1 COAT OF EPOXY ENAMEL LAC37-722, TYPE III OR △1.	MIL-C-8514 WASH PRIMER, 3 COATS OF LAC37-722 TYPE I EPOXY PRIMER
ALL METALS NOT COVERED BY THE ABOVE	MUST BE PAINTED △1	1 COAT TT-P-1757 ZINC CHROMATE PRIMER
EXTERIOR SURFACES REQUIRING FINISHES OTHER THAN SMALL MARKINGS.	△1	
UNPAINTED SURFACES REQUIRING SMALL MARKINGS	ONE COAT STM37-306 STENCIL ENAMEL	SAME AS EXTERIOR
STEEL-HEAT TREATED ABOVE 220, 000 PSI	ONE COAT LAC 37-722 TYPE I EPOXY PRIMER AND ONE COAT LAC37-722 TYPE III EPOXY ENAMEL △4 ; IN WHEEL WELLS, APPLY ADDITIONAL TWO COATS OF COROGARD 14 △4	NOTHING IF METAL COATED; SAME AS EXTERIOR IF **UNPAINTED**

NOTE

△1 USE AN EPOXY PRIMER/ALIPHATIC POLYURETHANE ENAMEL SYSTEM.
USE AN ACID WASH PRIMER ON UNPAINTED METAL OPTIONAL. (SEE 51.8.4)

△2 PRIOR TO TOUCH UP PAINTING, SURFACES MUST BE CLEAN. USE CLEAN COTTON RAGS WET WITH O-T-620 TRICHLOROETHANE, P.D. 680 STANDARD SOLVENT OR NAPHTHA, FOLLOWED BY DRYING WITH CLEAN DRY COTTON CLOTHS. SCUFF SAND AND/OR SOLVENT ACTIVATE AS REQUIRED.

△3 USE △1 PAINT SYSTEM FOR LARGE PAINT APPLIED MARKINGS ON UNPAINTED SURFACES.

△4 FINISH SYSTEM △1 MAY BE USED FOR ALTERNATE

Figure 51-23. Organic Finishes

MATERIAL	SPECIFICATION	VENDORS
COMPOUND, METAL CONDITIONER	MIL-C-38334	INTEX PRODUCTS INC., INTEX 8019 TURCO PRODUCTS, METAL GLO NO. 6 PENNWALT CHEM CO. PENNWALT 715K
EPOXY PRIMER	LAC 37-722 TYPE I	KOPPERS CO., INC. P-908
EPOXY ENAMEL, TYPE II (SEMI GLOSS), TYPE III (HIGH GLOSS), TYPE IV (LOW GLOSS)	LAC 37-722 TYPE II TYPE III TYPE IV	KOPPERS CO., INC. NO. A-316
COATING NEOPRENE RAIN EROSION RESISTANT	MIL-C-7439 CLASS I	GATES ENGINEERING CO. GACO N-79 GOODYEAR TIRE AND RUBBER CO. 23-56S
COATING NEOPRENE RAIN EROSION RESISTANT AND ANTI-STATIC	MIL-C-7439 CLASS II	GATES ENGINEERING CO., NO. N-810 BLACK GOODYEAR TIRE AND RUBBER CO. NO. 23-57S, (IN KITS CONTAINING 1 PINT OR 1 QUART OF COATING).
COATING; WALKWAY	MIL-W-5044 TYPES I & II	AKRON PAINT AND VARNISH COMPANY 3M CO.
COMPOUND; GREASE, SILICONE	COMMERCIAL	DOW CORNING NO. 3 DOW CORNING NO. 4
COMPOUND; SEALING	MIL-S-8802	PRODUCTS RESEARCH CO. "PR-1422-B-4" COAST PRESEAL AND MFG CO. "890-B-4"
ALUMINIZED ABRASION RESISTANT COATING	COMMERCIAL	3M CO. COROGARD 14 ALUMINIZED GENERAL SEALANTS CO. GS-140
ENAMEL; HEAT RESISTING, BLACK	MIL-E-5557 TYPE I	KOPPERS CO., INC. GLIDDEN COMPANY SHERWIN-WILLIAMS COMPANY
ENAMEL; STENCIL	STM-37-306	ADVANCE COATING AND CHEMICAL CO. UNIFORM STENCIL ENAMEL WITH 9H2 HARDENER, DEXTER MIDLAND CORP LAMINAR X 500 STENCIL ENAMEL
COATING SYSTEM, POLYURETHANE RAIN EROSION RESISTANT AND ANTI STATIC	MIL-C-83231 TYPE 2	GOODYEAR TIRE AND RUBBER NO. 23-71 STERLING LACQUER ASTROCOAT 7200
LACQUER; ACID-RESISTANT VINYL	STM37-405	KOPPERS CO., INC. W.P. FULLER
LACQUER; NITROCELLULOSE, CAMOUFLAGE	TT-L-20	KOPPERS CO., INC. RANDOLPH PRODUCTS
LACQUER; NITROCELLULOSE, GLOSS	TT-L-32	KOPPERS CO., INC. E.I. DUPONT
OIL; LINSEED	TT-O-369	USE ANY COMMERCIAL RAW LINSEED OIL
PAINT; ALUMINIZED, HIGH TEMPERATURE RESISTANT	TT-P-28	ATLAS PAINT AND VARNISH CO. SENTRY PAINT AND CHEMICAL CO.

Figure 51-24. Finish Supply Data (Sheet 1 of 2)

MATERIAL	SPECIFICATION	VENDORS
PAINT; RUBBER BASE	COMMERCIAL	GATES ENGR CO. H-10
PRIMER; PRE-TREATMENT WASH	MIL-C-8514	KOPPERS CO., INC. GLIDDEN COMPANY
PRIMER; ZINC CHROMATE	TT-P-1757 COLOR NO. 34151	KOPPERS CO., INC. "SN704C" E.I. DU PONT "1063-174" GLIDDEN COMPANY "RGL 29974" MINNESOTA PAINTS "13498"
FILLER, EPOXY PLASTIC LAMINATE	COMMERCIAL	KOPPERS CO., INC. BOSTIK-FINCH 467-9
TAPE; PLASTIC, POLYESTER	LAC 24-791	MINNESOTA MINING AND MFG. CO.
VARNISH; ELECTRICAL INSULATING	MIL-V-1137 TYPE M	ACME WIRE COMPANY STANDARD T CHEMICAL COMPANY
VARNISH; SPAR, GLYCERYL PHTHALATE	MIL-V-6894 CLASS 1 & 3	E.I. DU PONT GLIDDEN COMPANY W.P. FULLER COMPANY
COATING, EPOXY ANTI-STATIC TYPE III	STM 37-510 TYPE III	BOSTIC-FINCH 463-6-14
COATING, EPOXY ANTI-STATIC, TYPE I	STM37-510 TYPE I	STERLING LACQUER 22-2686
COMPOUND, STRIPPING	MIL-R-25134	WYANDOTTE CHEMICAL CORP. NO. 1632 McGEAN CHEMICAL CO. NO. A-29-C PUREX CORP., TURCO PRODUCTS DIV., NO. 4260B PENNSALT NO. 19A
COATING SYSTEM, ALIPHATIC POLYURETHANE, WITH WASH PRIMER AND/OR EPOXY PRIMER	COMMERCIAL	U.S. PAINT ALUMIGRIP SYSTEM BOSTIK FINCH 600 SERIES SYSTEM DESOTO SUPER DESOTHANE SYSTEM [1]
COATING, WALKWAY, ROUGH TEXTURE	MIL-W-5044 TYPE II	AKRON PAINT AND VARNISH CO. 3M CO.
CORROSION PROTECTIVE COATING, FUEL TANKS	MIL-C-27725, TYPE II, CLASS B	PRODUCTS RESEARCH PR-1560MC, PR-1560MK, PR-1563 DESOTO CHEMICAL 823-730
COLOR CONVERSION COATING	MIL-C-81706	SEALANTS, ADHESIVES AND COATINGS, INC., SAC 5541-COLORED

[1] SUGGESTED SOURCES - OTHER SYSTEMS MAY BE SATISFACTORY.

Figure 51-24. Finish Supply Data (Sheet 2 of 2)

COATING	SINGLE COAT, DRY FILM THICKNESS
MIL-C-8514 WASH PRIMER	0.2 TO 0.3 MIL (.0002 TO .0003 INCH)
TT-P-1757 ZINC CHROMATE PRIMER	0.3 TO 0.4 MIL (.0003 TO .0004 INCH)
LAC 37-722 TYPE I EPOXY PRIMER	0.3 TO 0.5 MIL (.0003 TO .0005 INCH)
TT-L-20N INTEROCELLOLOSE LACQUER AND STM37-405 VINYL LACQUER	0.4 TO 0.5 MIL (.0004 TO .0005 INCH)
LAC 37-722, TYPE II, III & IV EPOXY ENAMELS	0.6 TO 0.8 MIL (.0006 TO .0008 INCH)
WALKWAY COATING	2.5 TO 5.0 MILS (.0025 TO .005 INCH)
COROGARD 14	1.5 TO 2.0 MILS (.0015 TO .002 INCH)
ALUMINIZED HIGH TEMPERATURE RESISTANT PAINT	1.0 TO 1.5 MILS (.001 TO .0015 INCH)

Figure 51-25. Finish Thicknesses

MATERIAL	FINISH
TITANIUM, CORROSION-RESISTANT AND HEAT-RESISTANT ALLOYS, PLATED METAL, HOT-DIP-COATED METAL OR SPRAYMETALLIZED	NOT REQUIRED
TUBULAR STEEL NOT COVERED ABOVE	2 COATS TT-P-1757 ZINC CHROMATE PRIMER
1100, 3003, 5052, 6061, CLAD 2024, CLAD 7075, CLAD 7178	NOT REQUIRED
ALL OTHER MATERIALS NOT LISTED ABOVE.	1 COAT TT-P-1757 ZINC CHROMATE PRIMER

Figure 51-26. Protective Treatment for Firewalls and Interior Surfaces on Engine Side

CATEGORY OF PART OR SURFACE REQUIRING INSULATION FROM MAGNESIUM	METHODS				
	STEP I	STEP II	STEP III	STEP IV	STEP V
	LAC 37-722 **TYPE I EPOXY PRIMER** (ON NON MAGNESIUM SURFACES)	MIL-S-8802 SEALER	SILICONE GREASE COMPOUND (MANDATORY ON ALL REMOVABLE SURFACES - OPTIONAL ON OTHERS)	LAC 24-791 PLASTIC POLYESTER TAPE	5052 OR 5056 ALUMINUM ALLOY
FAYING SURFACES, JOINTS & SEAMS	TWO COATS (TOTAL) DRY △1	ONE COAT ON MAGNESIUM △2 △8	ONE COAT ON NON-MAGNESIUM	NONE	FOIL △8
REMOVABLE PINS	ONE COAT - DRY	NONE	ONE COAT	NONE	NONE
BOLTS & LOCKBOLTS	ONE COAT - WET OR DRY	NONE	NONE	NONE	WASHER
RIVETS △3	ONE COAT - WET OR DRY	NONE	NONE	NONE	NONE
SCREWS	ONE COAT - WET OR DRY	NONE	NONE	NONE	WASHER
STAKED BEARINGS	ONE COAT - WET	NONE	NONE	NONE	NONE
HIGH TEMPERATURE FAYING SURFACES	NONE	NONE	NONE	ONE LAYER △4	NONE
NUTS & LOCKBOLT COLLARS	ONE COAT - WET OR DRY	NONE	NONE	NONE	WASHER
CLOSE TOLERANCE BOLTS, PINS	ONE COAT - WET	NONE	NONE △5	NONE	WASHER
WASHERS △3	ONE COAT - WET OR DRY	NONE	NONE	NONE	NONE
SLIP AND PRESS FITS	ONE COAT - WET	NONE	NONE △5	NONE	NONE
ELECTRICAL BOND	NONE	NONE	NONE	NONE	△6 .016" SHIM
NAMEPLATES, INSTRUMENT & ELECTRONIC EQUIP △7	NONE	NONE	NONE	NONE	NONE
FASTENING DEVICES ATTACHED BY RIVETS	TWO COATS - DRY △1	NONE	NONE	ONE LAYER △4	NONE

Figure 51-27. Insulation Requirements for Magnesium Parts (Sheet 1 of 2)

NOTE

1 ZINC CHROMATE PRIMER MAY BE USED AS AN ALTERNATE ON NON-MAGNESIUM SURFACE.

2 THE SEALING COMPOUND MUST BE SQUEEZED OUT AT ALL BOUNDARIES LEAVING A CONTINUOUS FILLET.

3 ALUMINUM ALLOYS 5056 AND 5052 ("B" RIVETS AND ALUMINUM WASHERS) REQUIRE NO PAINT PROTECTION.

4 FOR APPLICATIONS OF 275°F AND 300°F. TAPE IS TO EXTEND AT LEAST 1/4" BEYOND THE CONTACTING BOUNDARIES.

5 NORMALLY, ASSEMBLE SLIP FITS AND CLOSE TOLERANCE REMOVAL BOLTS AND PINS WITH WET LAC37-722, TYPE I EPOXY PRIMER.IN INSTALLATIONS REQUIRING REMOVAL OF BOLTS AND PINS, MAKE ASSEMBLY WITH SILICONE GREASE COMPOUND.

6 WHERE IT IS NECESSARY FOR THE PURPOSE OF AN ELECTRICAL BOND TO OMIT OR REMOVE PAINT COATINGS FROM MAGNESIUM ALLOYS IN CONTACT WITH DISSIMILAR METALS, REFER TO PARAGRAPH 51.8.2.

7 CONTACTS SUCH AS THOSE BETWEEN NAMEPLATES, INSTRUMENTS, AND ELECTRONIC EQUIPMENT, ETC., AND PAINTED MAGNESIUM ALLOYS ON INTERIOR SURFACES OF THE AIRFRAME NEED NOT BE PAINTED.

8 WHERE IT IS IMPRACTICAL TO USE SEALING COMPOUND (MAGNESIUM TO A STAINLESS STEEL FIREWALL) USE FOIL PRIMED WITH LAC 37-722, TYPE I EPOXY PRIMER FOR INSTALLATION.

Figure 51-27. Insulation Requirements for Magnesium Parts (Sheet 2 of 2)

SURFACE TO BE PAINTED	FINISH SYSTEM
TT-P-1757 PRIMER ON METAL △1	STM37-405 ACID RESISTANT VINYL LACQUER
GLASS FIBER REINFORCED PLASTIC AND REINFORCED PHENOLICS △2	LAC37-722 TYPE II, III OR IV EPOXY ENAMEL, OR STM37-307 ACID RESISTANT VINYL LACQUER
RUBBER	RUBBER BASE PAINT
THERMOPLASTICS - "AVTRIM", "DURATRIM" AND "ROYALITE" △1	STM37-405 ACID RESISTANT VINYL LACQUER
EPOXY PRIMER, LAC 37-722, TYPE I, ON MAGNESIUM △1	LAC37-722 TYPE II, III OR IV EPODY ENAMEL.
CORROSION RESISTANT STEEL, CADMIUM & ZINC PLATED SURFACES △3	MIL-C-8514 ACID WASH PRIMER TT-P-1757; ZINC CHROMATE PRIMER, STM 37-405 ACID RESISTANT VINYL LACQUER

NOTE

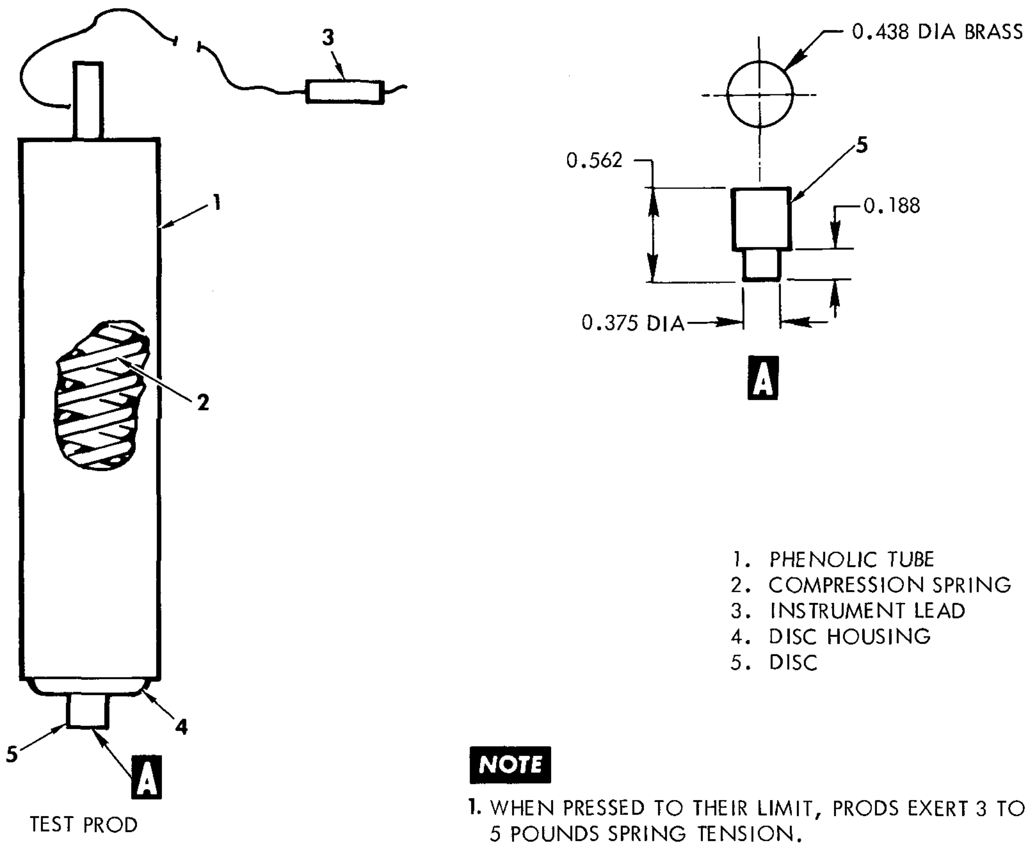
△1 CLEAN AND/OR REACTIVATE SURFACE WITH CLEAN COTTON RAGS MOISTENED WITH TT-T-266 LACQUER THINNER OR TT-T-548 TOLUENE.

△2 WET SAND WITH SPECIFICATION TT-T-548 TOLUENE AND WIPE DRY WITH CLEAN COTTON RAGS.

△3 CLEAN CORROSION RESISTANT STEEL WITH KELITE NO. L-17 OR TURCO NO. 2168 CLEANER.

4. LISTED BELOW ARE THE COLOR NUMBERS FOR LACQUER TOUCH UP PAINT THAT CAN BE USED IN THE FACTORY PAINTED GREY COCKPITS. ALL COLOR NUMBERS ARE PER THE FEDERAL STANDARD NUMBER 595. BLACK NO. 27038, LIGHT GREY NO. 36492, MED. GREY NO. 36293, DARK GREY NO. 36118.

Figure 51-28. Interior Decorative Trim Finishes



1. PHENOLIC TUBE
2. COMPRESSION SPRING
3. INSTRUMENT LEAD
4. DISC HOUSING
5. DISC

NOTE

1. WHEN PRESSED TO THEIR LIMIT, PRODS EXERT 3 TO 5 POUNDS SPRING TENSION.

JS-1-2-X0/2-5129

Figure 51-29. Spring Loaded Test Prod

SECTION 51.9

DEPLETED URANIUM COUNTERWEIGHTS

Contents Page

GENERAL 51-57
FINISHING 51-57
SAFETY PRECAUTIONS 51-57
REPAIR PROCEDURE 51-57

51.9.1 GENERAL.

Eleven depleted uranium counterweights are used on the flight control surfaces of the JetStar. Two are at the outboard end of each aileron; one is near the top of the rudder; two are near the bottom of the rudder torque tube; and two are on each elevator, one at each outboard end and one inboard on each of the torque tubes. Depleted uranium is one and a half times as dense as lead and is the heaviest naturally occurring metal.

51.9.2 FINISHING.

The counterweights have been flash-coated with copper followed by a nickle coat, then plated with cadmium to a minimum thickness of 0.001 inch and painted for protection against corrosion.

51.9.3 SAFETY PRECAUTIONS.

WARNING

Unauthorized Alterations Prohibited. Do not machine, grind, weld, sand, drill, or burn the counterweights. Removal of metal or corrosion by any method is prohibited.

Due to depleted uranium having the highest density (greatest mass within the smallest volume) of any metal, personnel under-estimate the weight of a counterweight and can cause strains or finger and foot injuries.

After any aircraft mishap where counterweights have been damaged, burned, or have lost their cadmium plating coating, counterweights should be covered with paper (packing material), then enclosed in plastic. Personnel should wear gloves, industrial eye protection, and respirator mask to protect against radioactive dust particle ingestion.

Depleted uranium metal will oxidize if the cadmium plating is damaged. The oxide is toxic because it is a heavy metal and has an equivalent toxicity to that of lead.

Gloves, wrapping material, wiping cloths, respirator filters or other articles used in the handling of damaged counterweights should be appropriately labeled as radioactive waste and disposed of accordingly.

51.9.4 REPAIR PROCEDURE.

Uranium counterweights found to be damaged during the repair of proximate parts or during routine inspection will have the damaged area sprayed with one coat of all purpose enamel paint immediately after the damage is noticed. No other type of repair should be made.

51.9.4.1 ADDITIONAL SUPPORT.

STARMET CMI, BARNWELL, SC 29812
Repair of depleted uranium counterweights
Telephone: (803) 259-3622
Fax: (803) 259-3622

Use and/or disclosure is governed by the statement on the title page of this document.

SECTION 51.10
AIRFRAME LONG TERM STORAGE

Contents Page

GENERAL 51-59

PRESERVATION OF THE AIRCRAFT STRUCTURE 51-59
  Aircraft Exterior 51-59
  External Openings 51-59
  Radome and Other Plastic Exterior Parts 51-59
  Pitot and Static Ports 51-59
  Rubber Seals 51-60
  Windows 51-60
  Windshield Wiper 51-60
  Water 51-60

MAINTENANCE OF PRESERVATION, AIRCRAFT STRUCTURE 51-60
  Aircraft Exterior 51-60
  Aircraft Interior 51-60
  Neoprene Coated Radome and Other Neoprene Coated Plastic 51-60

DEPRESERVATION, AIRCRAFT STRUCTURE 51-60
  Aircraft Exterior 51-60
  Aircraft Interior 51-60
  Neoprene Coated Radome and Other Neoprene Coated Plastics 51-60
  Rubber Seals 51-60
  Windshield Wipers 51-60

51.10.1 GENERAL. Paragraph 51.10.2 covers the requirements for aircraft storage during periods of inactivity. This section shows these requirements for the aircraft structure.

51.10.2 PRESERVATION OF THE AIRCRAFT STRUCTURE.

51.10.2.1 AIRCRAFT EXTERIOR. The aircraft should be cleaned in accordance with Section 12.14. Painted areas should be inspected for condition and repainted as required or protection applied where paint is missing or defective. Corrosion preventive compound, solvent cutback, cold application (MIL-C-16173 Grade 4, or equal) should be applied to all unpainted aluminum exterior areas. This compound may be diluted with "140°F Solvent" (P-D-680 Type II or equal) as necessary to suit application (spray, brush, or swab). Materials adversely affected by this compound, such as tires, deicer boots, windows, radomes, rubber, and plastic should be protected by masking. Aircraft stored indoors shall not require application of corrosion preventive compound to exterior aluminum surfaces for storage periods which do not exceed 60 days. External or exposed ferrous and plated working or bearing surfaces not specifically covered elsewhere in this manual shall be cleaned with "140°F Solvent" and then protected with an application of corrosion preventive compound (MIL-C-16173 Grade 3, MIL-C-11796 Class 3, or equal).

51.10.2.2 EXTERNAL OPENINGS. When outside inactivity exceeds or will exceed 30 days all openings located in positions that will allow entrance of water, dust or other foreign matter shall be covered with standard covers or with greaseproof paper (MIL-B-121 or similar) held in place with waterproof tape (Permacel 691, Polyken 231, or similar). Area where tape is to be applied should be precleaned with aliphatic naphtha or trichloroethane. Do not cover drain holes in bottom of fuselage but check to see that all of these are open and free of obstruction.

51.10.2.3 RADOME AND OTHER PLASTIC EXTERIOR PARTS. For outdoor storage of one week or longer neoprene (black or dark brown) coated radome or other plastic exterior parts should be coated with one coat of wax (MIL-W-18723, or equal). The painted aluminum skins within 12 inches of neoprene coatings should also be waxed. Allow wax to dry. Acrylic parts (transparent) should not be waxed but coated as indicated in Paragraph 51.10.2.6.

Note

Paragraph 51.10.2.3 does not apply to deicing boots. Refer to paragraph 51.10.2.5.

51.10.2.4 PITOT AND STATIC PORTS. Tape pitot and static port openings using grease proof paper and water proof tape. This includes the pitot mounted on each nacelle.

51.10.2.5 RUBBER SEALS. On aircraft stored 30 days or more all door and hatch rubber seals should be cleaned with a cloth moistened (not soaked) in aliphatic naphtha or trichloroethane, then rewipe with clean dry cloth to remove all grease, oil and soil. It is important to remove any hydraulic fluid as this is especially detrimental to rubber seals. After cleaning, the exposed sides of neoprene (black) seals should be coated with fluorocarbon dry film lubricant (Tiolon A-20, Miller Stephenson Chem MS-122, or similar). No coating is required for silicone (gray, white, or red) seals or for wing and empennage deicing boots. Doors and hatches with cleaned and coated seals should remain closed during periods of inactivity except as may be required by 51.10.3.2. below.

51.10.2.6 WINDOWS. Spray outside of acrylic windows (cabin windows and crew compartment skylight) plus an extra one-half inch all around with two heavy coats of strippable water emulsion protective coating (Spraylat SC-1090 or equal). No protective coating is required on inside of these windows or on any of the glass windows.

51.10.2.7 WINDSHIELD WIPER. Windshield wiper blades and arms will be removed from the aircraft during periods of long term inactivity.

51.10.2.8 WATER. Drain all water from galley, lavatory and any other location. Assurance should be obtained that water fire extinguishers are provided with the proper antifreeze (Reference: Section 25.2).

51.10.3 MAINTENANCE OF PRESERVATION, AIRCRAFT STRUCTURE.

51.10.3.1 AIRCRAFT EXTERIOR. Regular inspections of aircraft exterior will be made to assure good condition of protective coatings, covers, etc., applied by 51.10.2. Any defects will be corrected and additional coatings, covers, or corrosion preventive compounds will be applied as necessary to maintain adequate protection.

51.10.3.2 AIRCRAFT INTERIOR. Aircraft stored outdoors should be ventilated twice a week to prevent condensation and mildew. Results may permit a later extension of the frequency of this period, especially in dry climates. Forced warm air may be required to keep interior humidity down for some storage conditions. Ventilation should also be used to prevent interior temperature from exceeding the recommended maximum of 150°F. Check periodically for the appearance of mildew and corrosion. If found, corrections should be made as required.

An inspection should be made following rainy weather or heavy rainstorms for leakage. Any leakage found should be corrected.

Interior inspections on aircraft stored indoors may be accomplished only as considered necessary.

51.10.3.3 NEOPRENE COATED RADOME AND OTHER NEOPRENE COATED PLASTICS. Inspect adjacent unpainted aluminum skin waxed in accordance with Paragraph 51.10.2.3, weekly for any evidence of corrosion from leaching of acid from neoprene coating. Clean off wax with aliphatic naphtha applied with cloths, treat corrosion and replace wax as required.

51.10.4 DEPRESERVATION, AIRCRAFT STRUCTURE.

51.10.4.1 AIRCRAFT EXTERIOR. Clean exterior of aircraft in accordance with Section 12.14. Remove protective coating and covers. Clean acrylic windows (cabin windows and crew compartment skylight) with Morror Bright Polish Co. MGH-10 mirror glaze or Plexiglas cleaner. Inspect entire exterior for any evidence of corrosion, especially exposed machined parts. Touch up painted surfaces as required. Refer to Section 51.8.

51.10.4.2 AIRCRAFT INTERIOR. Inspect for leaks, mildew and corrosion. Clean and repair as necessary. Refer to Sections 51.2. and 51.7.

51.10.4.3 NEOPRENE COATED RADOME AND OTHER NEOPRENE COATED PLASTICS. Remove wax applied to neoprene coated parts with TT-N-95 aliphatic naphtha applied with cloths and wipe clean and dry.

51.10.4.4 RUBBER SEALS. Check condition of rubber seals. Clean as required using same cleaning methods indicated in Paragraph 51.10.2.5.

51.10.4.5 WINDSHIELD WIPERS. Install windshield wipers removed by Paragraph 51.10.2.7.

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CHAPTER 52
DOORS

TABLE OF CONTENTS

Section		Page
52.0	ENTRANCE DOOR	52-3
52.1	EMERGENCY ESCAPE HATCH	52-5
52.2	NOSE CONE	52-7
52.3	ACCESS AND INSPECTION PROVISIONS	52-9
52.4	ENTRANCE STAIRS	52-33
52.5	LANDING GEAR DOORS	52-49

SECTION 52.0
ENTRANCE DOOR

Contents Page

MAIN ENTRANCE DOOR 52-3

MAIN ENTRANCE DOOR WARNING SYSTEM 52-3
  Main Entrance Door Warning Switch 52-3
  Switch Adjustment 52-3

SEAL REPAIR 52-3

FIGURES
  52-1. Main Entrance Door (Airplanes 5001 thru 5065) 52-4
  52-2. Main Entrance Door Warning System Diagram 52-4
  52-2A. Removal and Replacement of Main Entrance Door (Airplanes 5066 and Up) 52-4A
  52-2B. Main Entrance Door, Helical Torsion Spring Replacement (Airplanes 5066 and Up) 52-4B
  52-2C. Main Entrance Door, Torsion Spring Replacement (Airplanes 5066 and Up) 52-4C

52.0.1 MAIN ENTRANCE DOOR. (See figure 52-1.) The entrance door, on the left side of the fuselage at the forward end of the cabin opens by moving inboard and sliding aft. A tubular-shaped rubber extrusion is attached to the door and forms a continuous seal. Holes are spaced along the seal to permit pressurizing air to enter the seal and expand it against the striker during pressurized flight to effect a tighter seal. The seal fits in an aluminum alloy seal retainer fixed to the door.

Note

The seal is installed with the holes facing the pressurized area (inboard).

The door can be opened from inside or outside the airplane. The outside handle is at the bottom of the door, where it can be reached from the ground, while the inside handle is at the center of the door. When open, the door is stowed in a compartment aft of the opening, together with the retracted stair.

52.0.2 MAIN ENTRANCE DOOR WARNING SYSTEM. (See figure 52-2.) A module on the annunicator panel goes on when the main entrance door is not closed and locked, and goes off when the door has been closed and locked. Power for this circuit is received from the 28 VDC essential bus, through the warning lights circuit breaker on the pilot's side circuit breaker panel.

52.0.2.1 MAIN ENTRANCE DOOR WARNING SWITCH. The warning switch is actuated by the door latching bayonet when the door is closed and locked.

52.0.2.2 SWITCH ADJUSTMENT. Adjust the main entrance door warning switch as follows: With the door and handle in the closed position and power on, using the switch adjusting/ retaining, position the switch until the warning light is extinguished. Continue adjusting in the "light out" direction an additional 3 turns, or approximately one eighth of an inch for overtravel. Tighten securely and safety-wire.

52.0.3 SEAL REPAIR. Should the door seal come loose from the seal retainer without damage to the seal, it may be bonded back as follows:

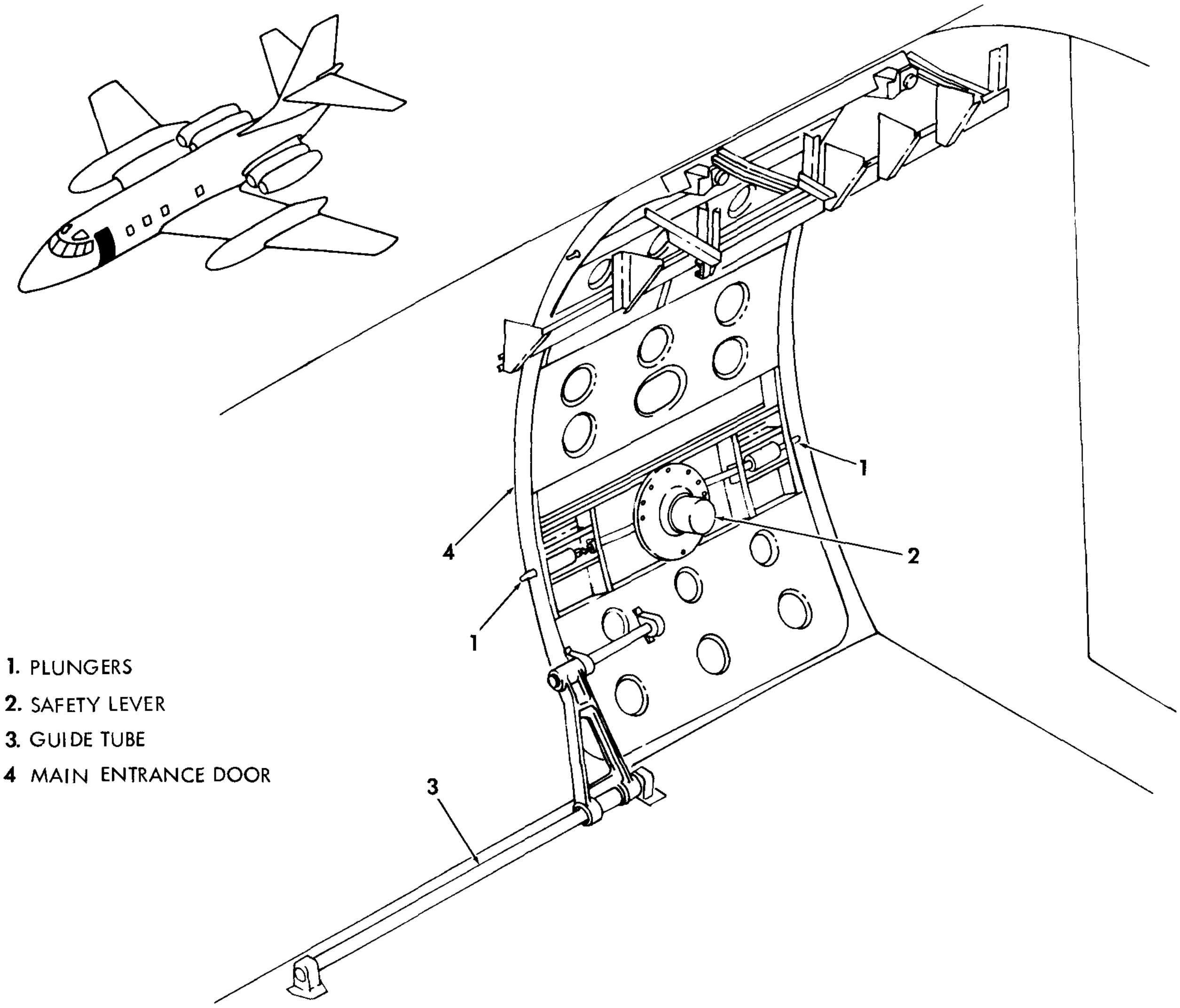
WARNING

Exercise extreme care when using aliphatic naphtha, this solvent is highly flammable.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

Clean surface to be bonded with aliphatic naphtha or trichloroethane. Rough up rubber surface with suitable abrasive and clean surface again as above. Apply an even coat of adhesive (Goodrich A-1082-B, 3M EC1300L or United Shoe Machinery Bostik 1142.) to all surfaces of the joint. Allow the adhesive to dry until it will adhere but not transfer to touch. Surfaces should then be joined with sufficient pressure to insure complete contact. Care shall be taken to eliminate air entrapment within the bond. Clean excess adhesive with methylethylketone. Initial strength will be sufficient for immediate use. Maximum strength will be obtained in 4 days at 70°F.



1. PLUNGERS
2. SAFETY LEVER
3. GUIDE TUBE
4 MAIN ENTRANCE DOOR

Figure 52-1. Main Entrance Door (Airplanes 5001 thru 5065)

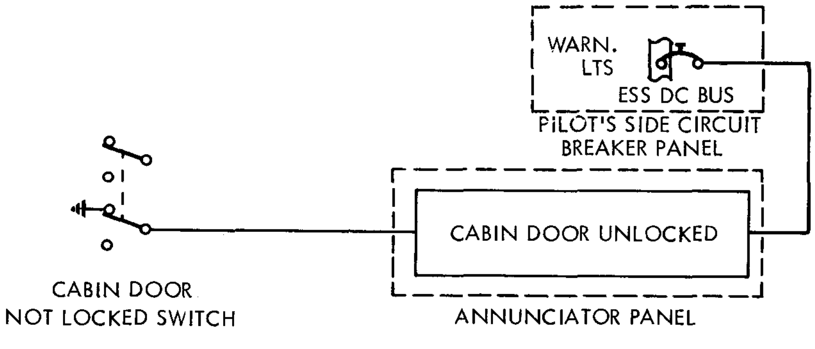
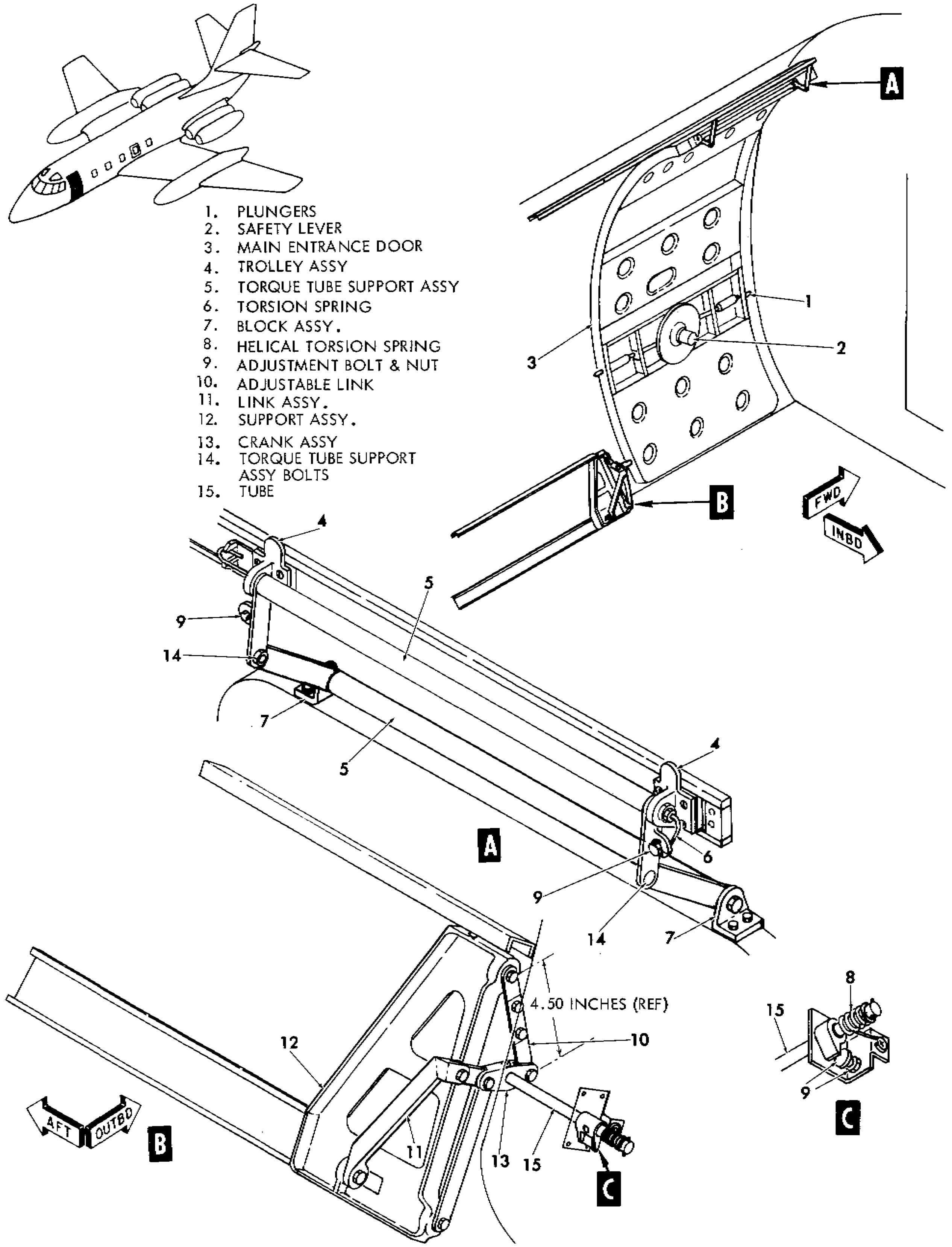

WARN.
LTS
ESS DC BUS
PILOT'S SIDE CIRCUIT BREAKER PANEL
CABIN DOOR UNLOCKED
ANNUNCIATOR PANEL
CABIN DOOR NOT LOCKED SWITCH

Figure 52-2. Main Entrance Door Warning System Diagram



1. PLUNGERS
2. SAFETY LEVER
3. MAIN ENTRANCE DOOR
4. TROLLEY ASSY
5. TORQUE TUBE SUPPORT ASSY
6. TORSION SPRING
7. BLOCK ASSY.
8. HELICAL TORSION SPRING
9. ADJUSTMENT BOLT & NUT
10. ADJUSTABLE LINK
11. LINK ASSY.
12. SUPPORT ASSY.
13. CRANK ASSY
14. TORQUE TUBE SUPPORT ASSY BOLTS
15. TUBE

Figure 52-2A. Removal and Replacement of Main Entrance Door (Airplanes 5066 and Up)

TO REMOVE THE ENTRANCE DOOR FROM THE AIRPLANE ACCOMPLISH THE FOLLOWING:

STEP 1

OPEN THE DOOR TO THE FULL INBOARD FORWARD POSITION AND PLACE A STRIP OF FOAM POLYURETHANE (OR EQUAL) BETWEEN THE FLOOR AND THE BOTTOM OF THE DOOR TO PROTECT THE DOOR PRESSURE SEAL.

STEP 2

REMOVE THE SCREW COMMON TO CRANK ASSEMBLY (13) AND TUBE (15).

STEP 3

FIRST REMOVE THE FORWARD BOLT COMMON TO THE TORQUE TUBE SUPPORT ASSEMBLIES (5) AND THEN THE REAR BOLT.

NOTE

LIFT ON THE FORWARD SIDE OF THE DOOR AND THE FORWARD BOLT CAN BE EASILY REMOVED (OR INSTALLED).

STEP 4

PUSH THE SUPPORT ASSEMBLY (12) AFT TO DISENGAGE THE TUBE (15) FROM THE CRANK ASSEMBLY (13).

STEP 5

PULL THE TOP OF THE DOOR INBOARD AND ROTATE THE REAR OF THE DOOR UP. THEN THE DOOR CAN BE EASILY REMOVED BOTTOM FIRST THROUGH THE DOOR OPENING.

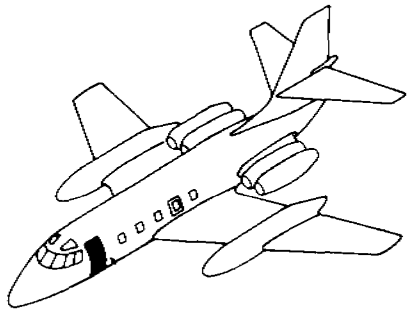
TO INSTALL THE ENTRANCE DOOR REVERSE THE STEPS FOR REMOVAL.

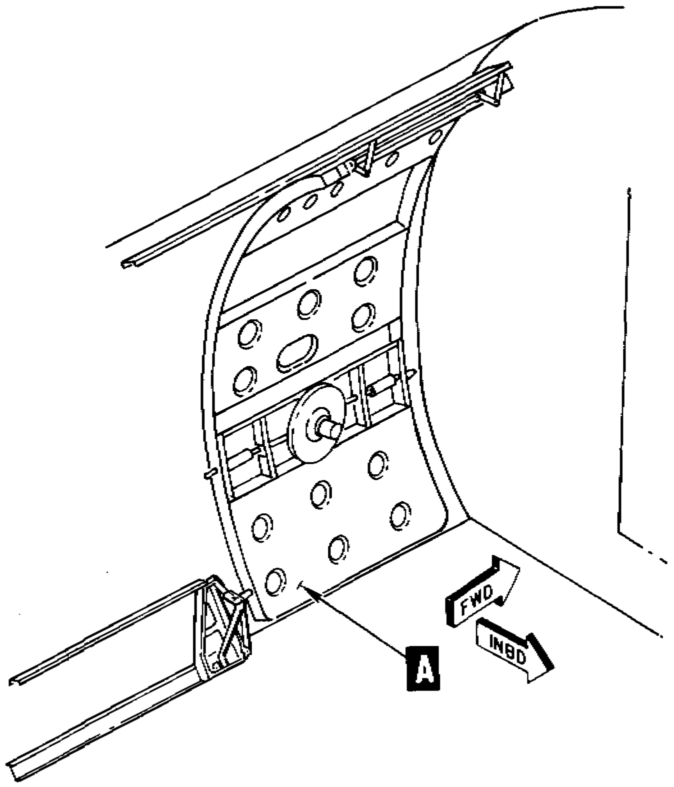
NOTE

IN STEP 3, INSTALL THE REAR BOLT FIRST. THE FRONT BOLT CAN BE EASILY INSTALLED BY LIFTING ON THE FRONT OF THE DOOR UNTIL THE HOLES IN THE TORQUE TUBE ASSEMBLY ARE ALIGNED.

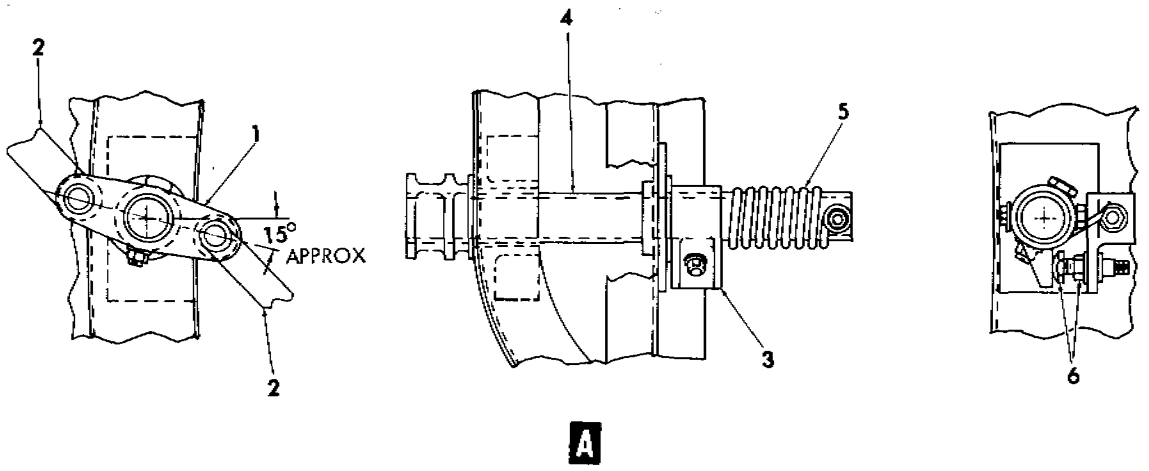
TO ADJUST THE INBOARD POSITION OF DOOR IN THE OPEN POSITION:

POSITION ADJUSTMENT BOLT (9) UNTIL THE DOOR IS IN THE POSITION DESIRED TO CLEAN STRUCTURE OR AUTOMATIC STAIR GEAR BOX ASSEMBLY.





1. CRANK ASSY
2. LINKS
3. STOP
4. TUBE
5. HELICAL TORSION SPRING
6. ADJUSTMENT BOLT AND NUT



TO REPLACE AND SET THE HELICAL TORSION SPRING (5) FOR MAXIMUM EFFECT, ACCOMPLISH THE FOLLOWING:

STEP 1

OPEN THE DOOR TO THE FULL INBOARD FORWARD POSITION AND INSERT A STRIP OF FOAM POLYURETHANE (OR EQUAL) BETWEEN THE FLOOR AND THE BOTTOM OF THE DOOR TO PROTECT THE DOOR PRESSURE SEAL.

STEP 2

REMOVE THE BOLTS COMMON TO THE CRANK ASSEMBLY (1) AND LINKS (2).

STEP 3

REMOVE THE BOLT COMMON TO THE STOP (3) AND TUBE (4) AND THE BOLT COMMON TO SPRING (5) AND TUBE (4).

STEP 4

TURN THE CRANK ASSEMBLY (1) 180° CLOCKWISE FROM THE POSITION SHOWN AND REINSTALL THE BOLT COMMON TO THE SPRING (5) AND THE TUBE (4) IN THE NEAREST HOLE.

NOTE

SIX EQUALLY SPACED HOLES ARE PROVIDED IN THE END OF THE TUBE FOR SPRING ADJUSTMENT.

STEP 5

TURN THE CRANK 180° COUNTER-CLOCKWISE (OR MORE AND INSTALL THE BOLT COMMON TO THE STOP (3) AND TUBE (4).

STEP 6

ASCERTAIN THAT THE HEIGHT OF THE ADJUSTMENT BOLT (5) DOES NOT EXCEED 0.52 INCHES AND THEN REINSTALL THE LINKS (2) TO THE CRANK ASSEMBLY (1).

STEP 7

CLOSE THE DOOR AND INSPECT THE SPRING (5). THE SPRING SHOULD NOT EXCESSIVELY BIND THE TUBE (4) DURING THE CLOSING OR OPENING CYCLE.

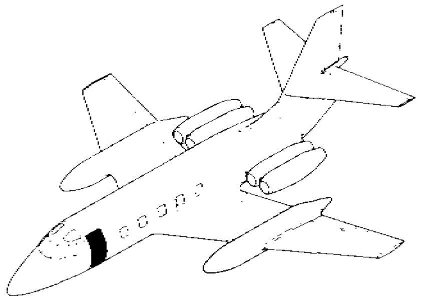
NOTE

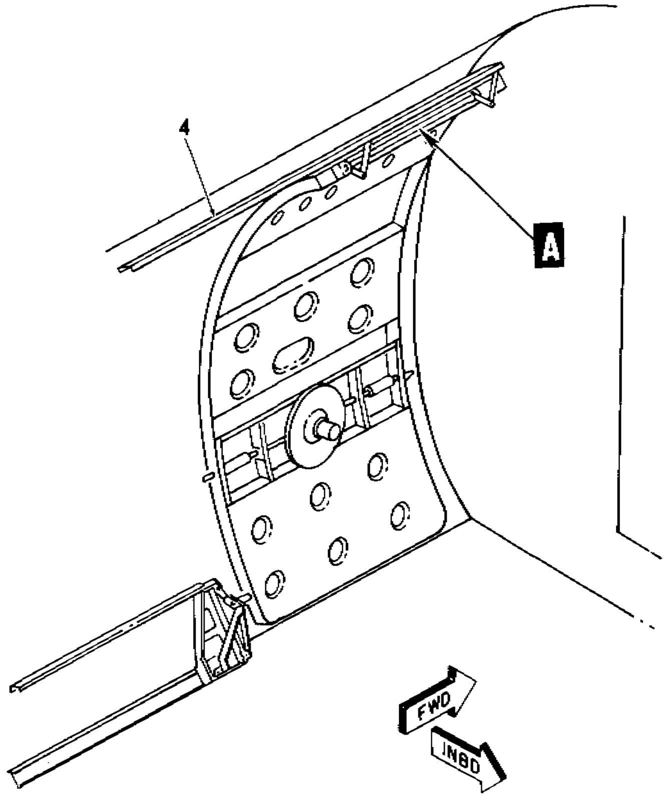
IF THE SPRING (5) IS A LOOSE FIT ON THE TUBE (4) GO THRU STEPS 1 THRU 5 AND SET THE SPRING AS TIGHT AS IT WILL OPERATE WITHOUT EXCESSIVELY BINDING THE TUBE WHEN THE DOOR OPENS AND CLOSES.

STEP 8

SET ADJUSTMENT BOLT (6) AS REQUIRED TO MAINTAIN DOOR CLEARANCE BETWEEN STRUCTURE OR AUTOMATIC STAIR GEAR BOX ASSEMBLY.

Figure 52-2B. Main Entrance Door, Helical Torsion Spring Replacement (Airplanes 5066 and Up)





1. TORSION SPRING
2. TORQUE TUBE SUPPORT ASSY
3. TROLLY ASSYS
4. UPPER DOOR TRACK
5. SPRING CLIP
6. ADJUSTMENT NUT AND WASHER

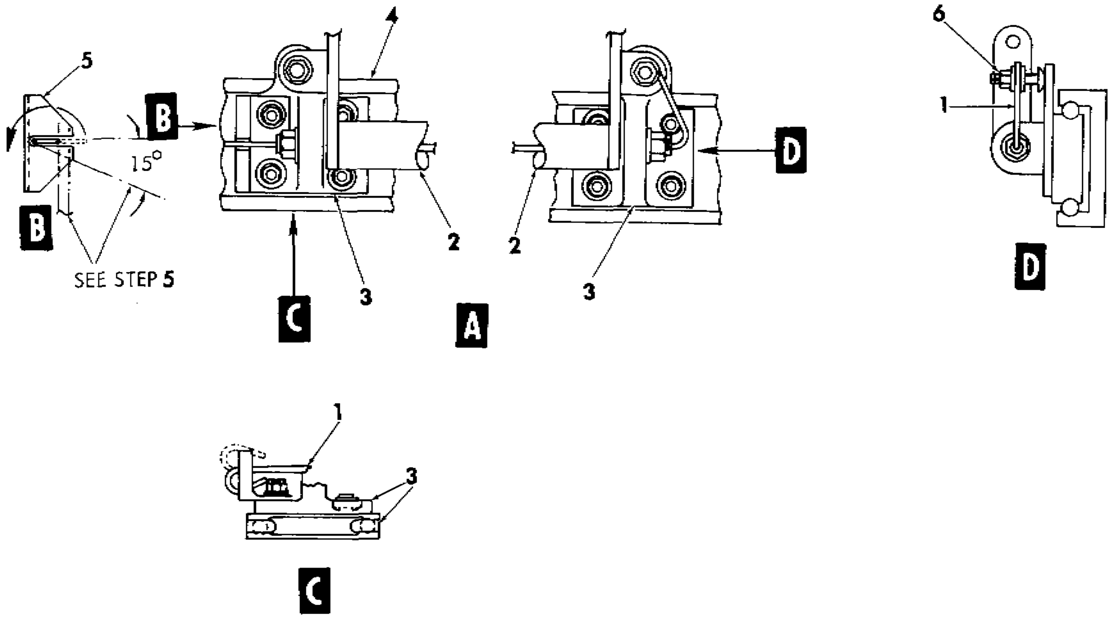


Figure 52-2C. Main Entrance Door, Torsion Spring Replacement (Airplanes 5066 and Up)

TO REPLACE AND INSTALL THE TORSION SPRING (1) FOR MAXIMUM EFFECTIVENESS, ACCOMPLISH THE FOLLOWING:

STEP 1
REMOVE THE UPPER DOOR TRACK (4), TROLLEY ASSEMBLIES (3) AND UPPER TORQUE TUBE SUPPORT ASSEMBLY (2) AS AN ASSEMBLY FROM THE AIRPLANE.

STEP 2
ALLOW THE TORQUE TUBE SUPPORT ASSEMBLY (2) TO ROTATE TO THE POSITION SHOWN IN DETAIL A.

STEP 3
LOOSEN THE SCREWS HOLDING THE SPRING CLIP (5) TO THE TROLLEY ASSEMBLY.

NOTE
DO NOT REMOVE THE TROLLEY ASSEMBLIES (3) FROM THE TRACK (4) DURING THIS OPERATION. PULL THE TROLLEY ASSEMBLY JUST FAR ENOUGH FROM THE END OF THE TRACK TO HOLD THE SCREWS WHILE UNTORQUING THE NUTS HOLDING THE SPRING CLIP TO THE TROLLEY ASSEMBLY. USE TAPE TO HOLD THE BALL BEARINGS IN PLACE ON THE TROLLEY ASSEMBLY.

STEP 4
CUT THE OLD SPRING FORWARD OF THE LOOP POSITION AND REMOVE. INSERT THE NEW SPRING IN THE TORQUE TUBE SUPPORT ASSEMBLY (2) AND INSTALL THE LOOPED END UNDER THE ADJUSTMENT NUT AND WASHER (6) AS SHOWN.

STEP 5
BEND THE LOOP ON THE END OF THE SPRING AS FOLLOWS:
(1) INSTALL SPRING CLIP (5).
(2) CLAMP A 3/16 DIAMETER STEEL ROD ON THE SPRING CLIP AS SHOWN.
(3) BEND UP THE END OF THE SPRING AS FAR AS POSSIBLE IN THE DIRECTION SHOWN, THEN ROTATE OVER SLOT AND CONTINUE BEND USING VISE GRIP PLIERS. CUT OFF EXCESS LENGTH AS SHOWN.

STEP 6
REMOVE CLAMP AND ROD AND LOOSEN SCREWS HOLDING THE SPRING CLIP.

STEP 7
WITH THE SPRING CLIP REMOVED, ROTATE THE LOOPED END OF THE SPRING 180° AS SHOWN AND REINSTALL THE SPRING CLIP.

NOTE
THE USE OF VISE GRIP PLIERS IS RECOMMENDED FOR THE ABOVE OPERATION.

STEP 8
REINSTALL THE TRACK (4), TROLLEY ASSEMBLIES (3) AND TORQUE TUBE SUPPORT ASSEMBLY (2) AS AN ASSEMBLY IN THE AIRPLANE.

STEP 9
CONNECT THE TORQUE TUBE ASSEMBLY (2) TO THE DOOR (SEE PREVIOUS INSTALLATION INSTRUCTIONS).

STEP 10
WITH THE DOORS IN FULL OPEN POSITION, THE TORSION SPRING (1) SHOULD HOLD THE UPPER TORQUE TUBE SUPPORT ASSEMBLY AGAINST THE STOPS ON THE TROLLEY ASSEMBLIES (3).

SECTION 52.1
EMERGENCY ESCAPE HATCH

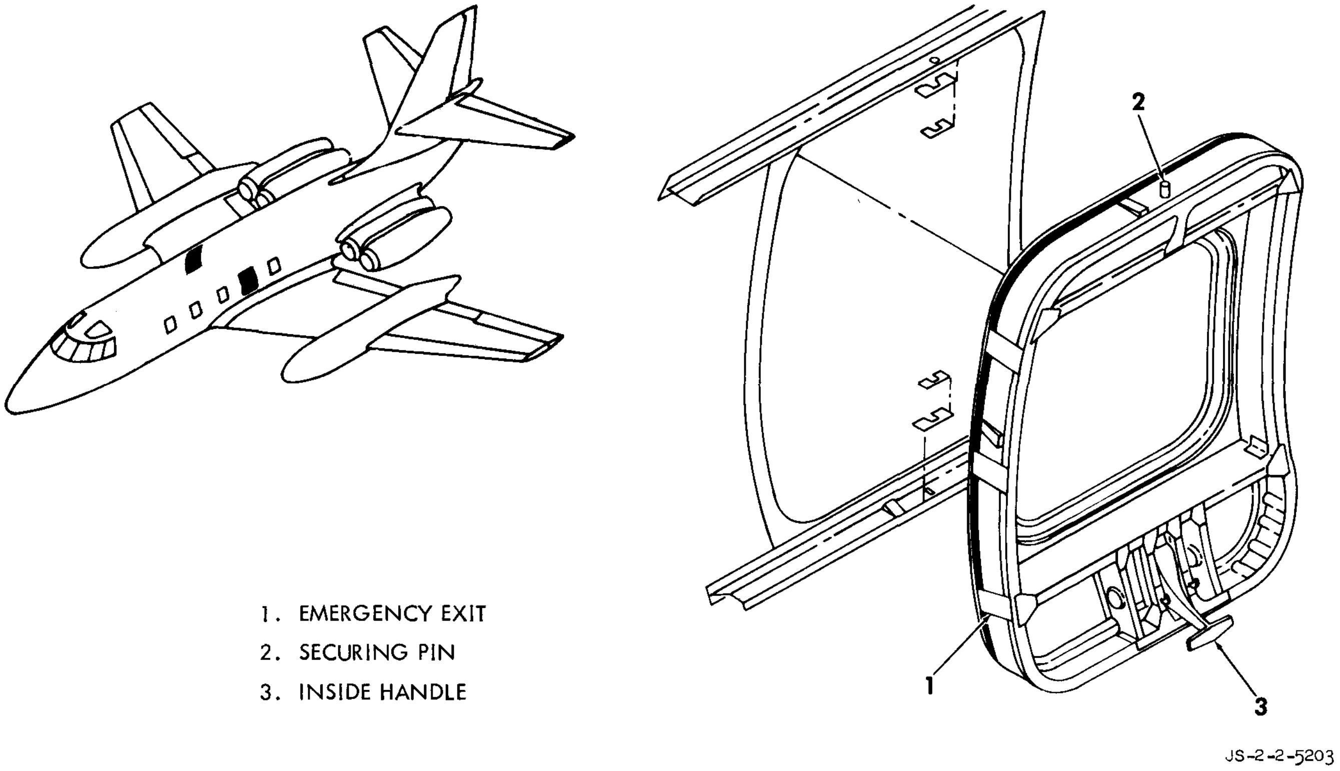
Contents Page

EMERGENCY ESCAPE HATCH 52-5

FIGURES
52-3. Emergency Escape Hatch 52-6

52.1.1. EMERGENCY ESCAPE HATCH. The fourth window aft, on each side of the cabin, is an emergency exit.

The emergency escape hatch release mechanism can be actuated from inside or outside the airplane. To actuate the emergency exit release mechanism from inside the airplane, grasp the PULL handle at the lower side of the window, and pull the window into the passenger compartment. To actuate the emergency exit release mechanism from outside the airplane, press the PUSH lever under the window and push the window into the passenger compartment. The emergency exits are sealed in the same manner as the entrance door. (Refer to paragraph 52.0.1.)



1. EMERGENCY EXIT
2. SECURING PIN
3. INSIDE HANDLE

Figure 52-3. Emergency Escape Hatch

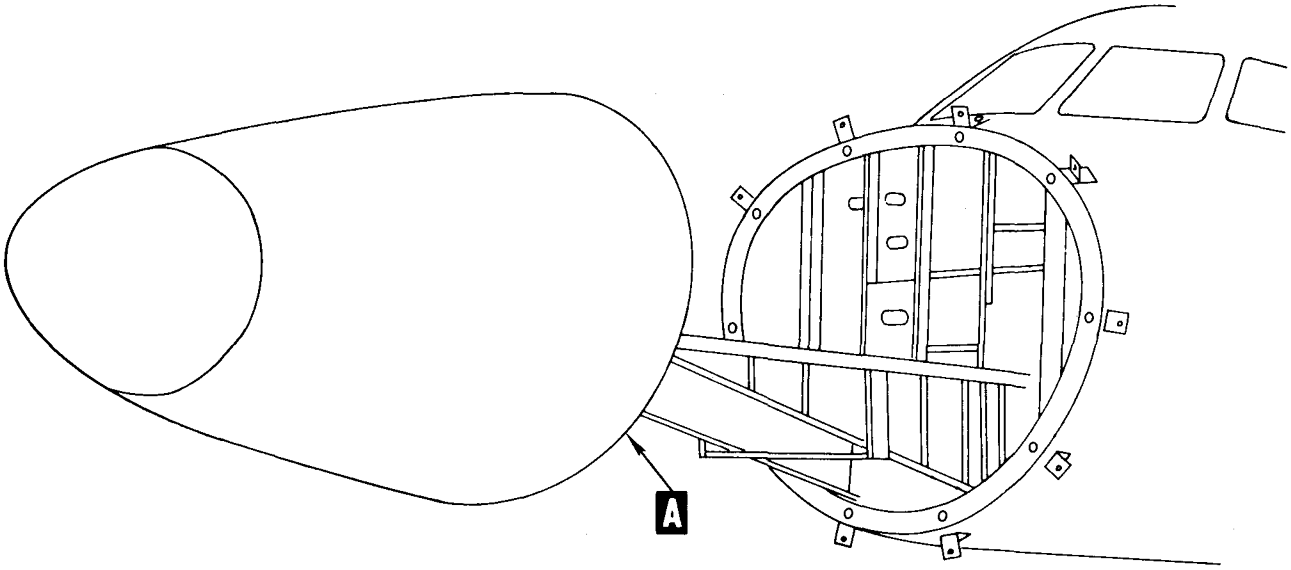
SECTION 52.2
NOSE CONE

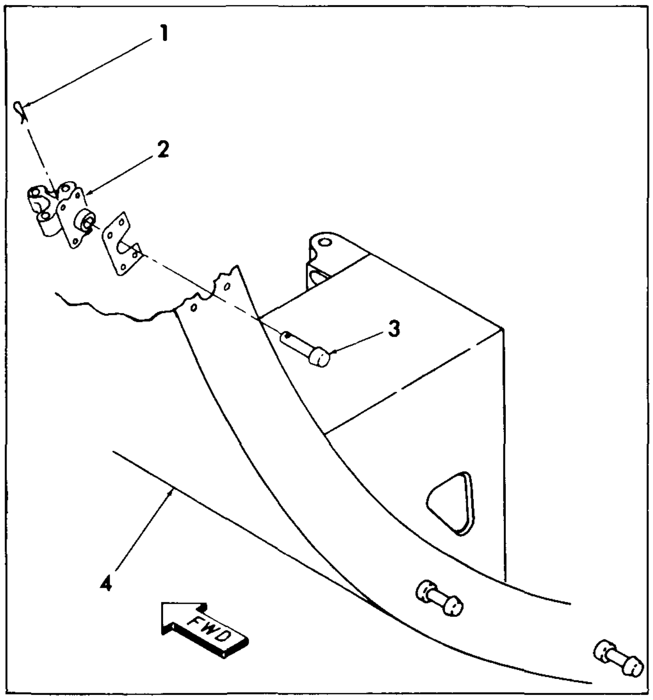
Contents Page

NOSE CONE 52-7

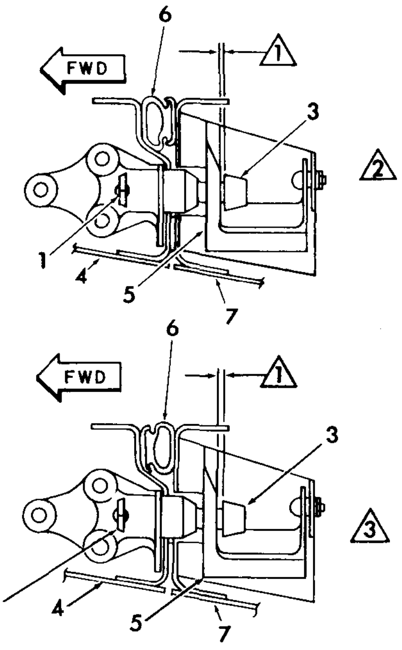
FIGURES
52-4. Nose Cone Latch Adjustment 52-8

52.2.1 NOSE CONE. The nose cone is a light exterior shell of skin, frames, longerons, and radome material. This cone is supported on tracks mounted on internal structures cantilevered from the forward pressure bulkhead. These structures serve as racks for the electronic equipment. The nose cone can be unlatched and rolled forward for service, or it can be removed completely. The nose cone is pressure-sealed in the same manner as the entrance door by a rubber seal attached to the frame at fuselage station 164 on airplanes 5001 through 5035, and to the nose cone aft frame on airplanes 5036 and up. The rubber seal encircles the perimeter of the frame, in both cases.





A



SECTION VIEWS OF LATCH WITH NOSE CONE IN LATCHED POSITION

1. COTTER PIN
2. FITTING
3. STUD
4. NOSE CONE
5. LATCH ASSEMBLY
6. SEAL
7. FUSELAGE SKIN

NOTE

△1 THIS DIMENSION NOT TO EXCEED 0.010 INCH ON ANY LATCH WITH ALL LATCHES CLOSED.

△2 AIRPLANES 5001 THRU 5035

△3 AIRPLANES 5036 AND UP

JS-1-2-X0/2-5204

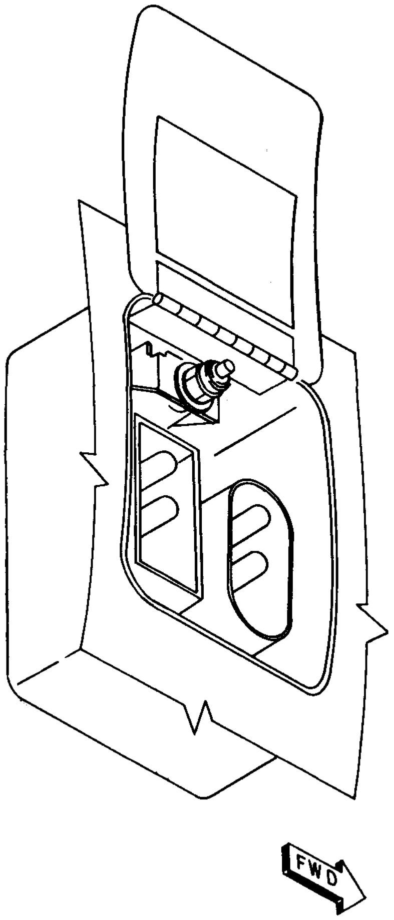
Figure 52-4. Nose Cone Latch Adjustment

SECTION 52.3
ACCESS AND INSPECTION PROVISIONS

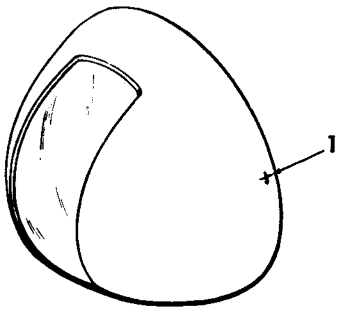
Contents Page

FIGURES

52-5. Access and Inspection Provisions 52-10



① EXTERNAL ELECTRICAL POWER RECEPTACLE DOOR

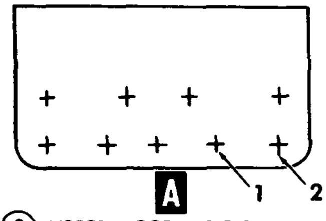


② VENT VALVE, FUEL QUANTITY PROBE, LANDING AND TAXI LIGHTS ACCESS

1. SCREW NAS1623-3 QTY 17

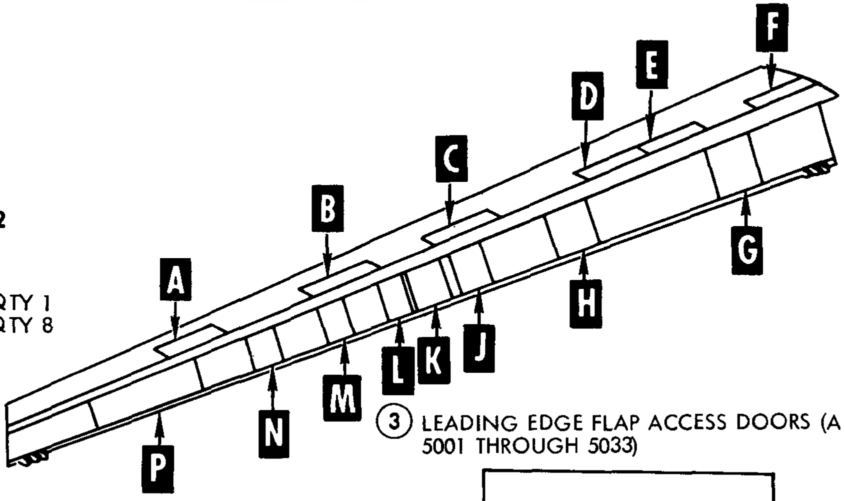
JS-2-2-5205-1

Figure 52-5. Access and Inspection Provisions (Sheet 1 of 22)

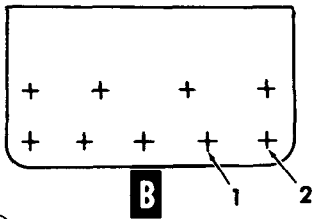


③ UPPER ACCESS DOOR

1. SCREW LS5900-3-9 QTY 1
2. SCREW LS5900-3-8 QTY 8

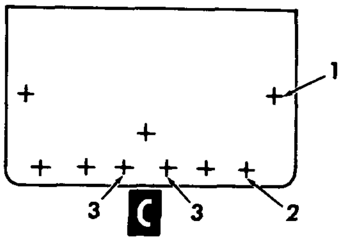


③ LEADING EDGE FLAP ACCESS DOORS (AIRPLANES 5001 THROUGH 5033)



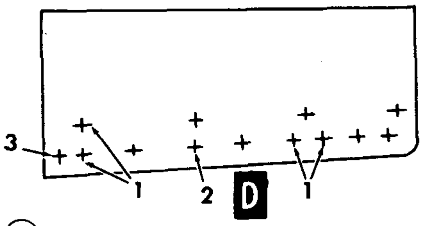
③ UPPER ACCESS DOOR

1. SCREW NAS1623-9 QTY 1
2. SCREW LS5900-3-8 QTY 8



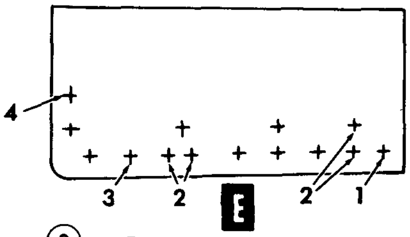
③ UPPER ACCESS DOOR

1. SCREW LS5900-3-6 QTY 2
2. SCREW LS5900-3-8 QTY 5
3. SCREW LS5900-3-9 QTY 2



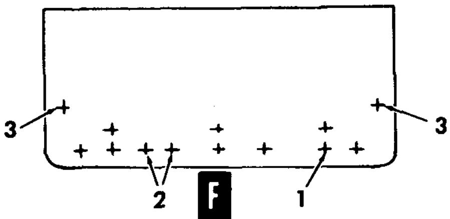
③ UPPER ACCESS DOOR

1. SCREW LS5900-3-9 QTY 4
2. SCREW LS5900-3-8 QTY 8
3. SCREW LS5900-3-5 QTY 1



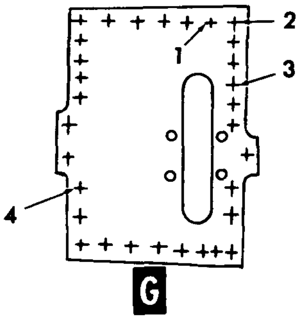
③ UPPER ACCESS DOOR

1. SCREW LS5900-3-5 QTY 1
2. SCREW LS5900-3-9 QTY 4
3. SCREW LS5900-3-8 QTY 8
4. SCREW LS5900-3-6 QTY 1



③ UPPER ACCESS DOOR

1. SCREW LS5900-3-8 QTY 9
2. SCREW LS5900-3-9 QTY 2
3. SCREW LS5900-3-6 QTY 2

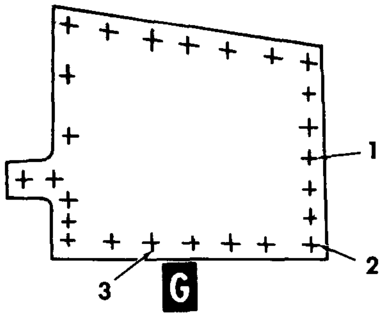


③ REAR BEAM ACCESS DOOR (AIRPLANES 5001 THRU 5015)

1. BOLT NAS1103-3 QTY 5
2. BOLT NAS1103-4 QTY 2
3. SCREW NAS1633-3 QTY 19
4. SCREW NAS1633-4 QTY 5

JS-1-2-X0/2-5205-2

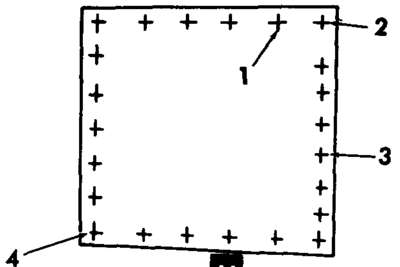
Figure 52-5. Access and Inspection Provisions (Sheet 2 of 22)



G

③ REAR BEAM ACCESS DOOR (AIRPLANES 5016 THRU 5033)

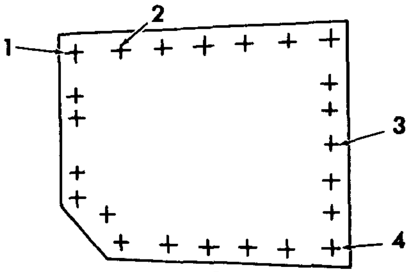
1. SCREW NAS1633-2 QTY 11
2. SCREW NAS1633-3 QTY 10
3. SCREW NAS1633-4 QTY 4



H

③ REAR BEAM ACCESS DOOR

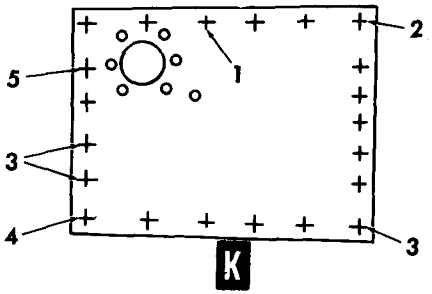
1. BOLT NAS1103-3 QTY 4
2. BOLT NAS1103-4 QTY 2
3. SCREW NAS1633-3 QTY 15
4. SCREW NAS1633-4 QTY 2



J

③ REAR BEAM ACCESS DOOR

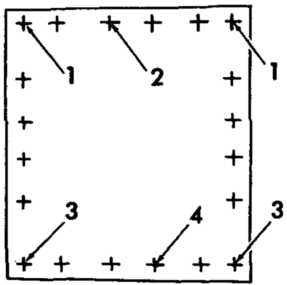
1. BOLT NAS1103-3 QTY 2
2. BOLT NAS1103-2 QTY 5
3. SCREW NAS1633-2 QTY 15
4. SCREW NAS1633-3 QTY 1



K

REAR BEAM ACCESS DOOR

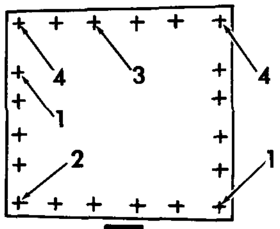
③ 1. BOLT NAS1103-3 QTY 5
2. BOLT NAS1103-5 QTY 1
3. SCREW NAS1633-3 QTY 3
4. SCREW NAS1633-4 QTY 3
5. SCREW NAS1633-2 QTY 9



L

③ REAR BEAM ACCESS DOOR

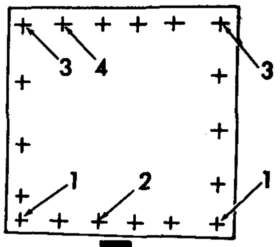
1. BOLT NAS1103-4 QTY 2
2. BOLT NAS1103-3 QTY 4
3. SCREW NAS1633-3 QTY 2
4. SCREW NAS1633-2 QTY 12



M

③ REAR BEAM ACCESS DOOR

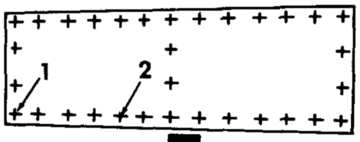
1. SCREW NAS1633-3 QTY 12
2. SCREW NAS1633-4 QTY 2
3. BOLT NAS1103-3 QTY 4
4. BOLT NAS1103-4 QTY 2



N

③ REAR BEAM ACCESS DOOR

1. SCREW NAS1633-4 QTY 2
2. SCREW NAS1633-3 QTY 10
3. BOLT NAS1103-4 QTY 2
4. BOLT NAS1103-3 QTY 4



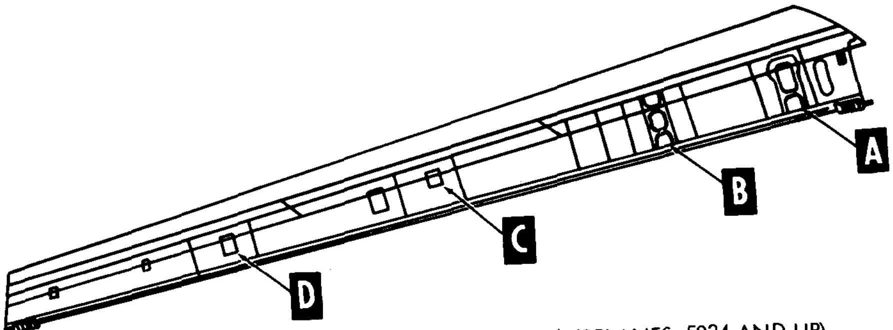
P

③ REAR BEAM ACCESS DOOR

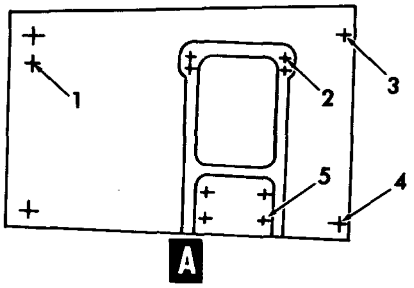
1. SCREW NAS1633-4 QTY 6
2. SCREW NAS1633-3 QTY 26

JS-2-2-5205-3

Figure 52-5. Access and Inspection Provisions (Sheet 3 of 22)

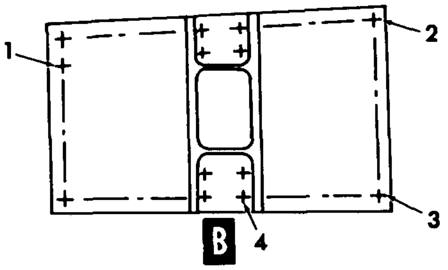


④ LEADING EDGE FLAP ACCESS DOORS (AIRPLANES 5034 AND UP)



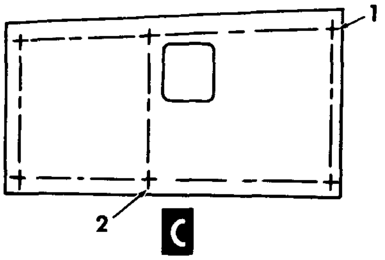
④ LEADING EDGE FLAP ACTUATOR DOOR, STA 252.238

1. SCREW NAS1633-3 QTY 9
2. SCREW NAS1633-4 QTY 4
3. SCREW NAS1623-3 QTY 5
4. BOLT NAS1103-3 QTY 6
5. BOLT NAS1103-4 QTY 4



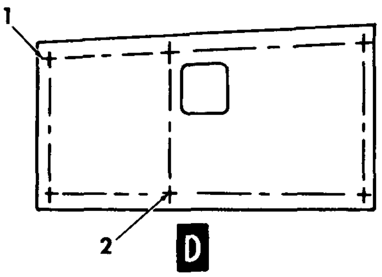
④ LEADING EDGE FLAP ACTUATOR DOOR, STA 271.691

1. SCREW NAS1633-3 QTY 11
2. SCREW NAS1623-3 QTY 5
3. BOLT NAS1103-3 QTY 8
4. BOLT NAS1103-4 QTY 4



④ LEADING EDGE FLAP STOP DOOR STA 306.919

1. SCREW NAS1623-3 QTY 6
2. SCREW NAS1633-4 QTY 29

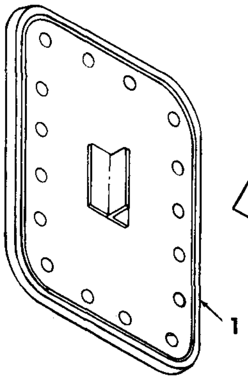


④ LEADING EDGE FLAP STOP DOOR STA 338.466

1. SCREW NAS1623-3 QTY 5
2. SCREW NAS1633-3 QTY 27

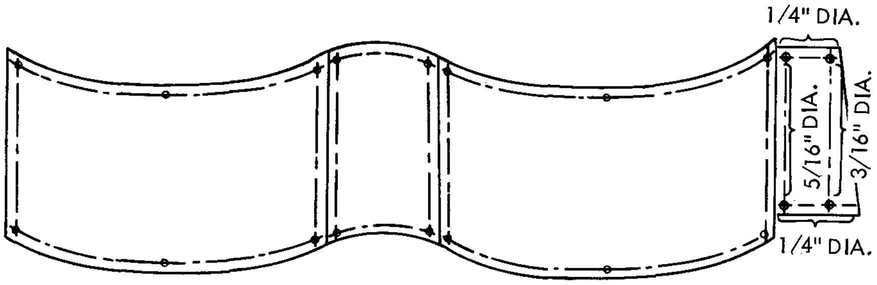
JS-1-2-X0/2-5205-4

Figure 52-5. Access and Inspection Provisions (Sheet 4 of 22)



⑤ VENT VALVE ACCESS (FLAPS DOWN)

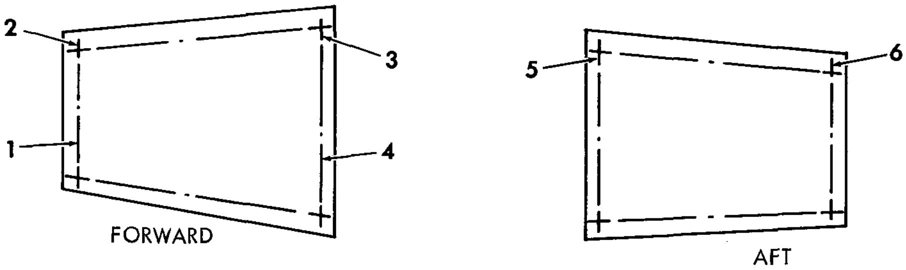
1. SCREW NAS1634-6 QTY 16



⑥ NACELLE BOX BEAM ACCESS

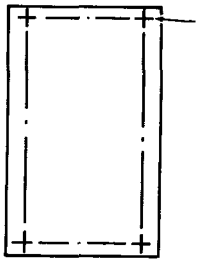
NOTE

FOR SCREW SIZES NOT SHOWN ON THIS DRAWING SEE FIGURE 54-5.



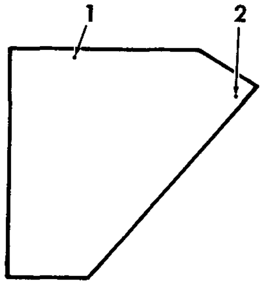
⑦ ACCESS FOR INSPECTION, INSTALLATION, AND HOISTING, FIXED NACELLE

1. STUD 2700-2 QTY 13
2. STUD 2700-4 QTY 2
3. STUD 2700-4 QTY 2
4. STUD 2700-3 QTY 5
5. STUD 2700-4 QTY 6
6. STUD 2700-5 QTY 2



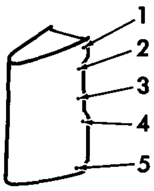
SCREW LENGTHS VARY. REPLACE SAME LENGTH SCREW AS WAS REMOVED

⑧ NACELLE BOX BEAM INBOARD LOWER ACCESS



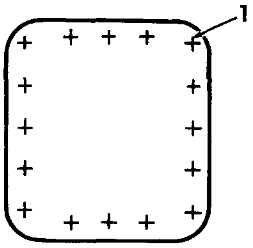
⑨ HORIZONTAL STABILIZER LEADING EDGE ACCESS DOOR

1. SCREW NAS1623-3 QTY 17
2. SCREW NAS1623-4 QTY 2



⑩ VERTICAL STABILIZER LEADING EDGE, LOWER

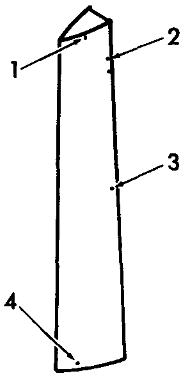
1. SCREW NAS1623-6 QTY 24
2. SCREW NAS1633-6 QTY 10
3. SCREW NAS1633-10 QTY 6
4. SCREW NAS1623-10 QTY 6
5. SCREW NAS1623-4 QTY 6



⑪ VERTICAL STABILIZER LEADING EDGE ACCESS DOOR (RH SIDE ONLY)

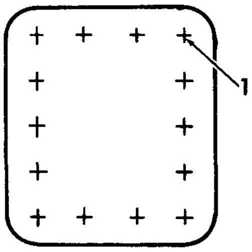
1. SCREW NAS1623-2 QTY 16

Figure 52-5. Access and Inspection Provisions (Sheet 5 of 22)



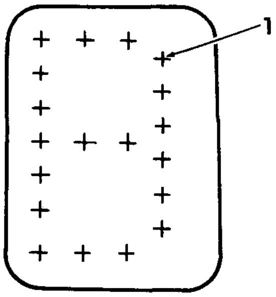
⑫ VERTICAL STABILIZER LEADING EDGE, UPPER

1. SCREW NAS1623-2 QTY 28
2. SCREW NAS1623-7 QTY 64
3. SCREW NAS1623-5 QTY 110
4. SCREW NAS1623-3 QTY 38



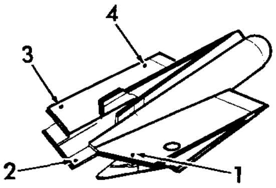
⑬ VERTICAL STABILIZER BOX BEAM ACCESS (LH SIDE ONLY)

1. SCREW NAS1623-2 QTY 14



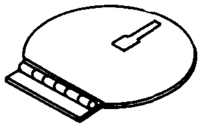
⑭ RUDDER TAB MECHANISM ACCESS (BOTH SIDES)

1. SCREW NAS1623-2 QTY 19

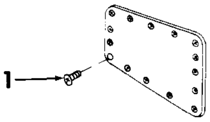


⑮ TRAILING EDGE FAIRING INSTALLATION

1. SCREW NAS1623-7 QTY 1
2. SCREW NAS1623-6 QTY 5
3. SCREW NAS1623-4 QTY 2
4. SCREW NAS1623-3 QTY 68

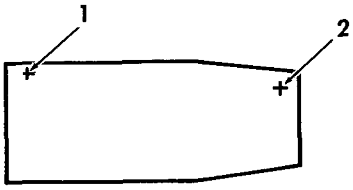


⑯ ELEVATOR TORQUE TUBE ACCESS DOOR



⑰ TRAILING EDGE FAIRING DOOR

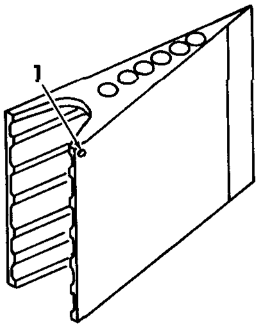
1. SCREW NAS1623-2 QTY 15



⑱ HORIZONTAL STABILIZER AFT BOX BEAM ACCESS DOOR

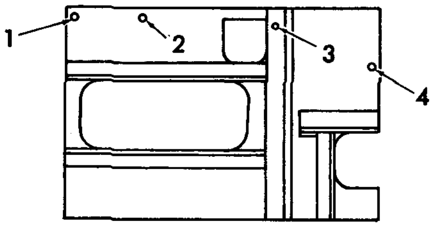
1. SCREW NAS1633-3 QTY 12
2. SCREW NAS1633-4 QTY 5

Figure 52-5. Access and Inspection Provisions (Sheet 6 of 22)



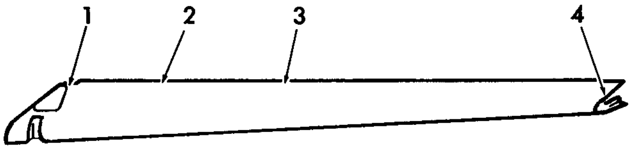
(19) VERTICAL STABILIZER LOWER FAIRING

1. SCREW NAS1623-6 QTY 62



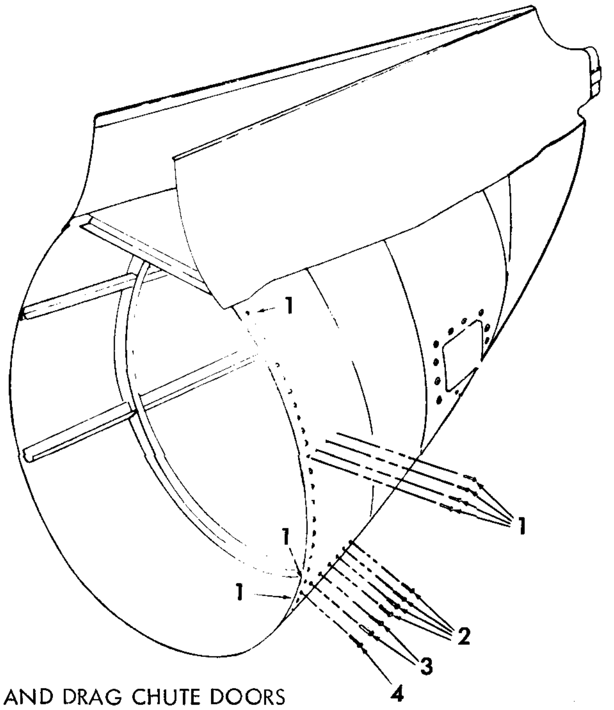
(20) ELEVATOR BOOSTER ACCESS DOOR

1. SCREW NAS1633-4 QTY 6
2. SCREW NAS1633-3 QTY 43
3. SCREW NAS1633-5 QTY 2
4. SCREW NAS1633-2 QTY 5



(21) HORIZONTAL STABILIZER REMOVABLE LEADING EDGE

1. SCREW NAS1623-4 QTY 2
2. SCREW NAS1623-5 QTY 24
3. SCREW NAS1623-3 QTY 98
4. SCREW NAS1623-2 QTY 5

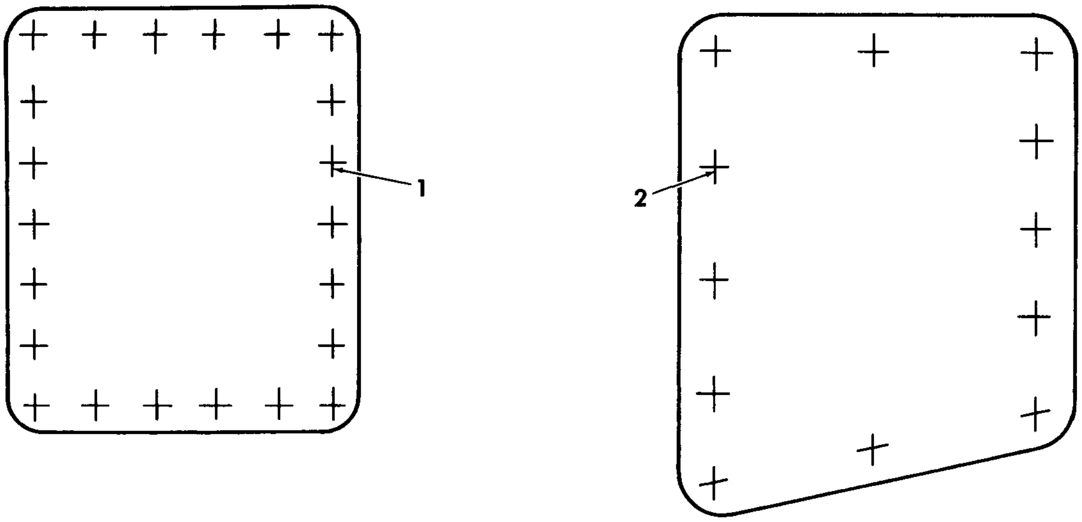


(22) REMOVABLE TAIL CONE AND DRAG CHUTE DOORS

1. SCREW NAS1623-2 QTY 56
2. SCREW NAS1623-4 QTY 8
3. SCREW NAS1623-5 QTY 4
4. SCREW NAS1623-6 QTY 2

JS-2-2-5205-7

Figure 52-5. Access and Inspection Provisions (Sheet 7 of 22)

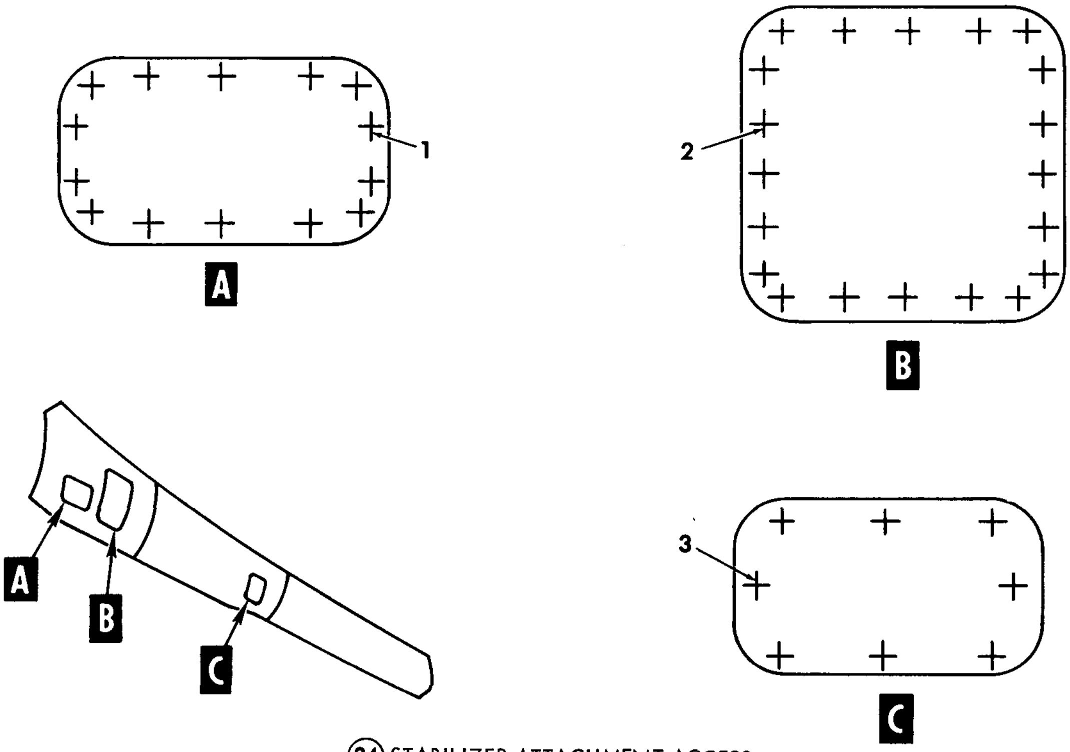


FORWARD DOOR (LH SIDE ONLY)

AFT DOOR (BOTH SIDES)

㉓ ELEVATOR BOOSTER ACCESS DOORS

1. SCREW NAS1633-2 QTY 22
2. STUD 4002-4 QTY 24

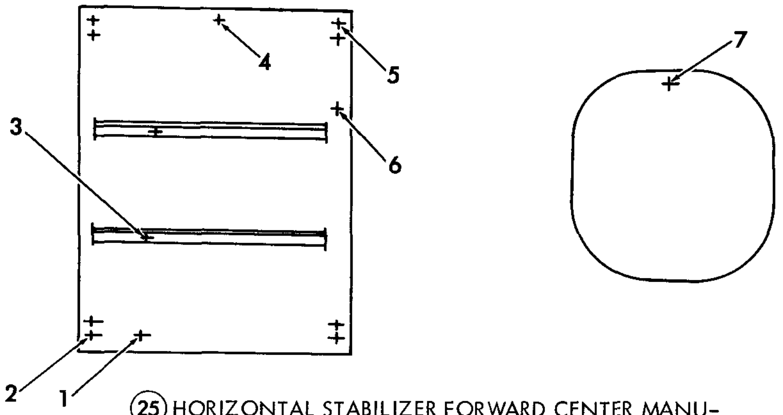


㉔ STABILIZER ATTACHMENT ACCESS

1. STUD 4002-2 QTY 14
2. SCREW NAS1623-2 QTY 19
3. STUD 4002-2 QTY 8

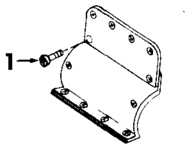
JS-2-2-5205-8

Figure 52-5. Access and Inspection Provisions (Sheet 8 of 22)



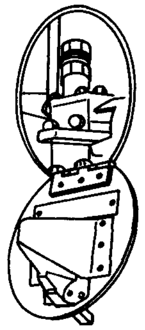
㉕ HORIZONTAL STABILIZER FORWARD CENTER MANUFACTURING AND INSPECTION ACCESS DOOR

1. SCREW NAS1623-4 QTY 4
2. SCREW NAS1623-5 QTY 2
3. SCREW NAS1633-2 QTY 10
4. SCREW NAS1633-4 QTY 30
5. SCREW NAS1633-5 QTY 20
6. SCREW NAS1633-7 QTY 24
7. SCREW NAS1633-3 QTY 20



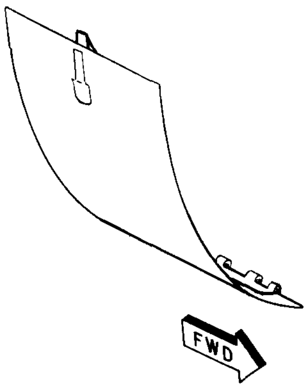
㉖ ELEVATOR AND RUDDER CONTROLS ACCESS DOOR

1. SCREW NAS1623-3 QTY 12



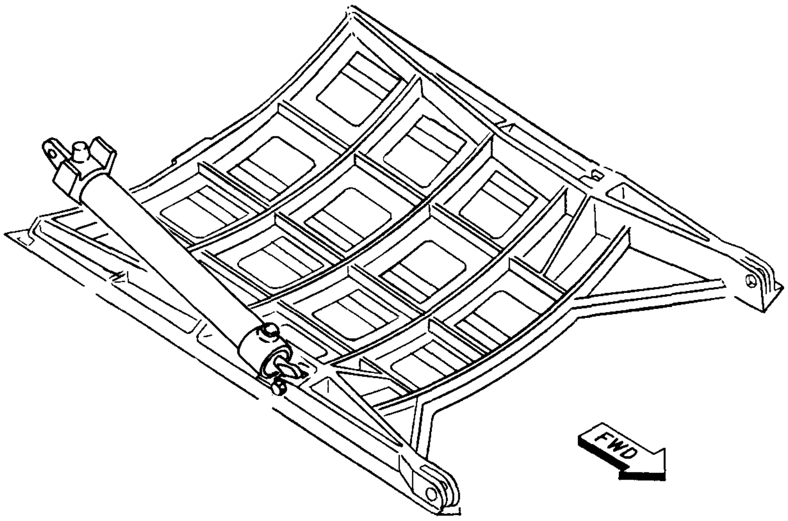
㉗ SPEED BRAKE GROUND CONTROL ACCESS

(AIRPLANES 5001 THRU 5068)

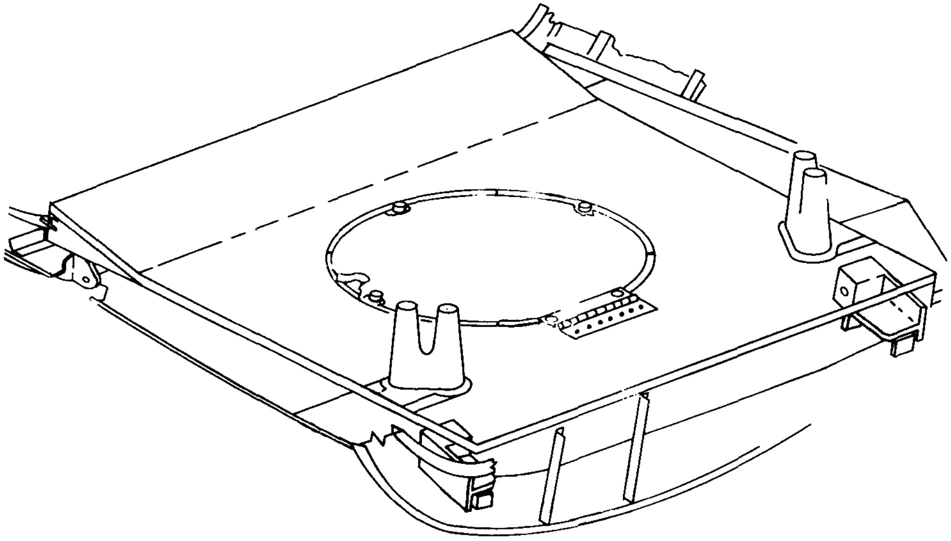


㉘ HYDRAULIC GIG CONNECTION ACCESS DOOR

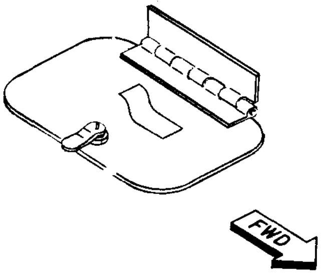
Figure 52-5. Access and Inspection Provisions (Sheet 9 of 22)



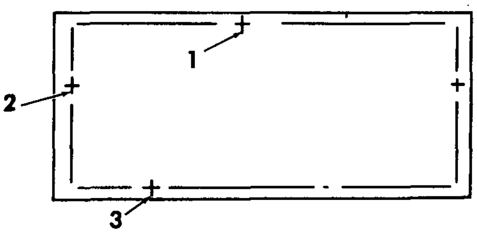
(29) SPEED BRAKE DOOR ACCESS TO AFT COMPARTMENT



(30) SPEED BRAKE OPENING DOOR ASSEMBLY
1. STUD S4650W1-53 QTY 3

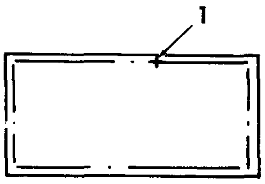


(31) SPEED BRAKE UPLOCK ACCESS DOOR



(32) FUEL VALVES ACCESS DOOR (AIRPLANES 5046 AND UP, AND AIRPLANES INCORPORATING SERVICE BULLETIN 329-155)

1. STUD 2600-14 QTY 4
2. STUD 2600-16 QTY 12
3. STUD 2600-15 QTY 8

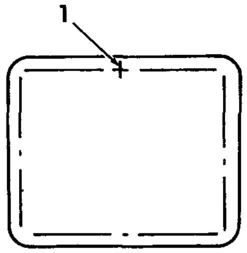


(33) DEFUELING VALVE ACCESS DOOR

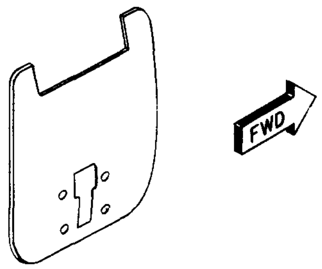
1. SCREW NAS1623-3 QTY 26

JS-2-2-5205-10

Figure 52-5. Access and Inspection Provisions (Sheet 10 of 22)

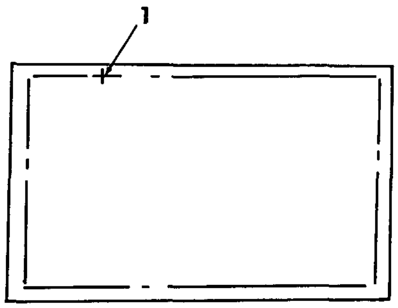


㉞ FUEL CHECK VALVES ACCESS
1. STUD 4002-3 QTY 16



㉟ WATER RECEPTACLE ACCESS DOOR

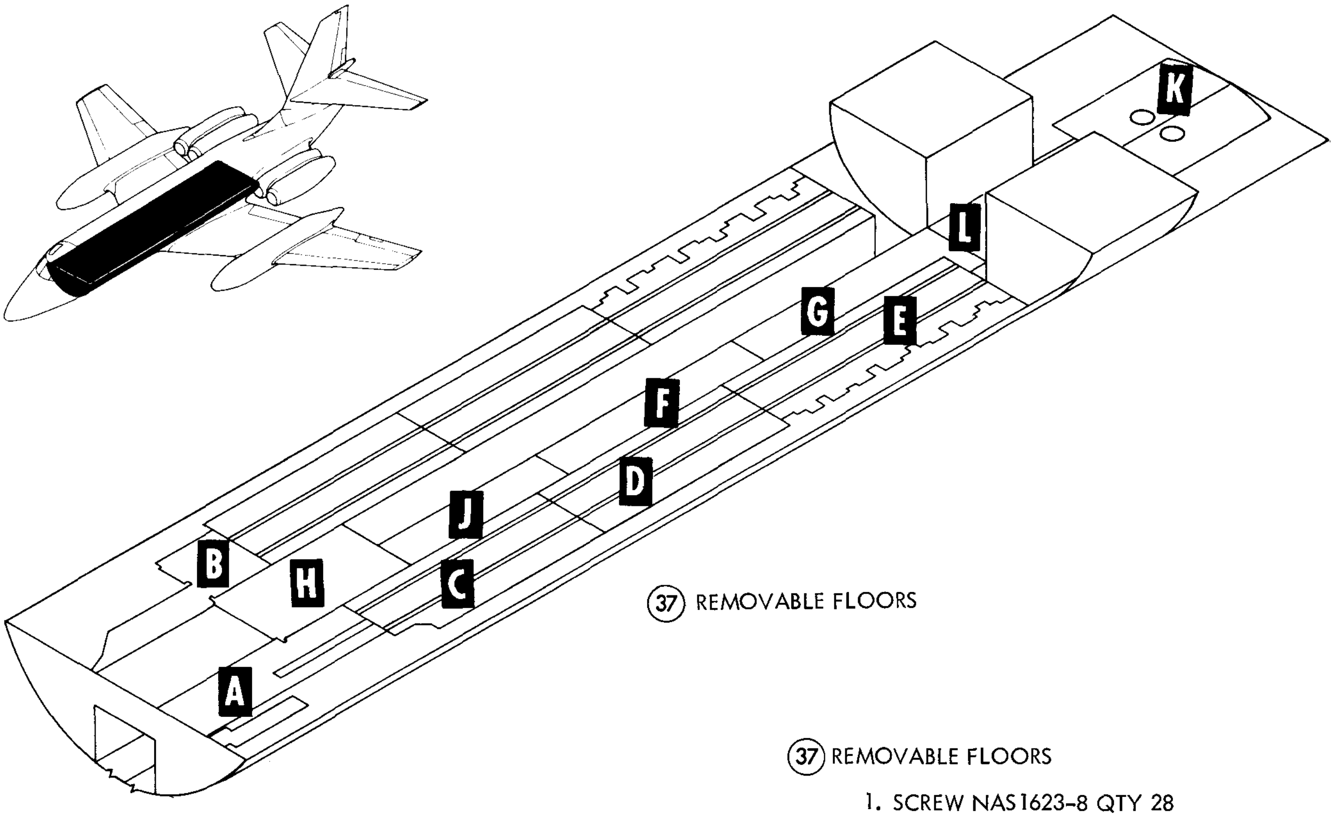
JS-2-2-5205-11



㊱ AILERON BOOSTER AND TRAILING EDGE FLAP MOTOR ACCESS DOOR

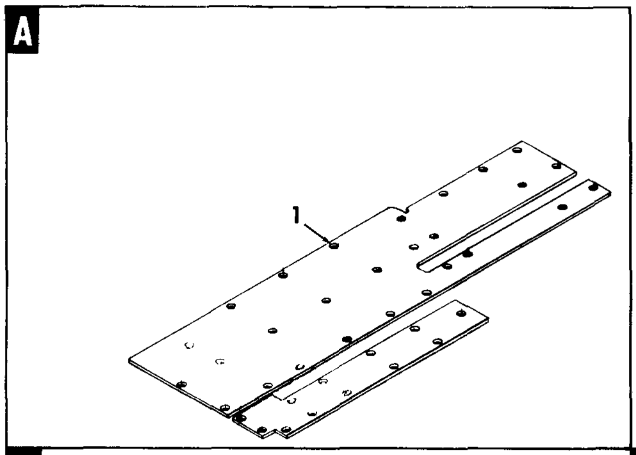
1. STUD 4002-3 QTY 28

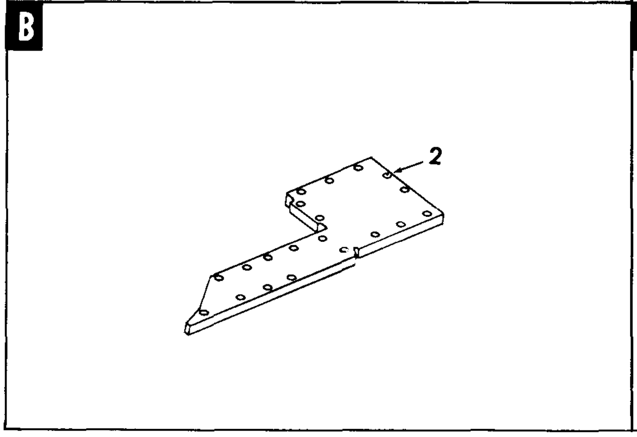
Figure 52-5. Access and Inspection Provisions (Sheet 11 of 22)

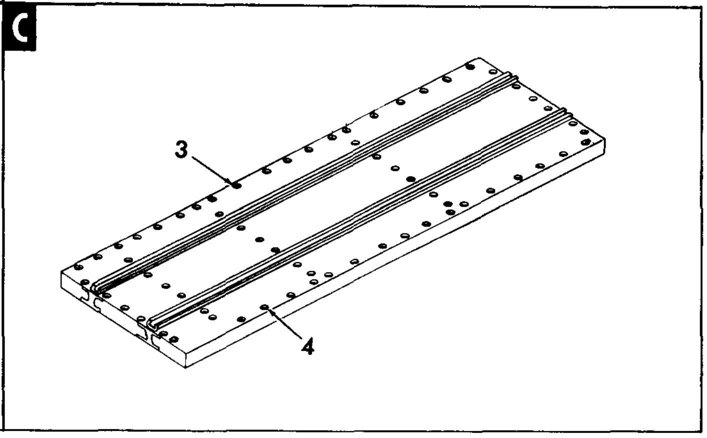


(37) REMOVABLE FLOORS

1. SCREW NAS1623-8 QTY 28
2. SCREW NAS1623-8 QTY 20
3. SCREW NAS1623-14 QTY 49
4. SCREW NAS1623-15 QTY 16
5. SCREW NAS1623-14 QTY 40
6. SCREW NAS1623-16 QTY 1
7. SCREW NAS1623-15 QTY 17
8. SCREW NAS1623-14 QTY 88
9. SCREW NAS1634-15 QTY 5
10. SCREW NAS1633-14 QTY 12
11. SCREW NAS1623-9 QTY 46
12. SCREW NAS1623-9 QTY 32
13. SCREW NAS1623-8 QTY 14
14. SCREW NAS1623-9 QTY 5
15. SCREW NAS1623-13 QTY 27
16. SCREW NAS1623-8 QTY 44
17. SCREW NAS1623-9 QTY 26
18. SCREW NAS1623-8 QTY 56
19. SCREW NAS1623-7 QTY 16
20. SCREW NAS1623-9 QTY 16







JS-2-2-5205-12

Figure 52-5. Access and Inspection Provisions (Sheet 12 of 22)

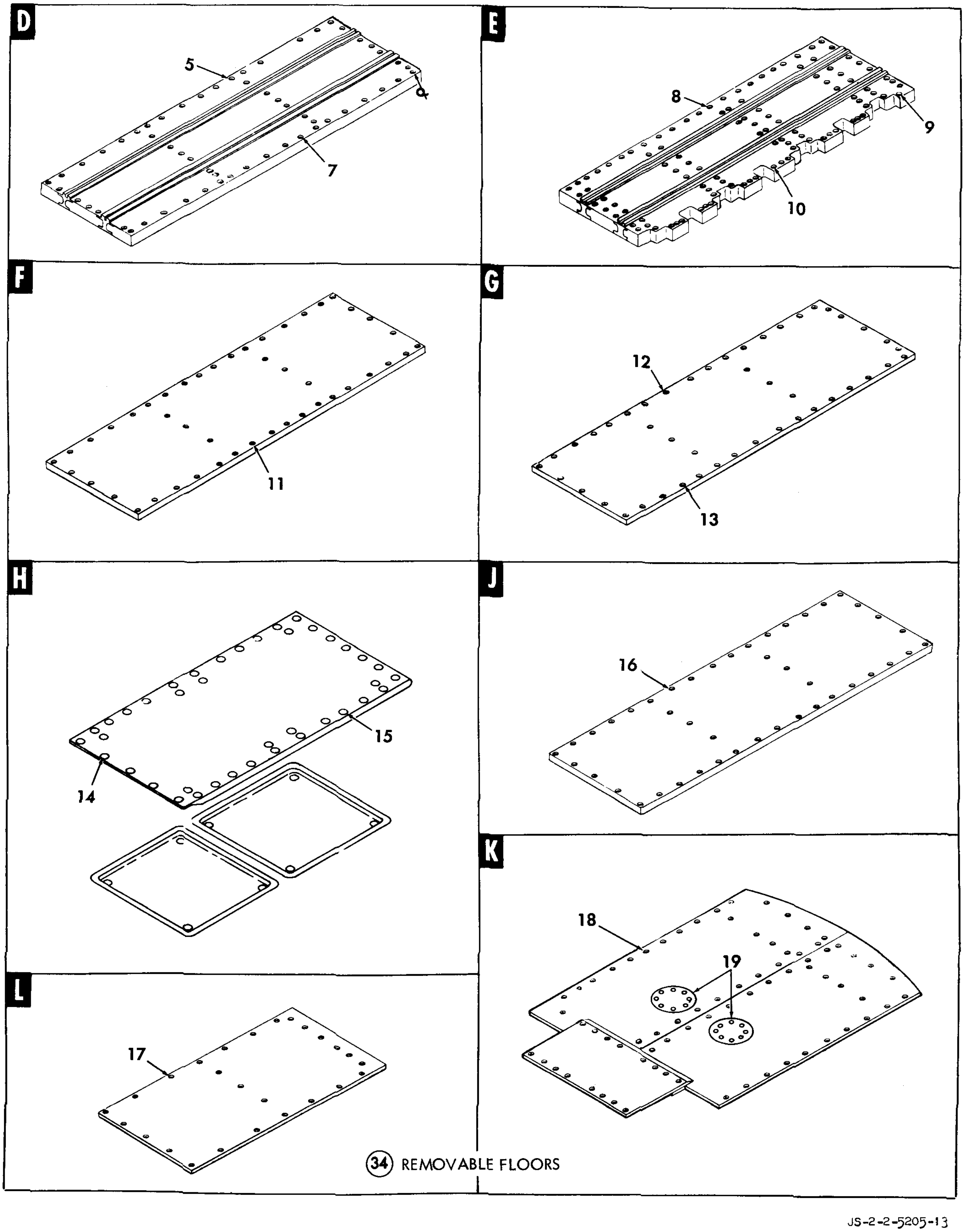
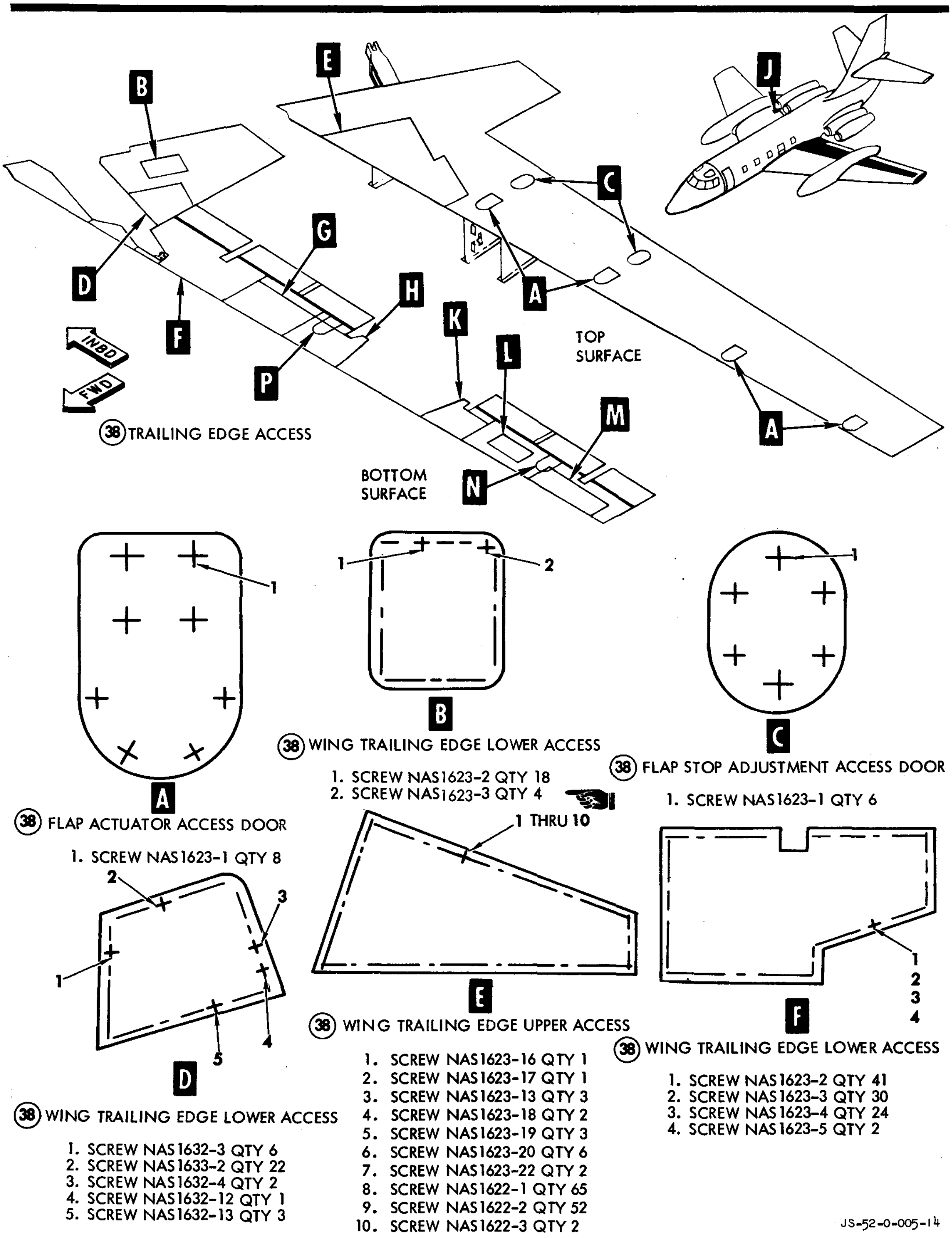


Figure 52-5. Access and Inspection Provisions (Sheet 13 of 22)



(38) TRAILING EDGE ACCESS

A (38) FLAP ACTUATOR ACCESS DOOR

1. SCREW NAS1623-1 QTY 8

B (38) WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1623-2 QTY 18
2. SCREW NAS1623-3 QTY 4

C (38) FLAP STOP ADJUSTMENT ACCESS DOOR

1. SCREW NAS1623-1 QTY 6

D (38) WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1632-3 QTY 6
2. SCREW NAS1633-2 QTY 22
3. SCREW NAS1632-4 QTY 2
4. SCREW NAS1632-12 QTY 1
5. SCREW NAS1632-13 QTY 3

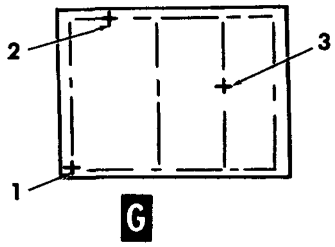
E (38) WING TRAILING EDGE UPPER ACCESS

1. SCREW NAS1623-16 QTY 1
2. SCREW NAS1623-17 QTY 1
3. SCREW NAS1623-13 QTY 3
4. SCREW NAS1623-18 QTY 2
5. SCREW NAS1623-19 QTY 3
6. SCREW NAS1623-20 QTY 6
7. SCREW NAS1623-22 QTY 2
8. SCREW NAS1622-1 QTY 65
9. SCREW NAS1622-2 QTY 52
10. SCREW NAS1622-3 QTY 2

F (38) WING TRAILING EDGE LOWER ACCESS

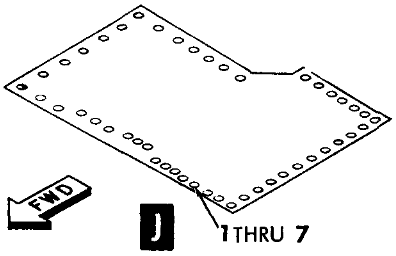
1. SCREW NAS1623-2 QTY 41
2. SCREW NAS1623-3 QTY 30
3. SCREW NAS1623-4 QTY 24
4. SCREW NAS1623-5 QTY 2

Figure 52-5. Access and Inspection Provisions (Sheet 14 of 22)



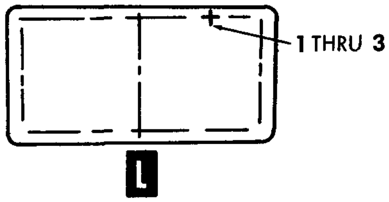
㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1622-3 QTY 1
2. SCREW NAS1622-1 QTY 24
3. SCREW NAS1622-2 QTY 33



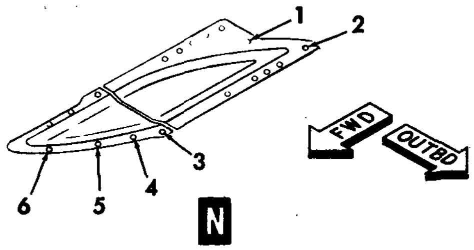
㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1623-3 QTY 6
2. SCREW NAS1633-3 QTY 3
3. SCREW NAS1622-1 QTY 7
4. SCREW NAS1622-2 QTY 13
5. SCREW NAS1632-1 QTY 24
6. SCREW NAS1632-2 QTY 17
7. SCREW NAS1632-3 QTY 1



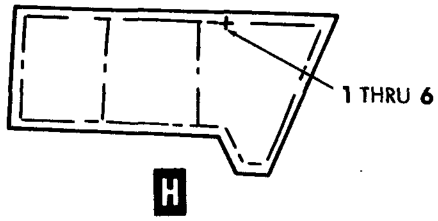
㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1622-1 QTY 24
2. SCREW NAS1622-1 QTY 5
3. SCREW NAS1622-3 QTY 1



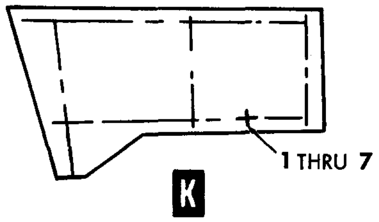
㊳ WING TRAILING EDGE LOWER ACCESS - FLAP ACTUATOR LINKAGE FAIRING

1. SCREW NAS1622-4 QTY 11
2. SCREW NAS1622-3 QTY 1
3. SCREW NAS1623-5 QTY 2
4. SCREW NAS1622-4 QTY 2
5. SCREW NAS1622-3 QTY 1
6. SCREW NAS1623-12 QTY 4



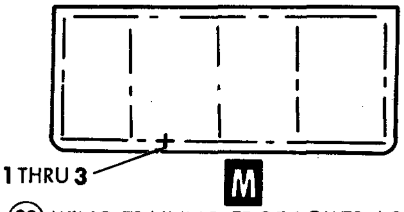
㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1623-3 QTY 3
2. SCREW NAS1633-3 QTY 3
3. SCREW NAS1622-1 QTY 7
4. SCREW NAS1622-2 QTY 13
5. SCREW NAS1632-1 QTY 4
6. SCREW NAS1632-2 QTY 15



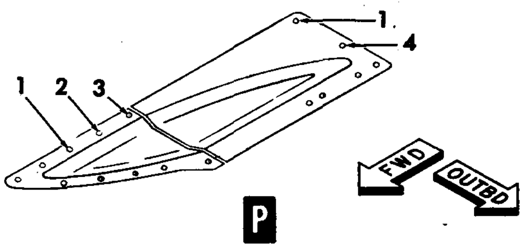
㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1633-3 QTY 3
2. SCREW NAS1632-1 QTY 5
3. SCREW NAS1632-2 QTY 12
4. SCREW NAS1623-3 QTY 3
5. SCREW NAS1622-1 QTY 1
6. SCREW NAS1622-2 QTY 17
7. SCREW NAS1622-3 QTY 1



㊳ WING TRAILING EDGE LOWER ACCESS

1. SCREW NAS1623-3 QTY 4
2. SCREW NAS1622-1 QTY 8
3. SCREW NAS1622-2 QTY 38

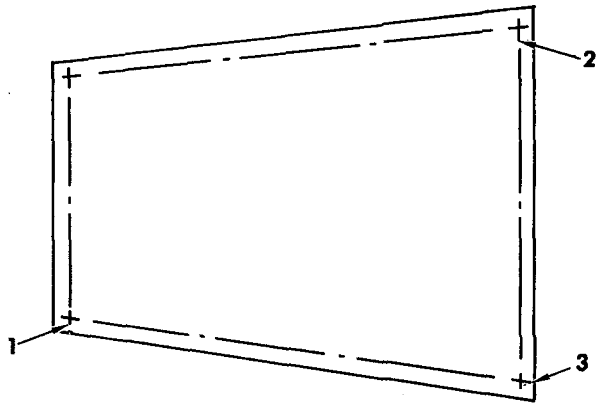


㊳ WING TRAILING EDGE LOWER ACCESS - FLAP ACTUATOR LINKAGE FAIRING

1. SCREW NAS1622-2 QTY 7
2. SCREW NAS1622-3 QTY 1
3. SCREW NAS1622-5 QTY 2
4. SCREW NAS1622-3 QTY 3

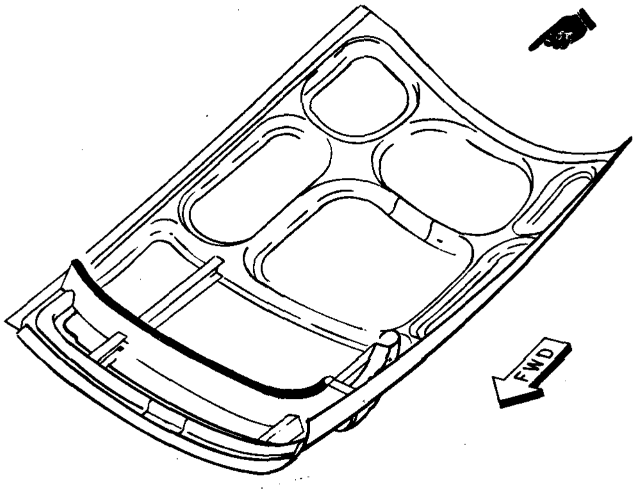
JS-2-2-5205-15

Figure 52-5. Access and Inspection Provisions (Sheet 15 of 22)

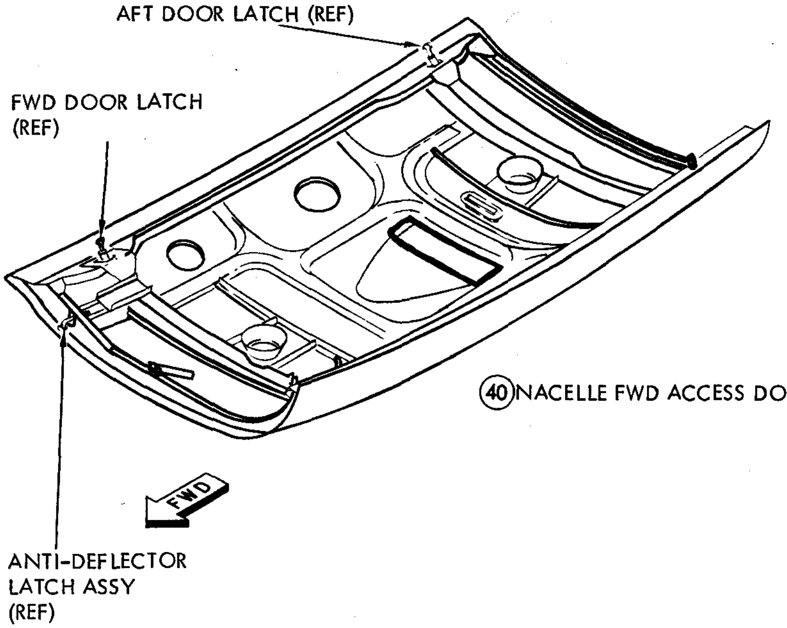


(39) NACELLE LOWER FORWARD ACCESS DOOR

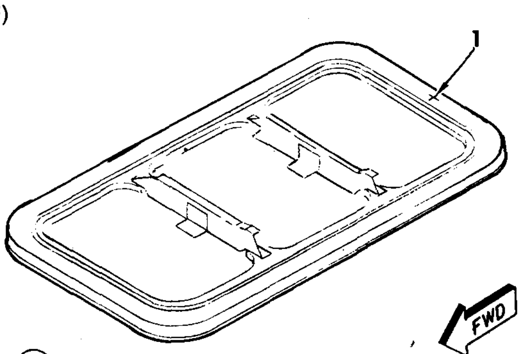
1. STUD 2700-5 QTY 1
2. STUD 2700-3 QTY 31
3. STUD 2700-4 QTY 1



(40) NACELLE AFT ACCESS DOOR
(RETAINED BY COWL FASTENERS)

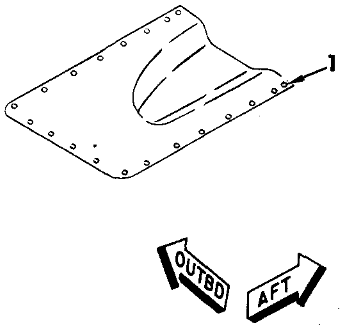


(40) NACELLE FWD ACCESS DOOR



(41) EXTERNAL WING TANK INNER ACCESS DOORS
(3 PLACES/TANK: ABOVE WING, BELOW WING
AND UPPER AFT COMPARTMENT)

1. SCREW NAS1633-5 QTY 74

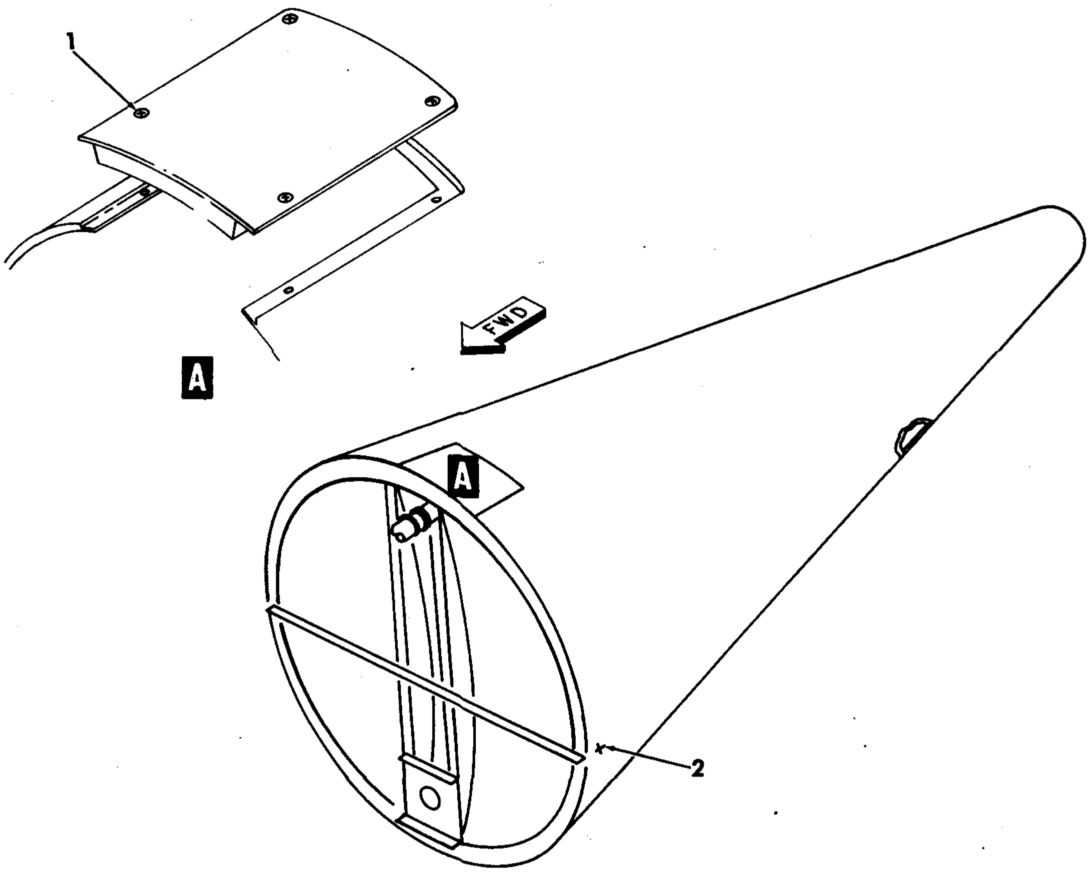


(42) AILERON TAB ACCESS

1. SCREW NAS1622-3 QTY 22

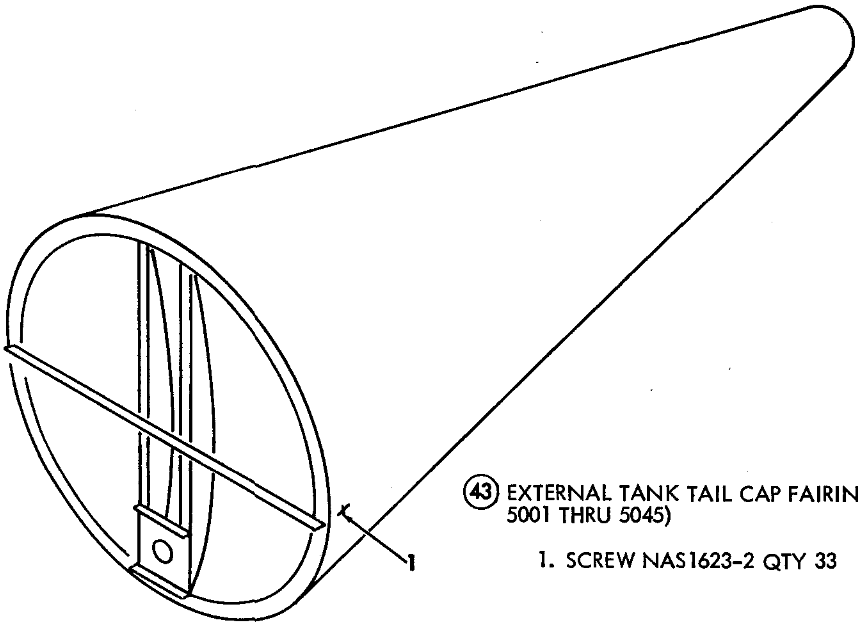
JS-52-0-005-16

Figure 52-5. Access and Inspection Provisions (Sheet 16 of 22)



㊸ EXTERNAL TANK TAIL CAP FAIRING (AIRPLANES 5046 AND UP)

1. SCREW NAS1623-4 QTY 8
2. SCREW NAS1623-2 QTY 38

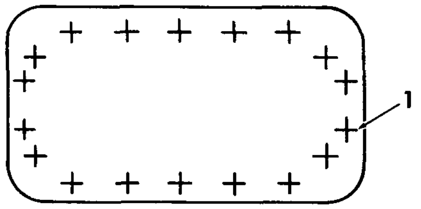


㊸ EXTERNAL TANK TAIL CAP FAIRING (AIRPLANES 5001 THRU 5045)

1. SCREW NAS1623-2 QTY 33

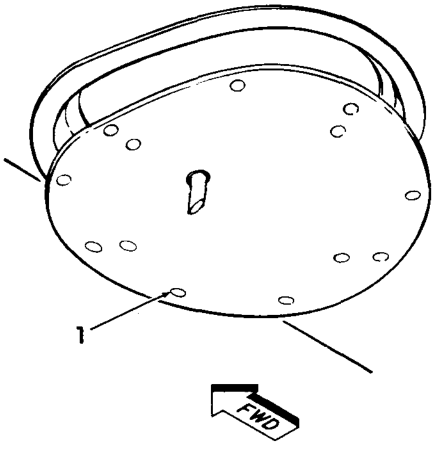
JS-2-2-5205-17

Figure 52-5. Access and Inspection Provisions (Sheet 17 of 22)



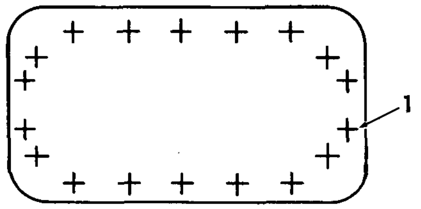
(44) WING OUTBOARD VENT VALVE INSPECTION DOOR

1. SCREW LS5915-3-3 QTY 18



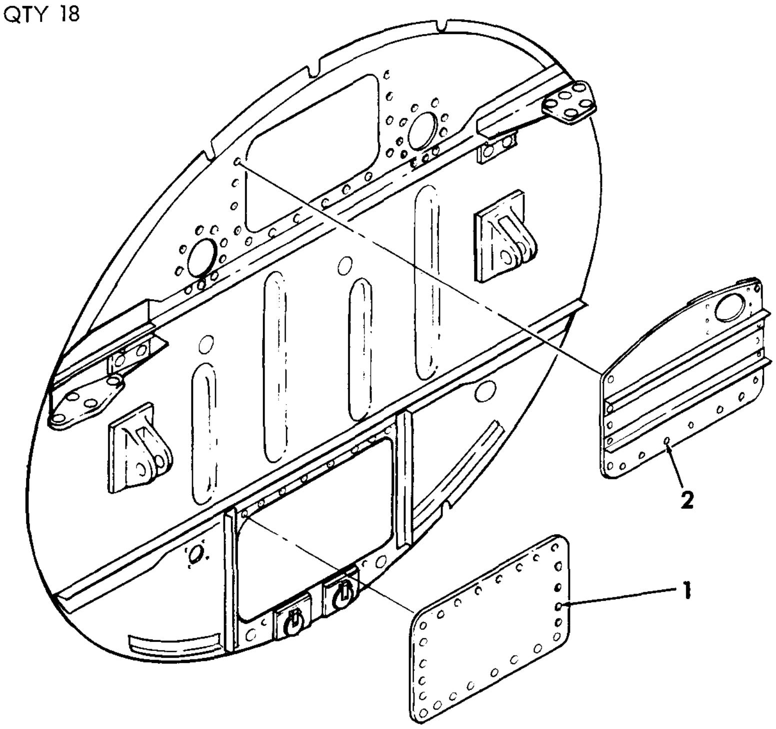
(45) EXTERNAL TANK FUEL BOOST PUMP ACCESS

1. SCREW NAS1623-2 QTY 9



(46) WING INBOARD VENT VALVE INSPECTION DOOR

1. SCREW LS5915-3-3 QTY 18

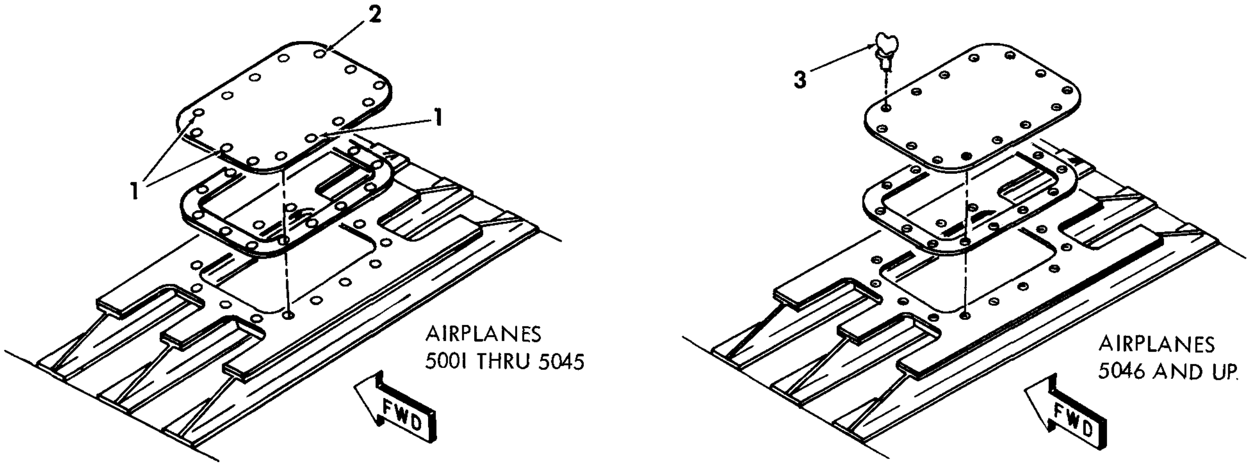


(47) EXTERNAL TANK BULKHEAD COVER PLATES

1. SCREW NAS1633-4 QTY 23
2. SCREW NAS1633-2 QTY 16

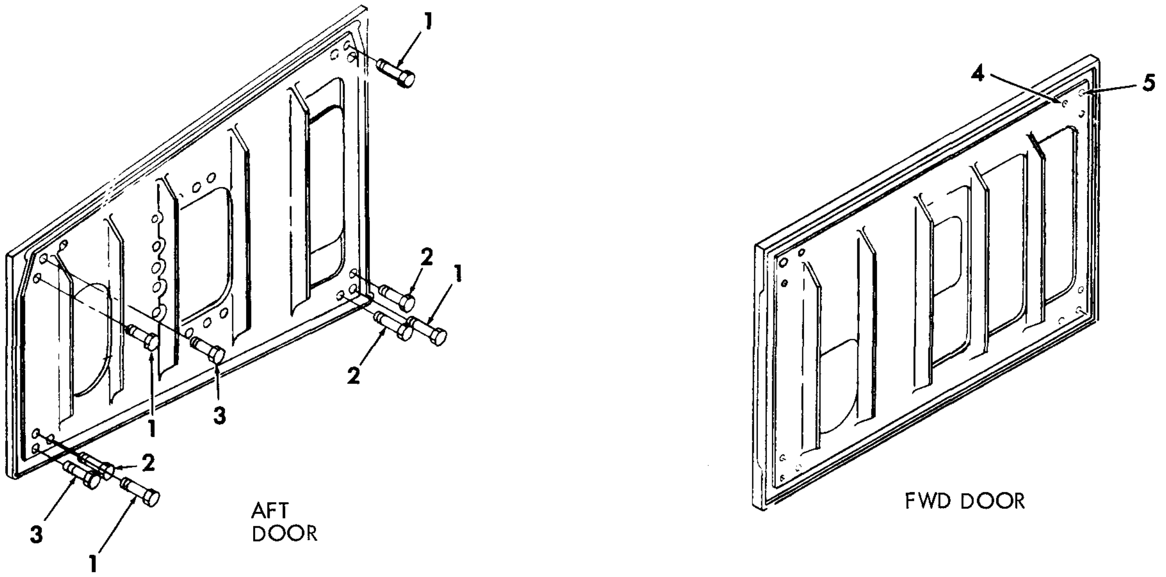
JS-2-2-5205-18

Figure 52-5. Access and Inspection Provisions (Sheet 18 of 22)



㊽ MAIN WHEEL WELL ACCESS DOOR

1. SCREW NAS1623-3 QTY 3 (AIRPLANES 5001 THRU 5045)
2. SCREW NAS1633-3 QTY 11 (AIRPLANES 5001 THRU 5045)
3. STUD 2600-8W QTY 8 (AIRPLANES 5046 AND UP)

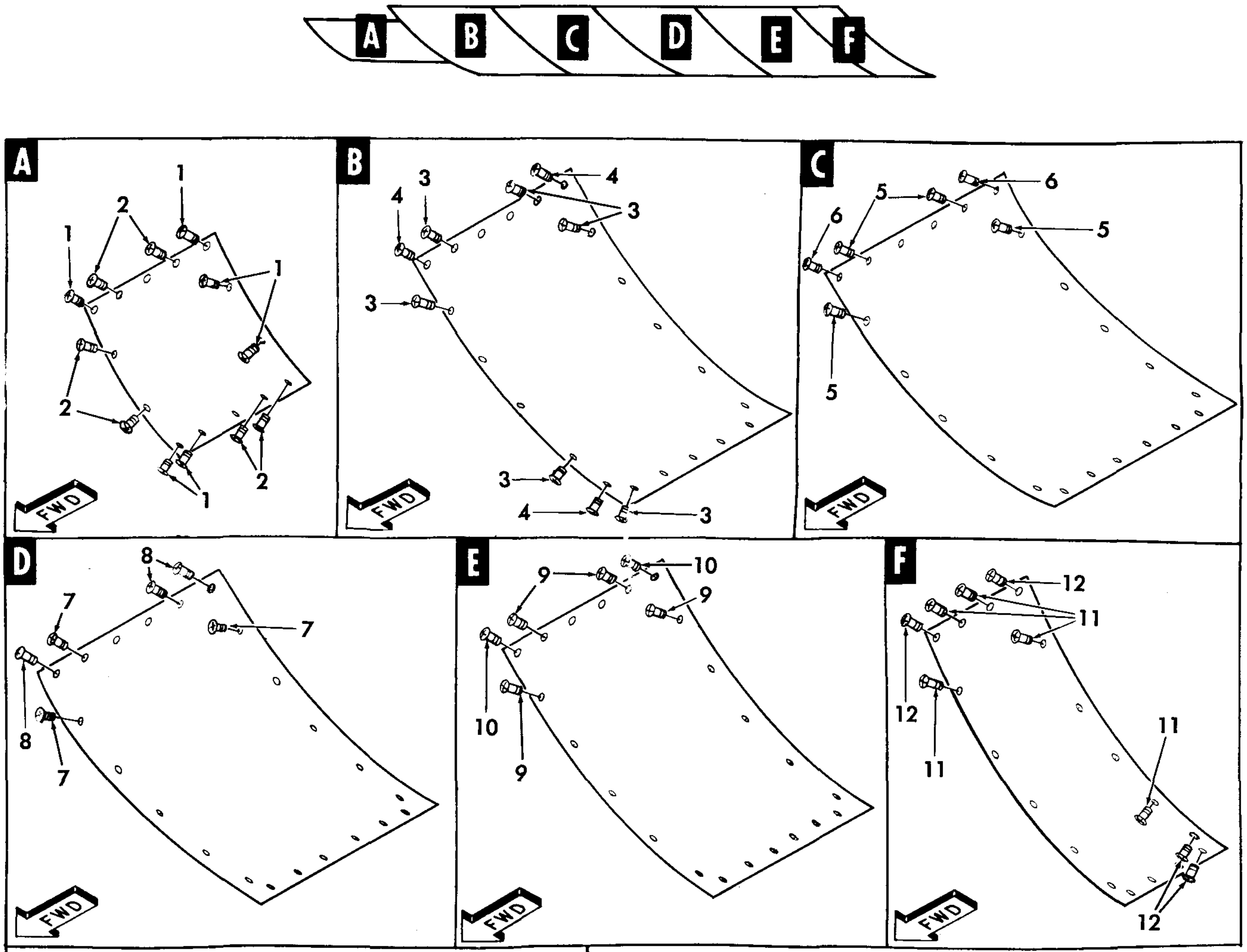


㊾ WING ROOT RIB FUEL BOOST PUMP ACCESS

1. BOLT AN174H11A QTY 10
2. BOLT AN174H10A QTY 42
3. BOLT AN174H12A QTY 2
4. BOLT AN174H7A QTY 56
5. BOLT AN174H10A QTY 4

JS-2-2-5205-19

Figure 52-5. Access and Inspection Provisions (Sheet 19 of 22)



㊿ BELLY FAIRINGS

1. SCREW NAS 1623-3 QTY 4
2. SCREW NAS1623-2 QTY 10
3. SCREW NAS1623-2 QTY 18
4. SCREW NAS1623-3 QTY 3
5. SCREW NAS1623-2 QTY 20
6. SCREW NAS1623-3 QTY 2
7. SCREW NAS1623-2 QTY 20
8. SCREW NAS1623-3 QTY 2
9. SCREW NAS1623-2 QTY 20
10. SCREW NAS 1623-3 QTY 2
11. SCREW NAS 1623-2 QTY 15
12. SCREW NAS 1623-3 QTY 3

(51) ROOT RIB AND LEADING EDGE FLAP MOTOR ACCESS PANELS

1. SCREW NAS1623-7 QTY 9
2. SCREW NAS 1623-2 QTY 23
3. SCREW NAS 1623-3 QTY 1
4. SCREW NAS 1623-4 QTY 1
5. SCREW NAS 1623-7 QTY 10
6. SCREW NAS 1623-2 QTY 28
7. SCREW NAS1623-7 QTY 10
8. SCREW NAS 1623-2 QTY 23
9. SCREW NAS1623-2 QTY 10

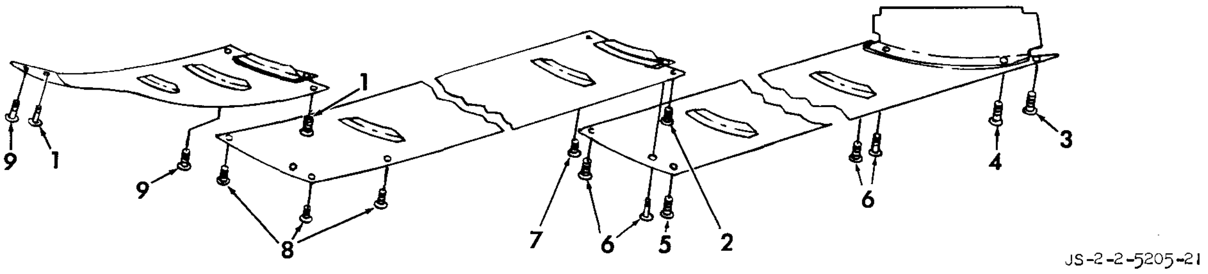
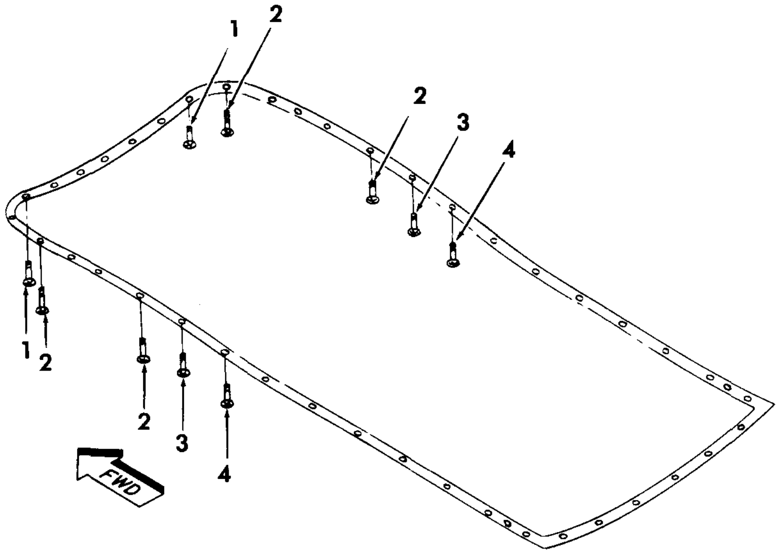
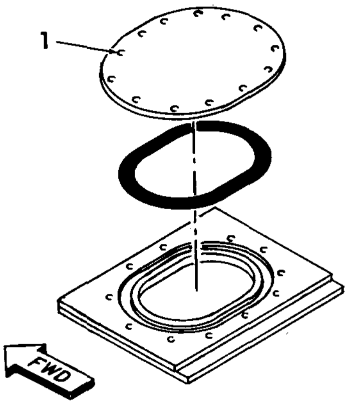


Figure 52-5. Access and Inspection Provisions (Sheet 20 of 22)



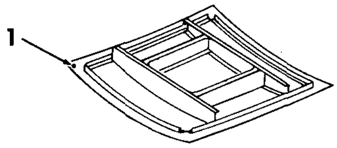
(52) LOOP ANTENNA ACCESS COVER

1. SCREW NAS1623-3 QTY 7
2. SCREW NAS1623-4 QTY 10
3. SCREW NAS 1623-9 QTY 2
4. SCREW NAS 1623-2 QTY 24



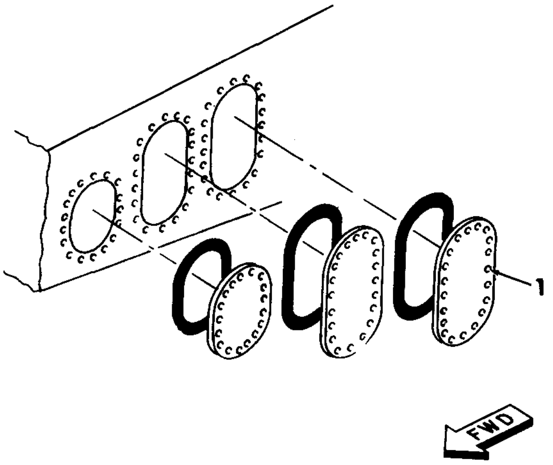
(53) NOSE WHEEL WELL HAND HOLE

1. SCREW NAS1623-4 QTY 12



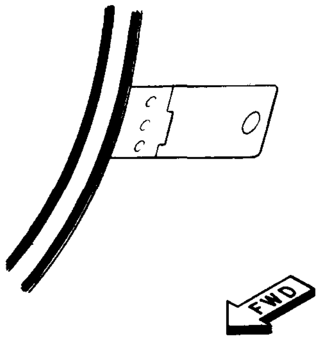
(54) NOSE SECTION LOWER ACCESS

1. SCREW NAS1623-2 QTY 48

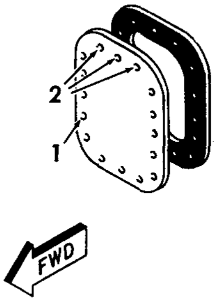


(55) NOSE WHEEL WELL ACCESS COVERS

1. SCREW NAS1633-3 QTY 62



(56) NOSE CONE LATCH COVER (10 PLACES)

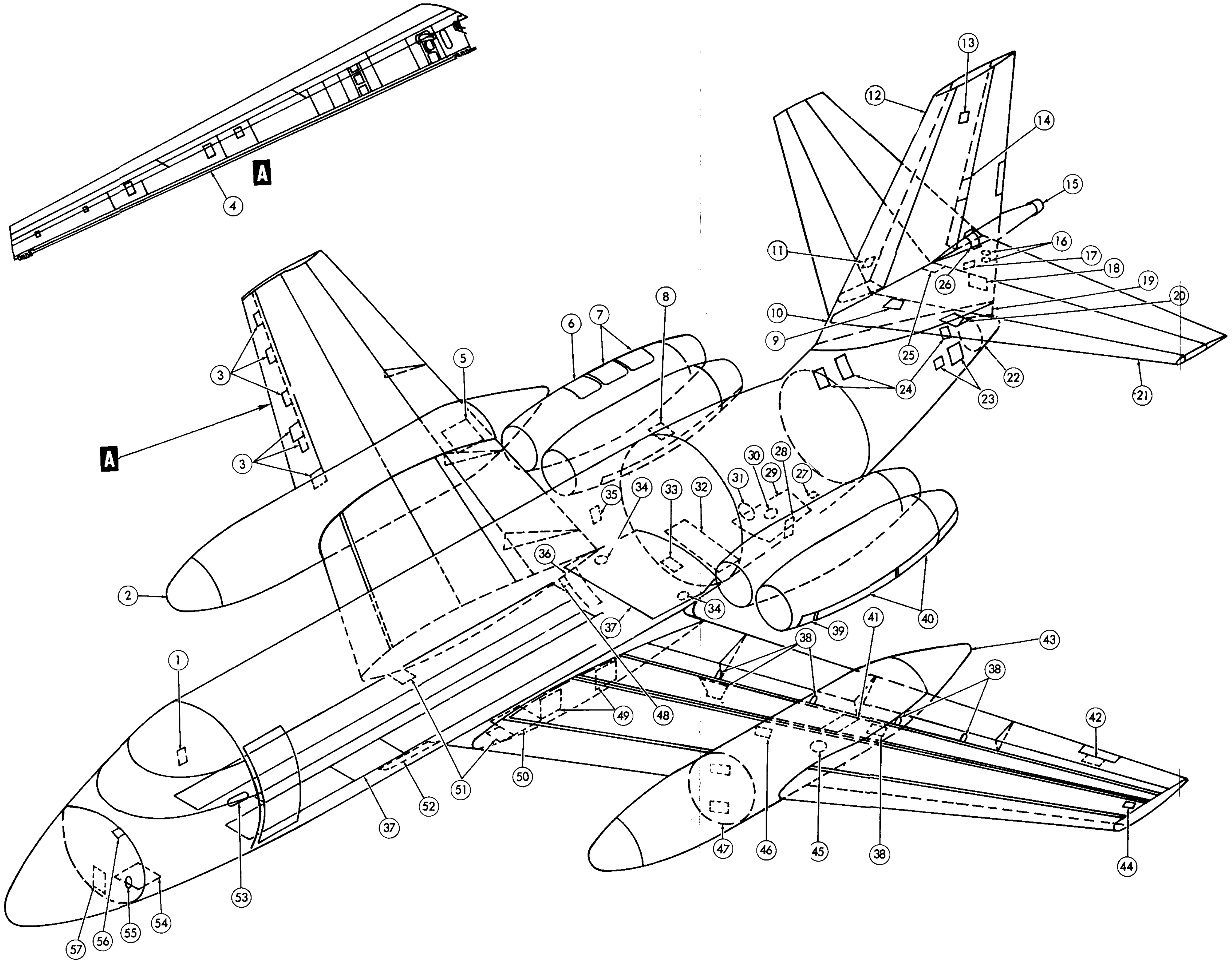


(57) PRESSURE BULKHEAD PLATE

1. SCREW NAS1633-3 QTY 13
2. SCREW NAS1633-5 QTY 3

JS-2-2-5205-20

Figure 52-5. Access and Inspection Provisions (Sheet 21 of 22)



1. EXTERNAL ELECTRICAL POWER RECEPTACLE DOOR
2. VENT VALVE, FUEL QUANTITY PROBE, LANDING AND TAXI LIGHTS ACCESS
3. LEADING EDGE FLAP ACCESS DOORS (AIRPLANES 5001 THROUGH 5033)
4. LEADING EDGE FLAP ACCESS DOORS (AIRPLANES 5034 AND UP)
5. VENT VALVE ACCESS (FLAPS DOWN)
6. NACELLE BOX BEAM ACCESS
7. ACCESS FOR INSPECTION, INSTALLATION, AND HOISTING, FIXED NACELLE
8. NACELLE BOX BEAM INBOARD LOWER ACCESS
9. HORIZONTAL STABILIZER LEADING EDGE ACCESS DOOR
10. VERTICAL STABILIZER LEADING EDGE, LOWER
11. VERTICAL STABILIZER LEADING EDGE ACCESS DOOR (RH SIDE ONLY)
12. VERTICAL STABILIZER LEADING EDGE, UPPER
13. VERTICAL STABILIZER BOX BEAM ACCESS (LH SIDE ONLY)
14. RUDDER TAB MECHANISM ACCESS (BOTH SIDES)
15. TRAILING EDGE FAIRING INSTALLATION
16. ELEVATOR TORQUE TUBE ACCESS DOOR
17. TRAILING EDGE FAIRING DOOR (5001 THRU 5105)
18. HORIZONTAL STABILIZER AFT BOX BEAM ACCESS DOOR
19. VERTICAL STABILIZER LOWER FAIRING
20. ELEVATOR BOOSTER ACCESS DOOR
21. HORIZONTAL STABILIZER REMOVABLE LEADING EDGE
22. REMOVABLE TAIL CONE AND DRAG CHUTE DOORS
23. ELEVATOR BOOSTER ACCESS DOORS
24. STABILIZER ATTACHMENT ACCESS
25. HORIZONTAL STABILIZER FORWARD CENTER MANUFACTURING AND INSPECTION ACCESS DOORS
26. ELEVATOR AND RUDDER CONTROLS ACCESS DOOR
27. SPEED BRAKE GROUND CONTROL ACCESS
28. HYDRAULIC GIG CONNECTION ACCESS DOOR
29. SPEED BRAKE DOOR ACCESS TO AFT COMPARTMENT
30. SPEED BRAKE OPENING DOOR ASSEMBLY
31. SPEED BRAKE UPLOCK ACCESS DOOR
32. FUEL VALVES ACCESS DOOR
33. DEFUELING VALVE ACCESS DOOR
34. FUEL CHECK VALVES ACCESS
35. WATER RECEPTACLE ACCESS DOOR
36. AILERON BOOSTER AND TRAILING EDGE FLAP MOTOR ACCESS DOOR
37. REMOVABLE FLOOR PANELS
38. TRAILING EDGE ACCESS
39. NACELLE FORWARD LOWER ACCESS DOOR
40. NACELLE ACCESS DOORS
41. EXTERNAL WING TANK INNER ACCESS DOORS
42. AILERON TAB ACCESS
43. EXTERNAL TANK TAIL CAP FAIRING
44. WING OUTBOARD VENT VALVE INSPECTION DOOR
45. EXTERNAL TANK FUEL BOOST PUMP ACCESS
46. WING INBOARD VENT VALVE INSPECTION DOOR
47. EXTERNAL TANK BULKHEAD COVER PLATES
48. MAIN WHEEL WELL ACCESS COVER
49. WING ROOT RIB FUEL BOOST PUMP ACCESS
50. BELLY FAIRINGS
51. ROOT RIB AND LEADING EDGE FLAP MOTOR ACCESS PANELS
52. LOOP ANTENNA ACCESS COVER
53. NOSE WHEEL WELL HAND HOLE
54. NOSE SECTION LOWER ACCESS
55. NOSE WHEEL WELL ACCESS COVERS
56. NOSE CONE LATCH COVER
57. PRESSURE BULKHEAD PLATE

Figure 52-5. Access and Inspection Provisions (Sheet 22 of 22)

SECTION 52.4

ENTRANCE STAIRS

Contents Page

DESCRIPTION AND OPERATION 52-33
General 52-33
Airborne Ladder 52-33
Automatic Stair - Manual Control 52-34
Automatic Stair - Electrical Controls 52-34
Automatic Stair Operation 52-34
Manual Stair Operation 52-34

OPERATIONAL CHECKOUT 52-34
Automatic Stair Electrical Operation (Exterior) 52-34
Automatic Stair Electrical Operation (Interior) 52-35
Automatic Stair Manual Operation (Interior) 52-35
Automatic Stair Manual Operation (Exterior) 52-35
Manual Stair Operation (Interior) 52-36
Manual Stair Operation (Exterior) 52-36

TROUBLESHOOTING 52-36
Automatic Stair 52-36
Manual Stair 52-39

REMOVAL AND INSTALLATION 52-40
Removal of Automatic Stair and Gearbox Assembly 52-40
Installation of the Automatic Stair and Gearbox Assembly 52-40
Removal of Manual Stair and Gearbox Assembly 52-40
Installation of Manual Stair and Gearbox Assembly 52-40

ADJUSTMENT 52-40
Adjustment of the Stair Assembly 52-40
Adjustment of the Automatic Stair Gearbox Assembly 52-40
Adjustment of the Manual Stair Assembly 52-40
Adjustment of the Manual Stair Gearbox Assembly 52-40

FIGURES

52-6. Airborne Ladder (Airplanes 5001 through 5065) 52-41
52-7. Automatic Stair Schematic Diagram (5066 and Up) 52-42
52-8. Adjustment of the Stair Assembly (Airplanes 5066 and Up) 52-43
52-9. Adjustment of the Automatic Stair Gearbox Assembly (Airplanes 5066 through 5120) . . . 52-45
52-10. Adjustment of the Manual Stair Gearbox Assembly (Airplanes 5121 and Up) 52-47

52.4.1 DESCRIPTION.

52.4.1.1 GENERAL. (See figures 52-6 and 52-8.) Airplanes 5001 through 5065 were delivered with a manually operated airborne ladder. The ladder is attached to a track which is fastened to the airplane floor. When not in use the ladder is simultaneously folded and retracted and then moved aft on the track into an enclosure for stowage. An automatic stair installed on airplane 5066 through 5120 and on airplanes incorporating Service Bulletin 1329-195 is an electrically powered stair that can also be manually operated. It is attached to a track that is fastened to the airplane floor, and it can be operated from either inside or outside the airplane with control switches. By depressing the up switch the stair is automatically sequenced to simultaneously fold and retract and then to move aft into a locked stowed position. By depressing the down switch the sequence is reversed and the stair lowers itself. Airplanes 5121 and up are equipped with a manually operated stair. This stair operates manually exactly the same way as the automatic stair operates in the manual mode.

52.4.1.2 AIRBORNE LADDER. (See figure 52-6.) The airborne ladder consists of two folding stair halves, a handrail, two handrail stanchions, a track, a latch handle assembly, and two handles. Handle (7) is used to pull the ladder along the track, and handle (4) is used to extend or retract it. Once the ladder is in the

52.4.1.2 (Cont).

stowed position, the latch handle assembly (8) is used to lock the ladder.

52.4.1.3 AUTOMATIC STAIR MANUAL CONTROL. A pull control handle for the automatic stair (located on the aisle-way partition and connecting to the alignment pin solenoid) provides for manual unlocking of the alignment pin in the stair stowed or unstowed position. This allows the stair assembly to be pushed forward or aft manually.

52.4.1.4 AUTOMATIC STAIR ELECTRICAL CONTROLS. (See figures 52-7.) The electrical control system for the automatic stair consists of the following components:

a. A 24 VDC .12 HP series motor drive.

b. A two position ("UP," "DOWN") LADDER CONTROL interior switch (on the forward left side at fuselage station 232.0) and separate flush pushbutton ("UP, DOWN") ENTRANCE LADDER exterior switches located at the entrance door at fuselage station 274 and waterline 78. Either one of these switches will operate the stairs.

c. An up and a down motor control relay whose function is to provide correct motor rotation for the selected stair operation. The two relays are so wired that if the down motor control relay is energized the up control circuit is opened by the down motor control relay. The reverse is true for the up motor control relay.

d. Forward-aft and up-down clutch solenoid control relays give correct stair movement when energized by signals from the appropriate limit switches.

e. Four limit switches (up, down, forward, and aft) used to signal the appropriate relays.

f. An up control relay which permits correct sequencing of the solenoid relays during the selected up position.

g. A door aft limit switch which precludes inadvertent stair operation from either control switch when the door is not fully opened.

h. An alignment pin solenoid which automatically locks the stair when either in the aft position or in the forward position.

52.4.1.5 AUTOMATIC STAIR OPERATION. (See figure 52-9.) With the main entrance door opened, the stair is lowered through a sequence of operations by depressing either the interior "LADDER CONTROL-DOWN" switch or the exterior "ENTRANCE LADDER-DOWN" switch. The stair is raised by depressing the same switches "UP."

Note

> To prevent inadvertent stair operation when the main entrance door is not fully opened, the door aft limit switch must be actuated by the door when in the fully opened and stowed position before the down or up switches will function.

When the down switch is depressed, the alignment pin solenoid (13) energizes, releasing the alignment pin from the aft stop (19). This unlocks the stair from the stowed position. Simultaneously the two-way solenoid (5) is energized. This engages the lower cone clutch (15), driving the retracted stair forward on the track (20) by means of a rack and pinion gear. As the stair assembly moves forward, the aft trip (18) actuates the aft limit switch (9), de-energizing the alignment pin solenoid (13) and causing the spring-loaded alignment pin to ride on the track (20). When the full forward position is reached, the alignment pin, because of spring pressure, drops into the hole of the alignment pin stop (16), positioning the stair for extension. Simultaneously with the action of the pin, the forward trip (21) actuates the forward limit switch (10), energizing the two-way solenoid (5). The solenoid engages the upper cone clutch (15), driving the actuator arm (3) to extend the retracted stair assembly to the down or entrance position. To raise the stair assembly, the reverse of the above procedure takes place when the exterior or interior "UP" switch is depressed.

Note

> When operating the stair by using the outside switch a ground through the switch is supplied to energize the airplane "BAT REL." This supplies battery power to the essential bus to allow stair operation. When operating the stair (using the inside switch) when ground power is not available, position the BAT switch to "BAT." This powers the essential bus with battery power. Once the inside switch is depressed and a selected sequence starts, the outside switch cannot interrupt this cycle. The reverse holds true when the outside switch is depressed first.

52.4.1.6 MANUAL STAIR OPERATION. (See figure 52-8.) With the entrance door opened and stowed aft, the folded stair can be pulled forward from the stowed position to the forward stops and then lowered. The stair can be raised, folded, and stowed by a sequence of motions described in paragraphs 52.4.2.5 and 52.4.2.6. All operations can be accomplished from inside or outside the aircraft.

52.4.2 OPERATIONAL CHECKOUT. A checkout for both the electrical and manual stair follows:

52.4.2.1 AUTOMATIC STAIR ELECTRICAL OPERATION (EXTERIOR). Perform the following checkout from outside the airplane.

a. Engage the AUTO STAIRS - CONTROL, MOTOR, and CLUTCH circuit breakers.

b. Turn the BAT switch to "OFF."

c. Open the entrance door and stow fully aft.

d. Depress the "ENTRANCE LADDER - DOWN" exterior switch until the down cycle is complete.
RESULT: The folded stair should travel forward in approximately seven seconds and extend itself to full down position in approximately five seconds.

e. Depress the "ENTRANCE LADDER - UP" exterior switch until the up cycle is complete.
RESULT: The stair should fully fold into the door opening in approximately seven seconds and travel aft to its stowed position in approximately seven seconds.

f. Move the door forward a distance of one to two inches from the stowed position and depress the "ENTRANCE LADDER - DOWN" exterior switch.
RESULT: The folded stair should not travel forward. This indicates that the aft limit switch is functioning properly.

52.4.2.2 AUTOMATIC STAIR ELECTRICAL OPERATION (INTERIOR). Perform the following checkout from inside the airplane.

a. With the AUTO STAIRS circuit breaker still engaged, turn the BAT switch to "ON."

b. Open the entrance door, and stow fully aft.

c. Depress the "LADDER CONTROL - DOWN" interior switch until the cycle is complete.
RESULT: The folded stair should travel forward and extend to the ramp.

d. Depress the "LADDER CONTROL - UP" interior switch until the cycle is complete.
RESULT: The stair should fold into the door opening and stow aft.

e. Repeat step (f), paragraph 52.4.2.1.

f. Disengage the AUTO STAIRS - CONTROL, MOTOR, and CLUTCH circuit breakers, and turn the BAT switch to "OFF."

52.4.2.3 AUTOMATIC STAIR MANUAL OPERATION (INTERIOR). Perform the following checkout from inside the airplane.

a. Open the entrance door and stow it fully aft.

b. Pull the solenoid alignment pin control handle to release the folded stair from the aft stowed position, and move the stair on the track to the most forward position.
RESULT: The solenoid alignment pin should fall into the forward hole.

c. Grasp the stair handrails near the ends and pull gently inboard while simultaneously pushing outboard with a foot or knee on a stair tread or riser. Hold onto the handrails, and gently lower the stair to the ramp.
RESULT: The stair should descend without evidence of binding.

d. To retract the stair, grasp the handrails and pull inboard. Just before the stair is fully folded, release the handrails and fully retract stairs by pulling on the top stair riser.
RESULT: The stair should retract without evidence of binding.

e. Pull the solenoid pin control handle to release the stair from the forward position, and move the stair aft on the track to the stowed position.
RESULT: The solenoid alignment pin should fall into the aft hole.

52.4.2.4 AUTOMATIC STAIR MANUAL OPERATION (EXTERIOR). Perform the following checkout from outside the airplane.

a. Open the entrance door, and stow fully aft.

b. With the stair in the forward position, pull outboard on the aft rear wheel or siderail (not the handrail) until the stair unfolds and starts to descend. Gently lower the stair to the ramp with the right hand placed approximately at the folding point.
RESULT: The stair should descend without evidence of binding.

c. To retract the stair, lift on the aft siderail (not the handrail) slightly up from the folding point until the stair is approximately one-half folded. Grasp the aft rear wheel with the left hand, and simultaneously lift and push inboard until the stair is fully folded.
RESULT: The stair should retract without evidence of binding.

Note

To stow the folded stair aft, reach into the airplane and pull the solenoid pin control arm to release the stair from the forward position, and move the stair aft on the track to the stowed position.

52.4.2.5 MANUAL STAIR OPERATION (INTERIOR). Perform the following checkout from inside the airplane.

a. Open the entrance door and stow it fully aft.

b. Pull the alignment pin control handle to release the folded stair from the aft stowed position, and move the stair on the track to the most forward position.
RESULT: The alignment pin should fall into the forward hole.

c. Grasp the stair handrails near the ends and pull gently inboard while simultaneously pushing outboard with a foot or knee on a stair tread or riser. Hold onto the handrails, and gently lower the stair to the ramp.
RESULT: The stair should descend without evidence of binding.

d. To retract the stair, grasp the handrails and pull inboard. Just before the stair is fully folded, release the handrails and fully retract stairs by pulling on the top stair riser.
RESULT: The stair should retract without evidence of binding.

e. Pull the alignment pin control handle to release the stair from the forward position, and move the stair aft on the track to the stowed position.
RESULT: The alignment pin should fall into the aft hole.

52.4.2.6 MANUAL STAIR OPERATION (EXTERIOR). Perform the following checkout from outside the airplane.

a. Open the entrance door, and stow fully aft.

b. With the stair in the forward position, pull outboard on the aft rear wheel or siderail (not the handrail) until the stair unfolds and starts to descend. Gently lower the stair to the ramp with the right hand placed approximately at the folding point.
RESULT: The stair should descend without evidence of binding.

c. To retract the stair, lift on the aft siderail (not the handrail) slightly up from the folding point until the stair is approximately one-half folded. Grasp the aft rear wheel with the left hand, and simultaneously lift and push inboard until the stair is fully folded.
RESULT: The stair should retract without evidence of binding.

Note

To stow the folded stair aft, reach into the airplane and pull the alignment pin control handle to release the stair from the forward position, and move the stair aft on the track to the stowed position.

52.4.3 TROUBLESHOOTING.

52.4.3.1 AUTOMATIC STAIR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MOTOR DOES NOT OPERATE WHEN THE CONTROL SWITCH IS ACTUATED IN THE DOWN POSITION.		
Circuit breaker not in.	Inspect circuit breaker panel.	Reset circuit breaker.
Entrance door not fully open.	Door structure and door should be at approximately the same fuselage station.	Push door aft against door stop.
Entrance door safety switch inoperative.	Run continuity check on switch.	Replace defective switch.
Control wiring opened.	Run continuity check on wiring.	Repair open circuit.
Control switches defective.	Run continuity check on switches.	Replace defective switches.
Motor defective.	Remove plug on top of gearbox assembly, and check motor operation with 24 VDC power source.	Replace defective motor assembly.
Motor control relay defective.	Run complete continuity check on relay.	Replace defective relay.

52.4.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MOTOR OPERATES BUT STAIR FAILS TO MOVE OUT OF THE STOWED POSITION.		
Alignment pin solenoid not energized.	Run continuity check on solenoid.	Replace defective solenoid.
Aft limit switch defective or misaligned.	Run continuity check on switch. Check alignment and location of trip and adjustment of limit switch.	Replace defective limit switch. Add or delete shims as required.
Forward-aft solenoid inoperative.	Run continuity check on solenoid.	Replace defective solenoid.
Alignment pin solenoid relay inoperative.	Run continuity check on relay.	Replace defective relay.
Forward-aft control relay defective.	Run continuity check on relay.	Replace defective relay.
Forward-aft clutch inoperative.	Push clevis on end of solenoid to check up and down movement. Total movement should not be over 0.10 inch.	Eliminate excessive up and down motion by removing laminations from washer between bearing block and clutch. Refer to Step 7 of figure 52-9.
Gear teeth failure.	Remove clutch housing box cover, and check for rotation of male cone clutch. If no rotation occurs, speed reducer gears may be stripped. Remove motor assembly from speed reducer box, and check gear teeth.	Replace damaged gears.
Motor shaft failure or brake failure.	Remove motor assembly from speed reducer, and check for shaft rotation.	Replace defective motor assembly.
STAIR MOVES TO FULL FORWARD POSITION BUT FAILS TO GO DOWN.		
Up-down relay defective.	Run continuity check on relay.	Replace defective relay.
Up-down solenoid defective.	Run continuity check on relay.	Replace defective relay.
Open control wiring.	Run continuity check.	Repair open circuit.
Forward-aft or up-down limit switch defective or misaligned.	Run continuity check on switches. Check location of forward trip and adjustment of actuators on limit switches.	Replace defective switches. Refer to figure 52-9.
Up-down clutch inoperative.	Remove clutch housing box cover, and check for rotation of upper female clutch and bevel gears.	Refer to Step 6 and Step 7 of figure 52-9.

52.4.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ACTUATOR ARM ON GEARBOX CONTINUES TO DRIVE IN DOWN POSITION.		
Down limit switch defective or misaligned.	Run continuity check on switch. Switch misaligned or mislocated.	Replace defective switch. Refer to Step 3 of figure 52-9.
Up-down solenoid defective.	Run continuity check on solenoid.	Replace defective solenoid.
Up-down cone clutches stuck together.	Remove clutch housing box cover, and check for jammed actuator arm assembly or rough cone clutch faces.	Remove jam condition. Disassemble cone clutch box assembly components, and polish faces of male and female cones with 300 grit emery cloth or equal.
MOTOR DOES NOT OPERATE WHEN THE CONTROL SWITCH IS ACTUATED TO THE UP POSITION.		
Circuit breaker not in.	Inspect circuit breaker panel.	Reset circuit breakers.
Up-down motor relay defective.	Run continuity check on relay.	Replace defective relay.
Control wire open.	Run continuity check on wiring.	Repair open circuit.
Motor defective.	Remove plug on top of gearbox assembly, and check motor operation with 24VDC power source.	Replace defective motor.
Control switches defective.	Run continuity check on switches.	Replace defective switches.
MOTOR OPERATES BUT STAIR FAILS TO MOVE OUT OF THE DOWN POSITION.		
Control wiring open.	Run continuity check on wires.	Repair open circuit.
Up-down relay defective.	Run continuity check on relay.	Replace defective relay.
Up limit switch defective, inoperative, or misaligned.	Run continuity check on switch. Switch misaligned or mislocated.	Replace defective switch. Refer to Step 2 of figure 52-9.
Up clutch inoperative.	Remove clutch housing box cover, and check for rotation of upper female clutch and bevel gears.	Refer to Step 6 and Step 7 of figure 52-9.
ALIGNMENT PIN SOLENOID DOES NOT LOCK STAIR IN THE STOWED OR IN THE FORWARD POSITION.		
Tension spring on manual release arm broken.	Check spring.	Replace broken or defective spring.
Hole in stop does not align with solenoid pin.	Manually move release arm of solenoid pin. Pin should engage and disengage without binding or grabbing.	Enlarge hole in stop or move stop to align hole with solenoid pin.

52.4.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
MOTOR CONTINUES TO DRIVE IN STOWED POSITION.		
Aft limit switch defective.	Run continuity check on switch.	Replace defective switch.
Forward-aft solenoid defective.	Run continuity check on solenoid.	Replace defective solenoid.
Forward-aft clutch defective.	Check for stuck clutch.	Disassemble cone clutch box assembly components, and polish faces of male and female cones with 300 grit emery cloth or equal.
Aft limit switch trip mislocated.	Check for switch not tripped.	Refer to Step 3 of figure 52-9.
Aft limit switch actuator arm not set correctly.	Check for switch not tripped.	Refer to Step 3 of figure 52-9.

52.4.3.2 MANUAL STAIR.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ALIGNMENT PIN DOES NOT LOCK STAIR IN THE STOWED OR IN THE FORWARD POSITION.		
Holes in stops do not align with pin.	Hold gear box assembly hard against stops. Pin should drop freely into holes.	Position stops correctly and/or enlarge holes so that alignment pin will drop freely into holes.
STAIR DIFFICULT TO RAISE AND LOWER.		
Power spring not adjusted properly.	Retract stair and remove bolt connecting actuator arm and crank assembly. Spring should hold actuator arm against stop.	Remove actuator arm from splined shaft, and rotate shaft clockwise to wind up spring. Reinstall arm. Arm should exert 1 to 5 pounds force against the actuator arm stop. Replace bolt through arm and crank.
Power spring broken.	Retract stair and remove bolt as above, move arm outboard. Spring is broken if arm does not return to up position.	Replace spring.
STAIR WILL NOT UNFOLD FROM DOWN POSITION.		
Catch (see 9, figure 52-8) not releasing.	Check tension in cable. (Refer to figure 52-8.)	Readjust cable tension.
Bolt and crank not engaging properly. (Refer to detail C, figure 52-8.)	Check .05/.07 dimension.	Readjust per step 4 of figure 52-8.

52.4.3.2 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FOLDED STAIR DIFFICULT TO SLIDE FORWARD AND AFT.		
Trolleys, ball bearings, and tracks have grit in bearing ways.	Inspect for lubrication.	Remove all traces of lubrication (except for solid film lubrication) from trolleys, balls, and track. Do not lubricate.
Metal burrs interfering with ball bearings.	Inspect track and trolleys for burrs.	Remove all burrs from trolleys and tracks.
HAND RAILS NOT RIGID IN STAIR DOWN POSITION.		
87° to 90° angle of post assemblies not met. (Refer to figure 52-8.)	With stairs in down position, check the 87° to 90° angle.	Readjust per figure 52-8. Angle of post assemblies with side rails may be less than 87° if required.

52.4.4 REMOVAL AND INSTALLATION.

52.4.4.1 REMOVAL OF AUTOMATIC STAIR AND GEARBOX ASSEMBLY.

(See figure 52-9.)

a. Open and stow the entrance door aft.

b. Move the folded stair on the track to the forward position.

CAUTION

Never remove the bolt in step c with the stair in the down position. Lowering the stair winds up the power spring, and releasing the actuator arm suddenly can cause damage to the gear and gearbox and can result in serious injury to personnel.

c. Remove the bolt connecting the actuator arm (3) to the crank on the stair.

d. Remove the two bolts attaching the stair to the trolleys, and remove the folded stair assembly from the airplane.

e. Remove the screws attaching the track to the floor.

f. Disconnect the control wire from the alignment pin solenoid (13), and disconnect the two electrical plugs on top of the gearbox.

g. Remove the gearbox, track, and trolleys from the airplane as a unit. Do not allow the trolleys to fall off the end of the track.

52.4.4.2 INSTALLATION OF THE AUTOMATIC STAIR AND GEARBOX ASSEMBLY. Reverse steps (c) through (g) in the preceding paragraph.

52.4.4.3 REMOVAL OF THE MANUAL STAIR AND GEAR BOX ASSEMBLY.

(See figure 52-10.)

a. Open and stow the entrance door aft.

b. Move the folded stair on the track to the forward position.

CAUTION

Never remove the bolt in step c with the stair in the down position. Lowering the stair winds up the power spring, and releasing the actuator arm suddenly can cause damage to the gear and gearbox and can result in serious injury to personnel.

c. Remove the bolt connecting the actuator arm (9) to the crank on the stair.

d. Remove the two bolts attaching the stair to the trolleys, and remove the folded stair assembly from the airplane.

e. Remove the screws attaching the track to the floor.

f. Remove the gearbox, track, and trolleys from the airplane as a unit. Do not allow the trolleys to fall off the end of the track.

52.4.4.4 INSTALLATION OF MANUAL STAIR AND GEAR BOX ASSEMBLY. Reverse steps (c) through (f) in the preceding paragraph.

52.4.5 ADJUSTMENT.

52.4.5.1 ADJUSTMENT OF THE STAIR ASSEMBLY.

(Refer to figure 52-8.)

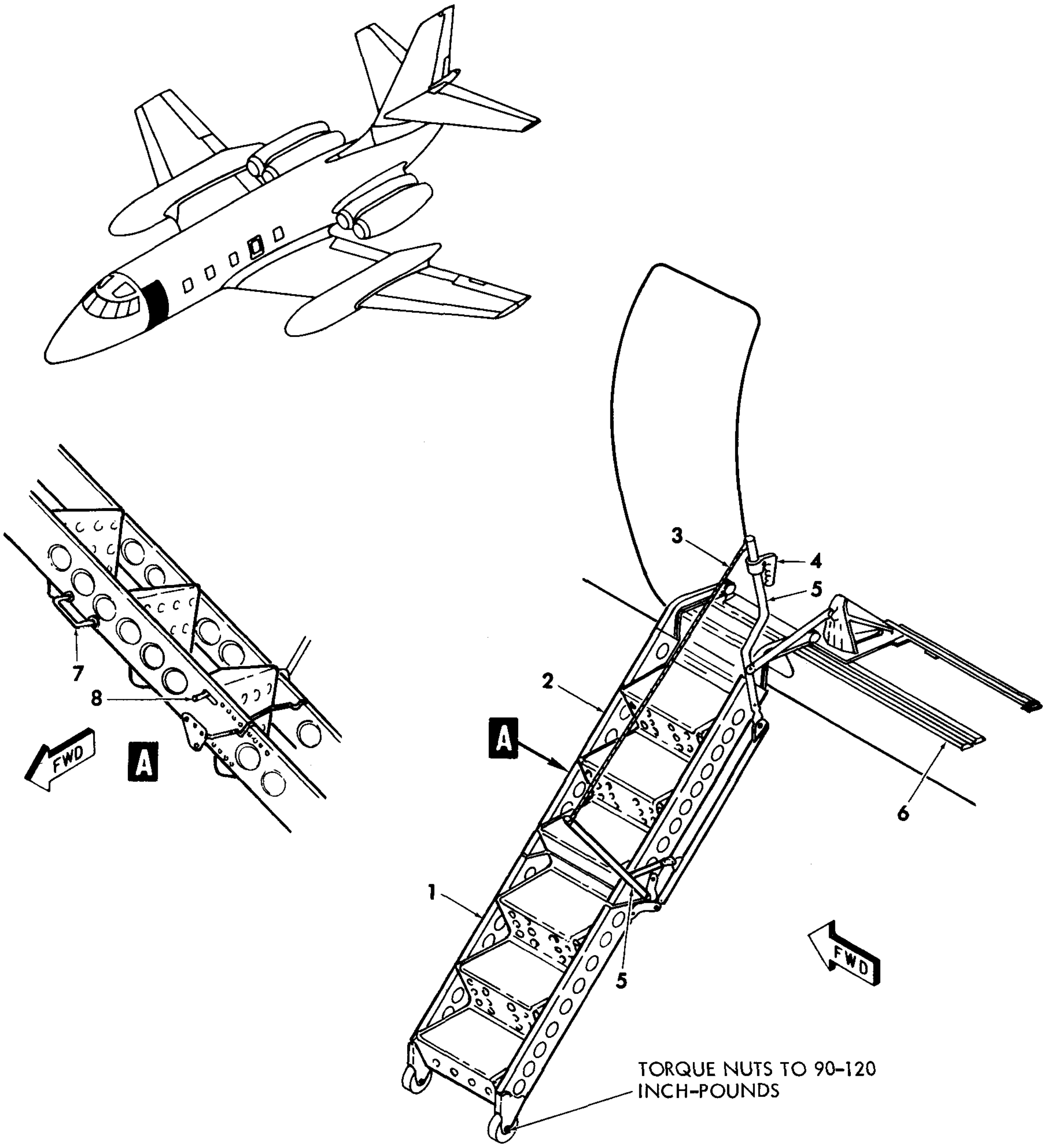
52.4.5.2 ADJUSTMENT OF THE AUTOMATIC STAIR GEARBOX ASSEMBLY.

(Refer to figure 52-9.)

52.4.5.3 ADJUSTMENT OF THE MANUAL STAIR ASSEMBLY. The adjustment of the manual stair assembly is identical to the adjustment of the automatic stair assemble. Refer to figure 52-8 for the step by step procedure.

52.4.5.4 ADJUSTMENT OF THE MANUAL STAIR GEARBOX ASSEMBLY.

(Refer to figure 52-10.)



TORQUE NUTS TO 90-120 INCH-POUNDS

1. STEP ASSEMBLY LOWER
2. STEP ASSEMBLY UPPER
3. HAND RAIL
4. HANDLE
5. STANCHION
6. TRACK
7. HANDLE
8. LATCH HANDLE ASSEMBLY

Figure 52-6. Airborn Ladder (Airplanes 5001 thru 5065)

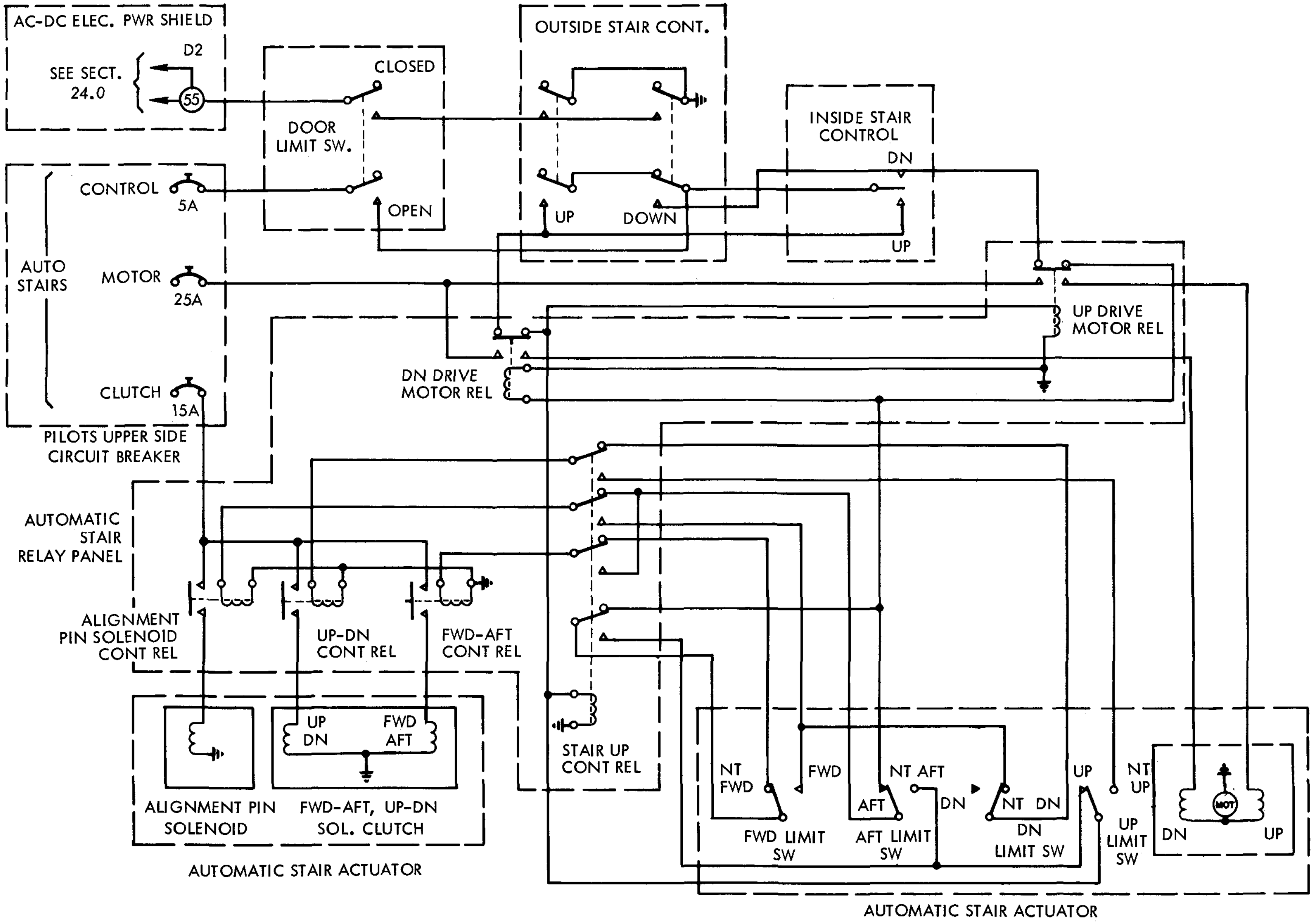
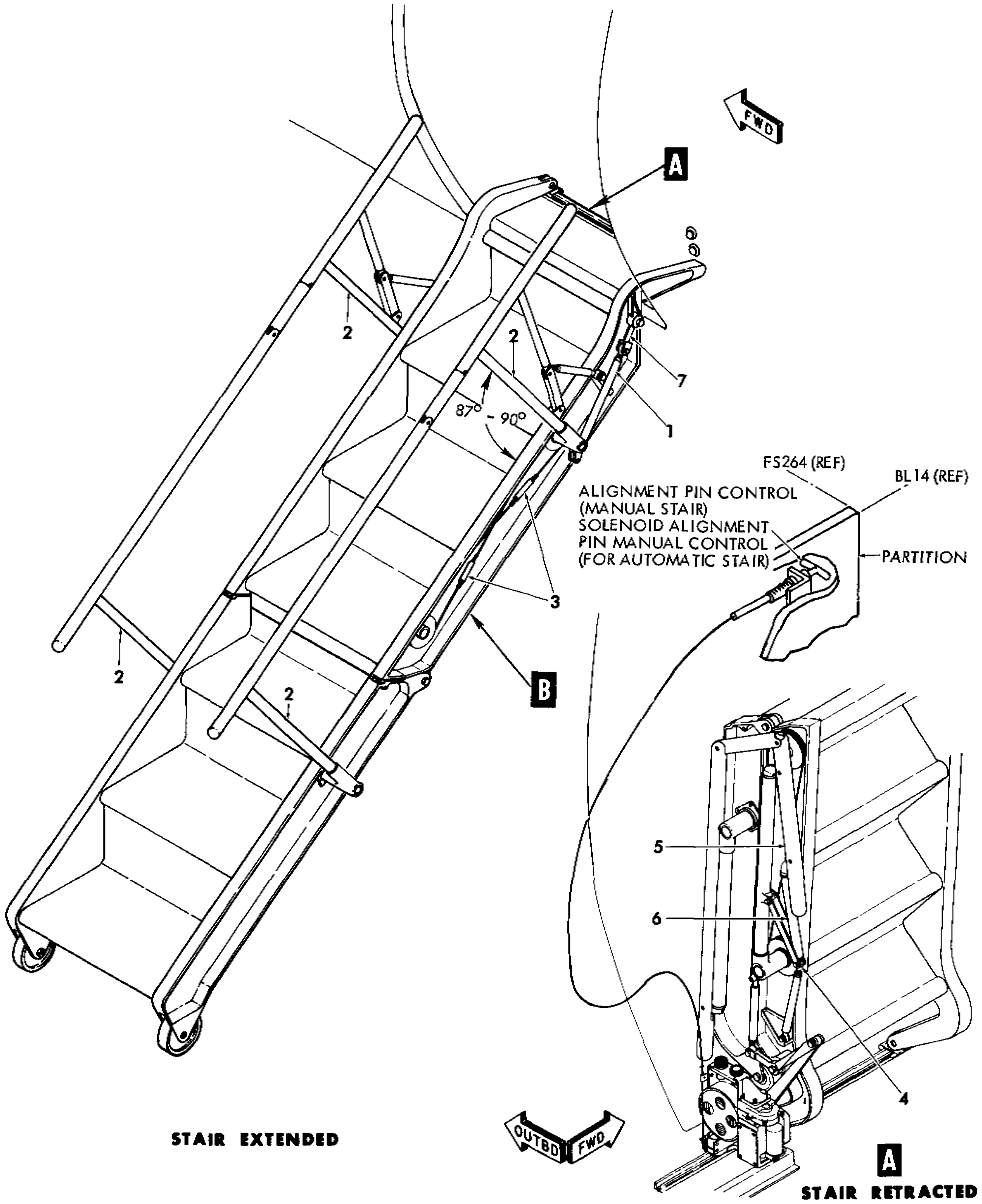


Figure 52-7. Automatic Stair Schematic Diagram (Airplanes 5066 thru 5120)


FWD
A
2
2
2
2
87° - 90°
7
1
3
B
FS264 (REF)
BL14 (REF)
ALIGNMENT PIN CONTROL (MANUAL STAIR)
SOLENOID ALIGNMENT PIN MANUAL CONTROL (FOR AUTOMATIC STAIR)
PARTITION
5
6
4
OUTBD
FWD
A

STAIR EXTENDED

STAIR RETRACTED

(AUTOMATIC STAIR GEAR BOX SHOWN -MANUAL GEAR BOX SIMILIAR)

STEP 1

WITH THE STAIR IN THE FULLY EXTENDED POSITION, DISCONNECT THE PUSH-PULL ROD (1) FROM THE CRANK ASSEMBLY (7).

STEP 2

ADJUST THE TURNBUCKLE BARRELS (3) TWO PLACES ON EACH SIDE OF THE STAIR UNTIL THE POST ASSEMBLIES (2) ARE AT 87° TO 90° AS SHOWN. MAKE FINAL ADJUSTMENT TO OBTAIN 200 ± 10 POUNDS TENSION IN THE CABLES AND RESAFETY BARRELS WITH MS21256-1 CLIPS.

STEP 3

SET THE ADJUSTABLE END OF THE CRANK ASSEMBLY (7) TO THE 2.90 ± .03 DIMENSION AND ADJUST THE END OF THE PUSH ROD (1) TO THIS LENGTH. RECONNECT THE ROD AND CRANK.

NOTE

IF THE DOWN LATCH ASSEMBLY IS INSTALLED, ACCOMPLISH STEPS 4 AND 5; IF NOT INSTALLED, SKIP TO STEP 6.

STEP 4

CHECK THE .05/ .07 DIMENSION IN DETAIL C. FILE OR SHIM AS REQUIRED TO OBTAIN THIS DIMENSION AND RESET THE TURNBUCKLE BARREL (8) IF NECESSARY.

STEP 5

CHECK THE CATCH (9) FOR FULL ENGAGEMENT WITH THE BOLT. IF NOT FULLY ENGAGED, ADJUST THE TURNBUCKLE BARREL (8) TO PROVIDE FULL ENGAGEMENT OF THE CATCH WITH THE BOLT BUT STILL MAINTAIN ENOUGH TENSION IN THE CABLE TO BALANCE THE LOAD IN SPRING (10).

STEP 6

MANUALLY RETRACT THE STAIRS AND DISCONNECT THE PUSH-PULL ROD (1) FROM THE CRANK ASSEMBLY (7). CHECK THE ALIGNMENT OF THE HOLE THRU (1) AND (7). IF THE HOLES DO NOT ALIGN, ADJUST THE PUSH-PULL ROD END TO AGREE WITH THE HOLE IN (7), AND RECONNECT (1) AND (7).

STEP 7

MANUALLY LOWER THE STAIRS AND DISCONNECT (1) FROM (7). CHECK THE ALIGNMENT OF THE HOLES THRU (1) AND (7). IF THE HOLES DO NOT ALIGN, LOOSEN THE TWO BOLTS THRU (7) AND ADJUST THE END TO MATCH THE HOLE IN (1). TIGHTEN THE CLAMP NUTS TO 90-100 POUND-INCH TORQUE AND RECONNECT (1) AND (7).

STEP 8

RECHECK STEPS 4 AND 5. READJUST AS REQUIRED.

NOTE

INCORRECT SETTING OF THE .05/ .07 DIMENSION IN DETAIL C WILL RESULT IN THE CATCH (9) NOT RELEASING WHEN THE STAIR IS RETRACTED. CAREFUL ADJUSTMENT OF THIS DIMENSION IS REQUIRED OR DAMAGE TO THE STAIR AND MECHANISM WILL OCCUR IF THE CATCH (9) DOES NOT RELEASE.

STEP 9

RETRACT AND EXTEND THE STAIRS THRU SEVERAL COMPLETE CYCLES AND RECHECK STEPS 2, 4 AND 5. READJUST AS REQUIRED.

STEP 10

WITH THE STAIR FULLY RETRACTED, ADJUST THE ROD AT (4) ON BOTH SIDES OF THE STAIRS UNTIL THE HAND RAIL (5) IS APPORXIMATELY PARALLEL TO THE ROD ASSEMBLY (6) . RESAFETY ALL PLACES AS REQUIRED

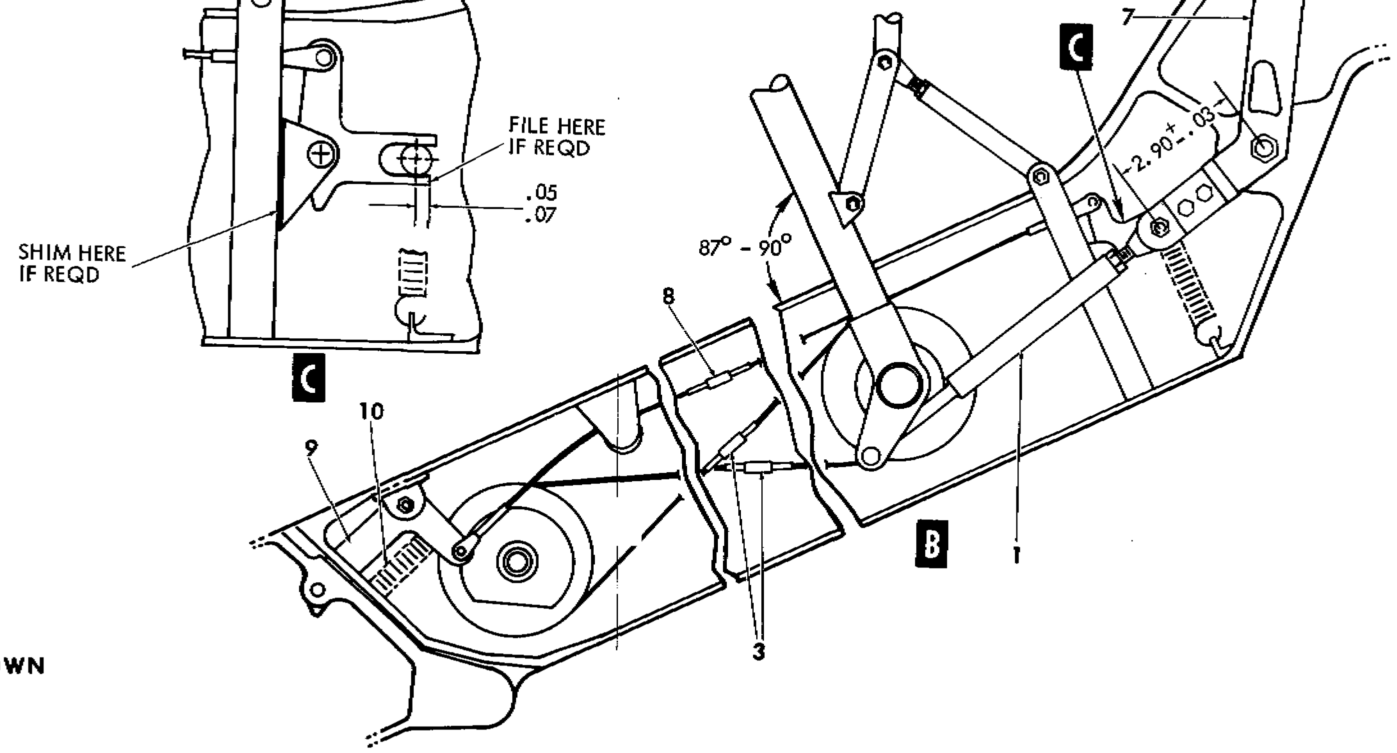
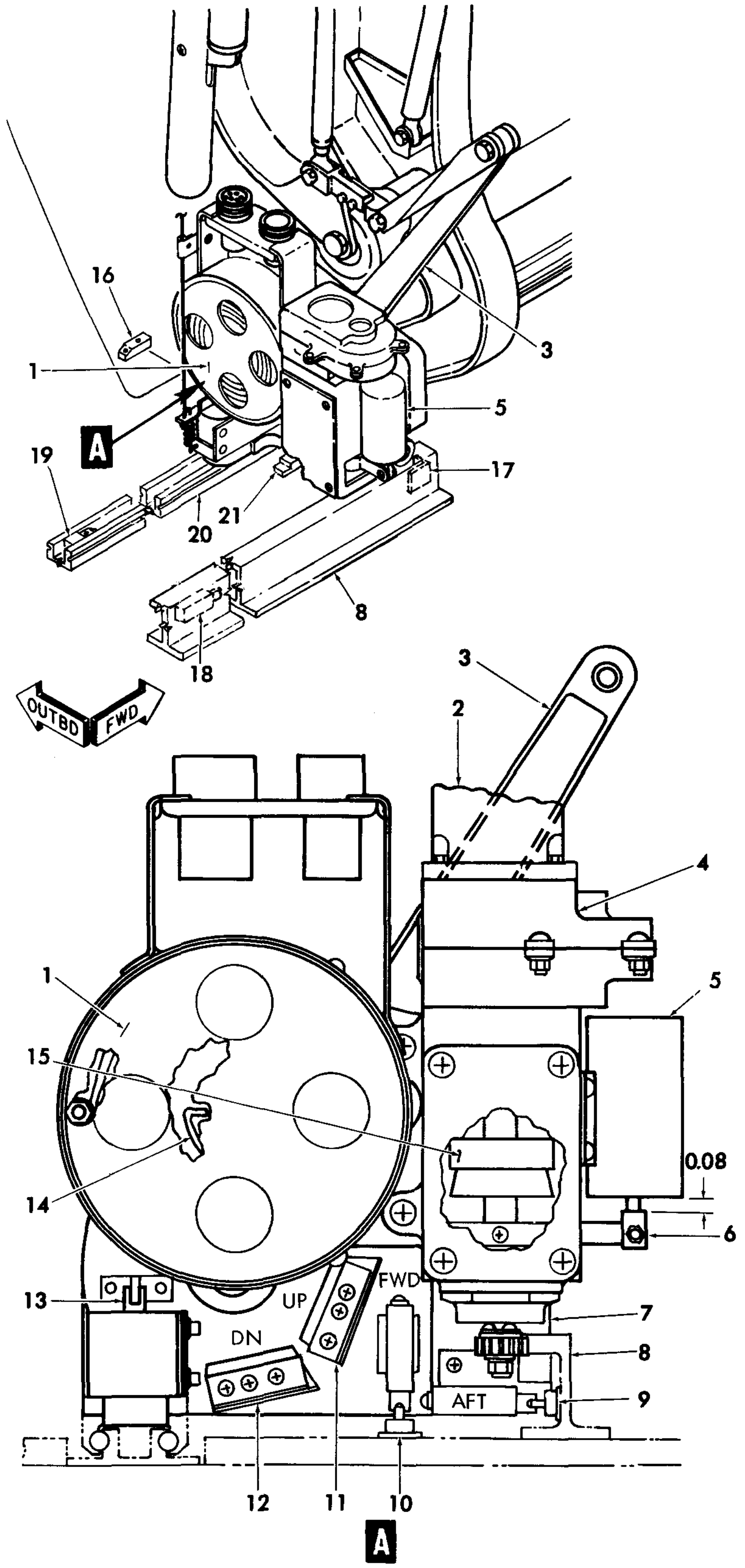


Figure 52-8. Adjustment of the Automatic and Manual Stair Assembly (Airplanes 5066 and Up)



1. POWER SPRING COVER
2. MOTOR
3. ACTUATOR ARM
4. SPEED REDUCER
5. TWO WAY SOLENOID
6. TWO WAY SOLENOID CLEVIS AND EXTENSION
7. FEMALE CLUTCH BEARING ASSEMBLY
8. RACK
9. AFT LIMIT SWITCH
10. FORWARD LIMIT SWITCH
11. UP LIMIT SWITCH
12. DOWN LIMIT SWITCH
13. ALIGNMENT PIN SOLENOID
14. MANDREL
15. CONE CLUTCHES
16. ALIGNMENT PIN STOP
17. FRONT STOP
18. AFT TRIP
19. AFT STOP
20. TRACK
21. FWD TRIP

STEP 1

ADJUST THE TENSION IN THE POWER SPRING AS FOLLOWS:

A. POSITION THE ACTUATOR ARM (3) IN THE FULLY RETRACTED (STAIR UP) POSITION.
B. REMOVE THE POWER SPRING COVER (1) AND PULL THE MANDREL (14) AFT UNTIL THE SPLINES DISENGAGES.
C. ROTATE THE MANDREL (14) WITH THE POWER SPRING END JUST ENGAGING THE SLOT IN THE MANDREL UNTIL THE ACTUATOR ARM (3) EXERTS 1 TO 5 POUNDS FORCE AGAINST THE ACTUTATOR ARM STOP.
D. SLIDE THE MANDREL (14) ON THE SHAFT IN THE NEAREST SPLINE ENGAGEMENT AND REINSTALL (1).

STEP 2

ADJUST THE UP LIMIT SWITCH (11) AS FOLLOWS: LOOSEN THE ATTACH BOLTS AND POSITION THE LIMIT SWITCH SO THAT THE CAM ON THE GEAR IN THE GEAR BOX WILL ACTUATE THE SWITCH WHEN THE ACTUATOR ARM (3) IS 0.18 TO 0.24 INCHES FROM THE STOP OR FULLY RETRACTED POSITION. TIGHTEN THE ATTACHING SCREWS AND CHECK THE OPERATION OF THE SWITCH BY MOVING THE ACTUATOR ARM THROUGH SEVERAL CYCLES.

NOTE

THE LIMIT SWITCH CAM ROLLER WHEN NOT ENGAGED TO THE CAM SHOULD JUST CONTACT THE SIDE OF THE GEAR. 0.02 INCH MOVEMENT OF THE ROLLER WILL ACTUATE THE SWITCH.

STEP 3

ADJUST THE DOWN LIMIT SWITCH (12) AS FOLLOWS:

POSITION AND SUPPORT THE STAIR END 12.0 INCHES FROM THE RAMP. LOOSEN THE ATTACHING SCREWS AND POSITION THE SWITCH SO THAT THE CAM ROLLER ON THE SWITCH JUST CONTACTS THE CAM ON THE GEAR. TIGHTEN THE ATTACHING SCREWS, REMOVE THE SUPPORT AND CYCLE THE STAIRS SEVERAL TIMES TO ASCERTAIN THE CORRECT POSITION OF THE LIMIT SWITCH.

NOTE

THE DOWN LIMIT SWITCH MUST ACTUATE WHEN THE STAIR END IS 9.0 TO 15.0 INCHES FROM THE RAMP AND THE LANDING GEARS ARE IN THE STATIC POSITION (2.5 INCH STRUT EXTENSION FOR THE NOSE GEAR AND 3.5 INCH FOR THE MAIN GEAR FROM THE FULLY COMPRESSED POSITION).

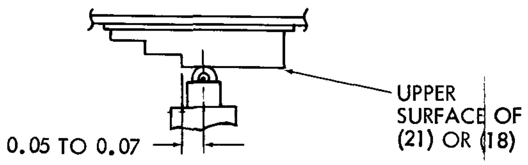
NOTE

THE LIMIT SWITCH CAM ROLLER WHEN NOT ENGAGED TO THE CAM SHOULD JUST CONTACT THE SIDE OF THE GEAR. 0.02 INCH MOVEMENT OF THE ROLLER WILL ACTUATE THE SWITCH.

STEP 4

ADJUST THE AFT LIMIT SWITCH (9) AS FOLLOWS:

LOOSEN THE ATTACHING SCREWS OF THE TRIP (18) LOCATED ON THE INSIDE OF THE RACK (8). POSITION THE TRIP SO THAT THE LIMIT SIWTCH ROLLER IS 0.05 TO 0.07 AFT ON THE UPPER SURFACE WHEN THE GEAR BOX ASSEMBLY IS AGAINST THE AFT STOP (19) AND THE ALIGNMENT SOLENOID PIN IS IN THE HOLE.



NOTE

THE LIMIT SWITCH MUST NOT ACTUATE UNTIL THE ROLLER REACHES THE UPPER SURFACE OF THE ADJUSTABLE TRIP. SHIM UNDER THE TRIPS AS REQUIRED TO ACCOMPLISH THIS. WHEN RETURNING, THE SWITCH WILL NOT DE-ACTUATE UNTIL THE ROLLER DESCENDS TO THE SECOND OR FIRST STEPS. THIS DELAY ALLOWS THE ALIGNMENT SOLENOID PIN TO CLEAR THE HOLE BEFORE THE SOLENOID IS DE-ENERGIZED.

STEP 5

ADJUST THE FORWARD LIMIT SWITCH (10) AS FOLLOWS:

THE FORWARD AND AFT LIMIT SWITCHES AND FORWARD (21) (LOCATED ON FLOOR) AND AFT TRIPS ARE IDENTICAL. SEE STEP 4 FOR ADJUSTMENTS.

STEP 6

TO ADJUST THE CLEVIS END EXTENSION (6) OF THE TWO WAY SOLENOID (5) USE A 5/32 ALLEN WRENCH IN THE HOLE AT THE TOP AND TURN THE ARMATURE ON THE CLEVIS ROD CLOCKWISE TO SHORTEN THE EXTENSION OF THE CLEVIS AND COUNTER CLOCKWISE TO LENGTHEN IT.

STEP 7

ADJUST THE CLUTCH AS FOLLOWS:

NOTE

ENGAGEMENT OF THE UPPER CONE CLUTCH ALLOWS THE STAIR TO RAISE OR LOWER AND ENGAGEMENT OF THE LOWER CONE CLUTCH ALLOWS THE GEARBOX TO MOVE AFT OR FORWARD DEPENDING ON THE ROTATION OF THE MOTOR (2).

A. IF THE STAIRS WILL NOT RETRACT FROM THE EXTENDED POSITION, ADJUST THE CLEVIS END EXTENSION (6) BY INSERTING A 5/32 ALLEN WRENCH IN THE HOLE ON TOP OF THE SOLENOID AND ROTATING CLOCKWISE ONE TURN. ATTEMP TO RETRACT STAIR. REPEAT AS REQUIRED.

B. WITH THE STAIR IN THE RETRACTED POSITION, ATTEMPT TO ELECTRICALLY STOW THE STAIR. IF CLUTCH DOES NOT ENGAGE AND STOW STAIR, THE NEXT STEP SHOULD BE ACCOMPLISHED.

C. IF TOTAL MOVEMENT OF THE SOLENOID CLEVIS (6) IS .20, EXCESSIVE WEAR OF THE CONE CLUTCH FACES HAS OCCURRED. TWO METHODS ARE RECOMMENDED TO OVERCOME THIS CONDITION:

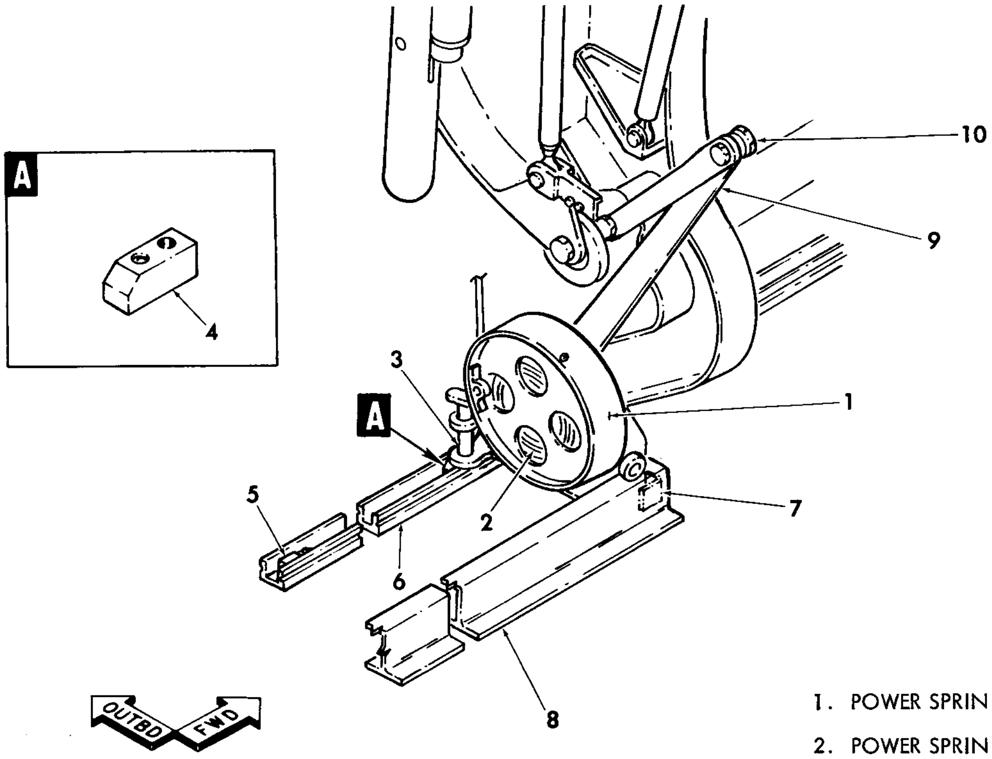
NOTE

REMOVE THE GEAR BOX ASSEMBLY FROM THE AIRCRAFT TO ACCOMPLISH THE FOLLOWING.

(1) USE A NO. 40 DRILL TO REMOVE THE CENTER PUNCH INDENTATIONS USED TO HOLD THE THREADED FITTING IN THE LOWER END OF THE SOLENOID HOUSING. SCREW THREADED FITTING OUT UNTIL FLUSH WITH THE SOLENOID HOUSING AND CENTER PUNCH TWO PLACES TO SECURE. THIS SHOULD GIVE ALL MOVEMENT OF THE SOLENOID CLEVIS NECESSARY TO ADJUST AS DETAILED IN "A" ABOVE, OR:

(2) REMOVE LAMINATIONS FROM THE WASHER BETWEEN THE FEMALE CLUTCH BEARING ASSEMBLY (7) AND THE CLUTCH BOX UNTIL THE TOTAL UP AND DOWN MOVEMENT OF THE ARM AT (6) IS 0.10 ± .01 INCHES. REASSEMBLE AND ADJUST AS DETAILED IN "A" ABOVE.

Figure 52-9. Adjustment of the Automatic Stair Gear Box Assembly (Airplanes 5066 and Up)



1. POWER SPRING COVER
2. POWER SPRING
3. ALIGNMENT PIN
4. ALIGNMENT PIN STOP
5. AFT STOP
6. TRACK
7. FWD STOP
8. RAIL
9. ARM
10. BOLT

STEP 1

ADJUST THE TENSION IN THE POWER SPRING (2) AS FOLLOWS:

A. WITH THE STAIR IN THE UP POSITION, REMOVE BOLT (10) AND PUSH TOP OF STAIR OUTBOARD SLIGHTLY TO CLEAR ARM (9).

B. IF THE POWER SPRING (2) DOES NOT HOLD ARM (9) AGAINST THE UP STOP WITH 1 TO 5 POUNDS FORCE, REMOVE ARM (9) FROM THE SPLINED SHAFT AND ROTATE THE SPLINED SHAFT CLOCKWISE, REPLACE ARM (9) AND CHECK FOR 1 TO 5 POUNDS FORCE OF ARM (9) AGAINST ITS STOP. READJUST AS NECESSARY.

STEP 2

CHECK ALIGNMENT PIN (3) FOR CORRECT FUNCTION AS FOLLOWS:

A. HOLD UP ALIGNMENT PIN (3) WITH A FINGER AND PUSH GEAR BOX ASSEMBLY AGAINST STOP (7).

B. ALIGNMENT PIN (3) SHOULD FALL FREELY INTO HOLE IN ALIGNMENT PIN STOP (4). ENLARGE HOLE IN STOP IF NECESSARY.

C. PUSH GEAR BOX ASSEMBLY AGAINST AFT STOP (5) AND REPEAT STEPS A AND B.

Figure 52-10. Adjustment of the Manual Gear Box Assembly (5121 and Up)

SECTION 52.5
LANDING GEAR DOORS

Contents Page

NOSE LANDING GEAR DOORS 52-49

MAIN LANDING GEAR DOORS 52-49

FIGURES

52-11. Nose and Main Landing Gear Doors 52-50
52-12. Nose Landing Gear Doors Skin and Structure 52-51
52-13. Main Landing Gear Doors Skin and Structure 52-53

52.5.1 NOSE LANDING GEAR DOORS. The nose gear doors are of aluminum alloy construction. The two main nose gear doors consist of inner skin, stiffeners, and outer skin. The strut door is constructed of honeycomb panel. The steering cylinder door is sheet metal.

52.5.2 MAIN LANDING GEAR DOORS. Each main gear has one inboard door and three outboard doors. All of the doors are of aluminum alloy construction. The inboard door, wheel cover door, and strut doors are constructed of honeycomb panels. The actuator cover door is made of an inner hat section spot-welded to an outer skin.

ITEM	COMPONENT	FIG. NO.
1	NOSE GEAR DOOR	52-12
2	NOSE GEAR STEERING CYLINDER DOOR	52-12
3	NOSE GEAR STRUT DOOR	52-12
4	MAIN GEAR INBOARD DOOR	52-13
5	MAIN GEAR ACTUATOR COVER DOOR	52-13
6	MAIN GEAR STRUT DOOR	52-13
7	MAIN GEAR WHEEL COVER DOOR	52-13

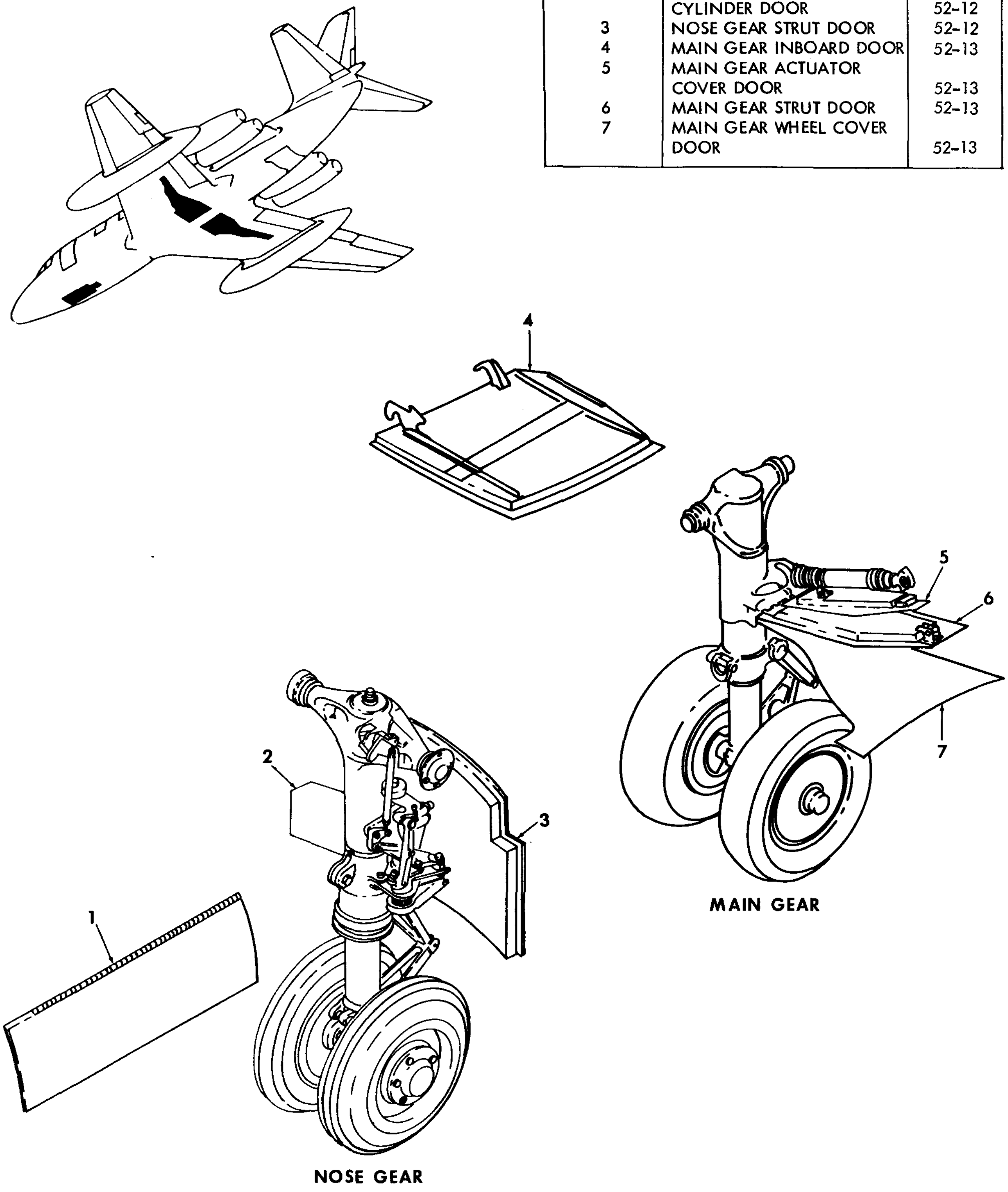
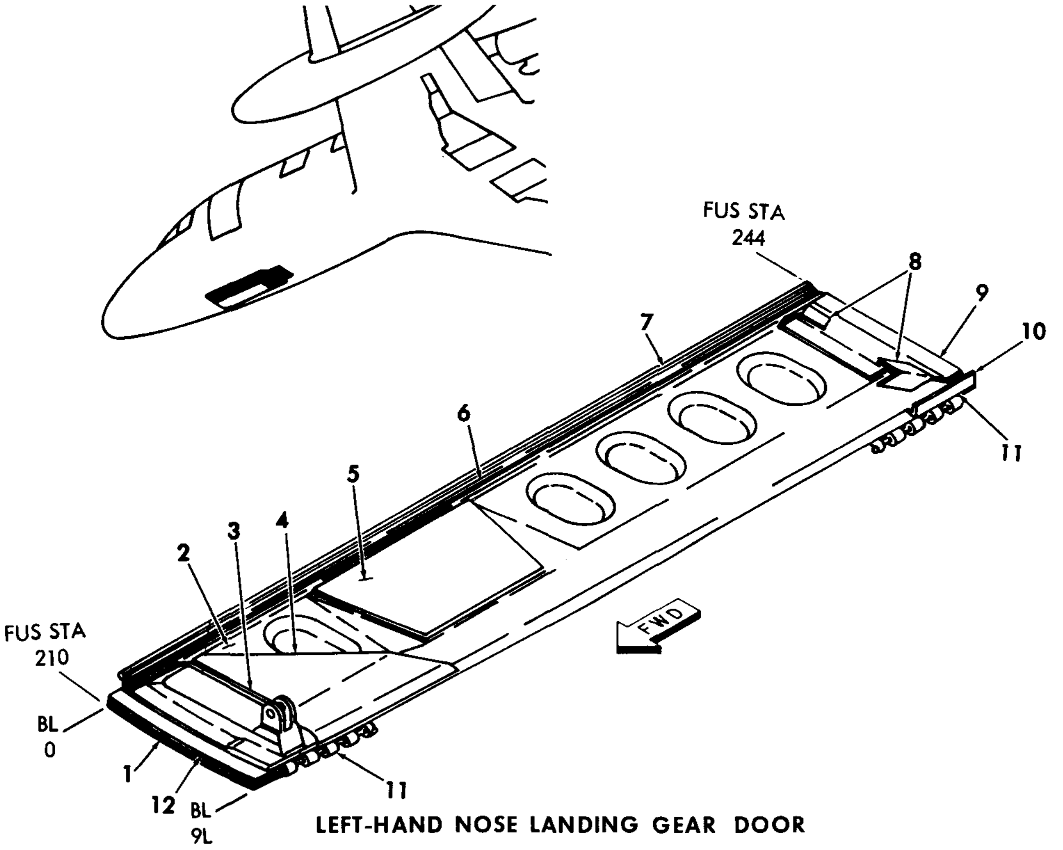
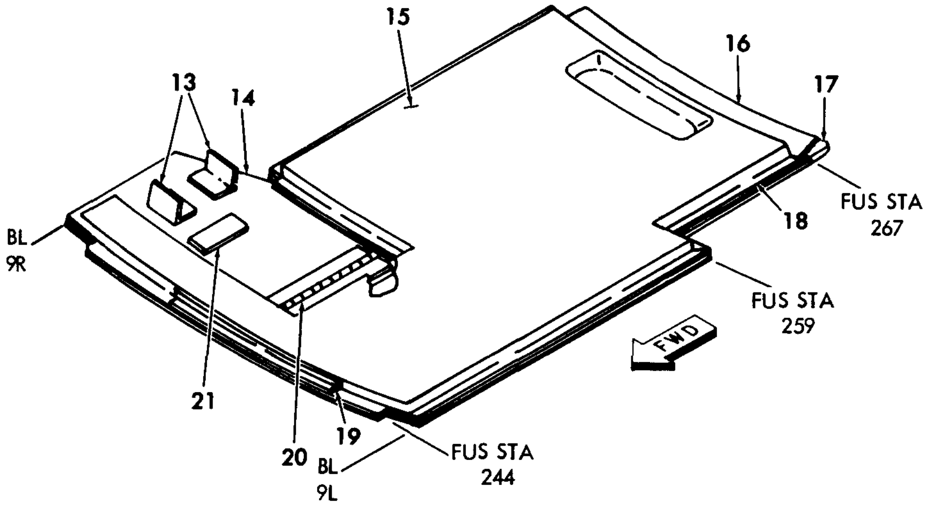


Figure 52-11. Nose and Main Landing Gear Doors



LEFT-HAND NOSE LANDING GEAR DOOR
(RIGHT-HAND NOSE LANDING GEAR DOOR OPPOSITE)



NOSE LANDING GEAR STRUT AND STEERING CYLINDER DOORS

Figure 52-12. Nose Landing Gear Doors Skin and Structure (Sheet 1 of 2)

ITEM NO.	DESCRIPTION	THICKNESS	MATERIAL
1	OUTER SKIN	0.050	2024-T4 CLAD
2	INNER SKIN	0.032	2024-T4 CLAD
3	FITTING		A-356 CASTING
4	DOUBLER	0.071	2024-T4 CLAD
5	RUB PAD	0.125	COTTON FILLED PHENOLIC
6	SEAL		SYNTHETIC RUBBER
7	SEAL RETAINER	0.032	2024-T4 CLAD
8	CLIP	0.050	2024-T4 CLAD
9	ZEE	0.050	2024-T4 CLAD
10	ANGLE		2024-T4 EXTRUSION
11	HINGE		4130 STEEL
12	DOUBLER	0.040	2024-T3 CLAD
13	CLIP		2024-T4 EXTRUSION
14	DOOR	0.090	7075-T6 CLAD
15	PANEL ASSEMBLY		
	PLATE	0.016	7075-T6
	HONEYCOMB CORE		5052-H38
16	STRAP	0.050	2024-T3
17	FILLER	0.071	2024-T3
18	ZEE	0.020	2024-T3
19	STIFFENER	0.032	2024-T4
20	MS20001PH12 HINGE		2024-T4 EXTRUSION
21	RUBBING STRIP	0.096	TEFLON

Figure 52-12. Nose Landing Gear Doors Skin and Structure (Sheet 2 of 2)

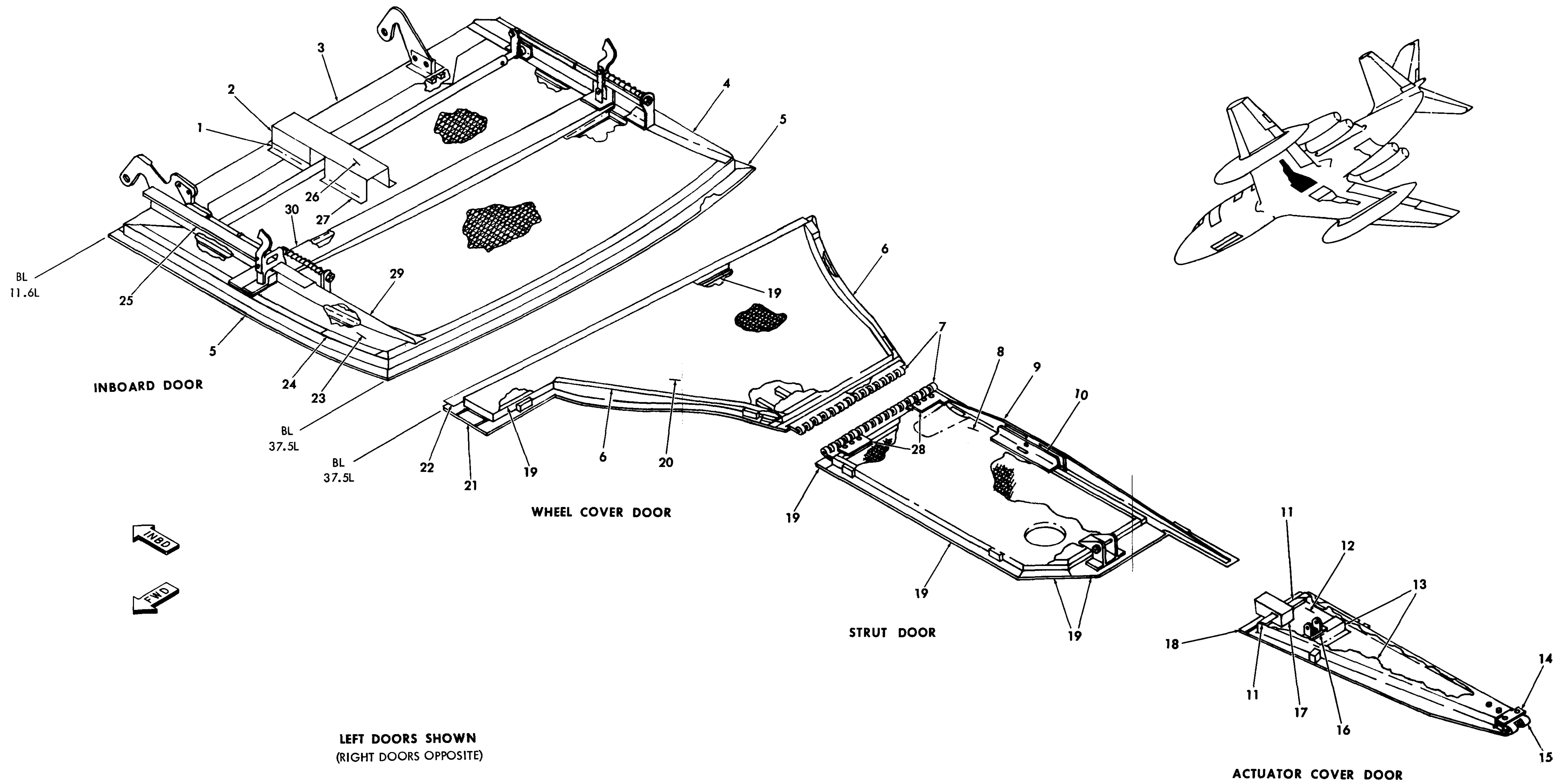


Figure 52-13. Main Landing Gear Doors Skin and Structure (Sheet 1 of 2)

ITEM NO.	DESCRIPTION	THICKNESS	MATERIAL
1	TEE		7075-T6 EXTRUSION
2	CHANNEL	0.063	2024-T4 CLAD
3	EDGE MEMBER	0.032	7075-T6 CLAD
4	AFT CHANNEL	0.050	7075-T6 CLAD
5	EDGE MEMBER	0.025	7075-T6 CLAD
6	SUPPORT	0.032	7075-T6 CLAD
7	HINGE		7075-T6 EXTRUSION
8	HONEYCOMB PANEL ASSEMBLY		
	PLATE	0.016	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	DISH	0.020	7075-T6 CLAD
	DOUBLER	0.100	2024-T3 CLAD
9	ANGLE		7075-T6 EXTRUSION
10	FITTING		2024-T4 BAR
11	ZEE	0.040	2024-T4 CLAD
12	PLATE	0.040	2024-T4 CLAD
13	HAT	0.040	2024-T4 CLAD
14	BLOCK		AZ-31-B EXTRUDED BAR
15	HINGE		2024-T4 EXTRUSION
16	FITTING		2024-T4 EXTRUSION
17	PAD		PHENOLIC SHEET
18	STRIKER PLATE	0.040	2024-T4 CLAD
19	SUPPORT		7075-T6 EXTRUSION
20	HONEYCOMB PANEL ASSEMBLY		
	PLATE	0.020	7075-T6 CLAD
	PLATE	0.050	2024-T4 CLAD
	HONEYCOMB CORE		5052-H38
	TUBE	0.050	7075-T6 EXTRUSION
	DOUBLER	0.020	7075-T6 CLAD
21	PAD	0.050	2024-T4 CLAD
22	STRIKER PLATE	0.060	2024-T4 CLAD
23	HONEYCOMB PANEL ASSEMBLY		
	OUTER SKIN	0.025	7075-T6 CLAD
	INNER SKIN	0.020	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	TUBE	0.050	7075-T6 EXTRUSION
	DOUBLER	0.061	7075-T6 CLAD
	DOUBLER	0.032	7075-T6 CLAD
	DOUBLER	0.025	7075-T6 CLAD
24	STRIKER PLATE	0.125	2024-T3 CLAD
25	STIFFENER	0.063	7075-T6 CLAD
26	RUB PLATE		COTTON FABRIC PHENOLIC
27	BRACKET	0.063	2024-T4 CLAD
28	FITTING		2024-T4 PLATE
29	FORWARD RIB	0.063	7075-T6 CLAD
30	BEAM	0.050	7075-T6

Figure 52-13. Main Landing Gear Doors Skin and Structure (Sheet 2 of 2)

CHAPTER 53
FUSELAGE

Contents Page

GENERAL 53-1

NOSE SECTION 53-1

FORWARD AND CENTER SECTIONS 53-1

AFT SECTION 53-2

DETAIL FUSELAGE COMPONENT MATERIAL AND IDENTIFICATION 53-2

FIGURES
53-1. Nose and Equipment Support Structure 53-4
53-2. Fuselage Floor Panels and Covers 53-6
53-3. Fuselage Skins 53-8
53-4. Forward Fuselage 53-10
53-5. Forward Fuselage Floor Support Structure 53-12
53-6. Center Fuselage Floor Support Structure and Wheel Well Box 53-14
53-7. Center Fuselage Upper Structure 53-16
53-8. Center Fuselage Lower Structure 53-18
53-9. Aft Fuselage Structure 53-20

53.1 GENERAL. (See figure 53-1) The fuselage group consists of the nose section, forward section, center section and aft section. The nose section extends from the fuselage station 94 to 158, the forward section from fuselage station 158 to 270, the center section from fuselage station 270 to 570, and the aft section from fuselage station 570 to 800.

53.2 NOSE SECTION. The nose section consists of electronic equipment racks enclosed in a light exterior shell structure.

a. The equipment racks, consisting of an upper and lower shelf, are of aluminum alloy construction using extrusions and formed members. These shelves are cantilever structures supported by the bulkhead at fuselage station 164. A track extends along each side of the lower shelf and provides support for the exterior shell structure.

b. The exterior shell structure consists of skins, longerons, and circumferential rings of aluminum alloy with a fiberglas nose radome attached at the forward bulkhead. Flush rivets are used in construction of the exterior shell. Latches are used to hold the shell structure firmly against the forward fuselage, and a circumferential rubber seal is used to render the junction airtight. The entire nose section exterior shell is sealed airtight to permit pressurization of the nose section.

53.3 FORWARD AND CENTER SECTIONS. The forward and center sections of the fuselage are of aluminum alloy construction consisting of skin, longerons, and circumferential rings and frames, and using flush rivets. The longerons are combinations of sheet metal, extrusions and machined fittings. Eleven main longerons are used. The frame members are at approximately 6.6-inch spacing. Five main frames between fuselage stations 410 and 490 provide for attaching

53-3 (Cont).

the wings to the fuselage. The forward and center fuselage sections are sealed, along with the nose section, to provide for pressurization. The bulkhead at fuselage station 570 is a pressure bulkhead and forms the aft boundary of the pressurized portion of the fuselage. The bulkhead at station 164 does not normally serve as a pressure bulkhead; however, it is capable of withstanding decompression pressure as a failsafe provision if the nose section fails.

53.4 AFT SECTION. The aft section of the fuselage is similar in construction to the center and forward sections with the exception that the aft section is not sealed for pressurization. Two heavy frames at fuselage stations 589 and 608 provide for attaching the nacelles to the fuselage, and two bulkheads at fuselage stations 682 and 739 provide for attaching the empennage to the fuselage.

53.5 DETAIL FUSELAGE COMPONENT MATERIAL AND IDENTIFICATION. For material and identification of detail fuselage components refer to figures 53-1 through 53-9 at the end of this section. For typical repairs refer to Chapter 51.

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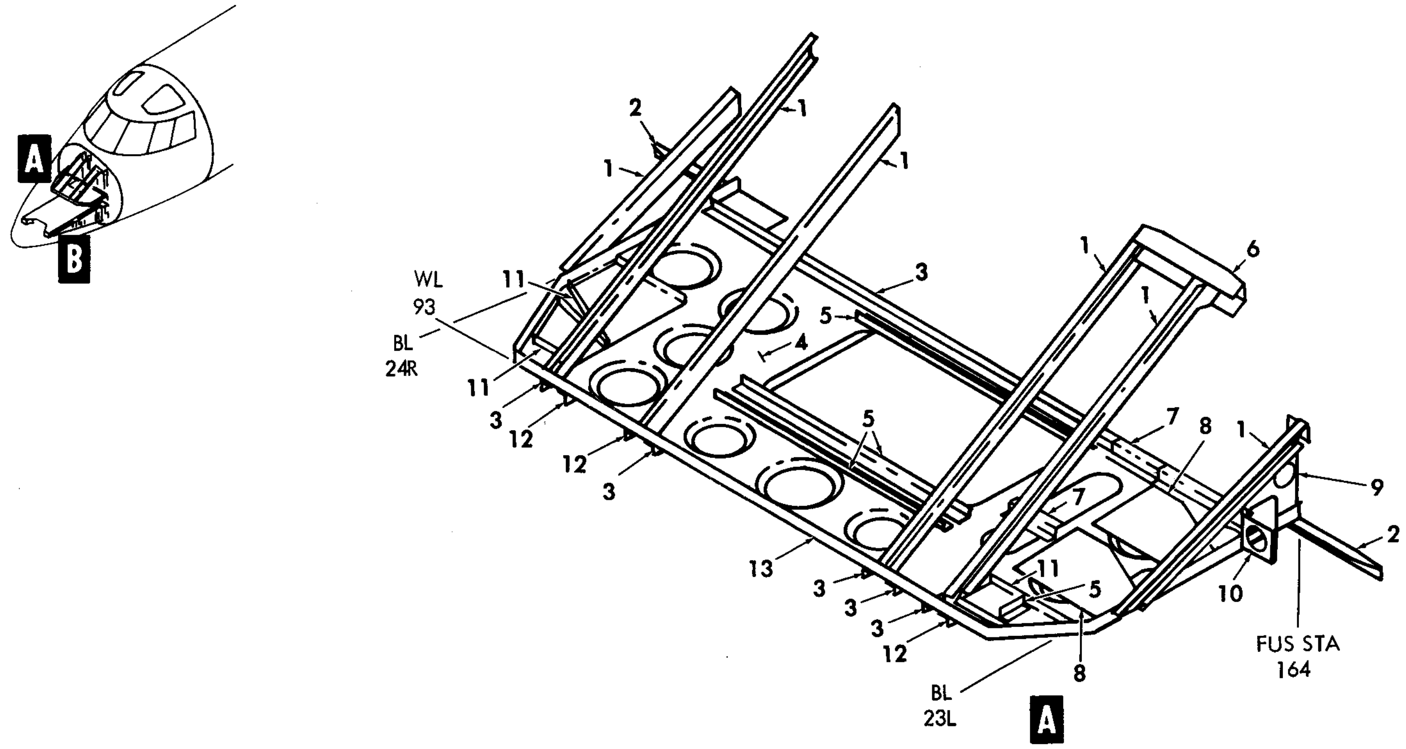
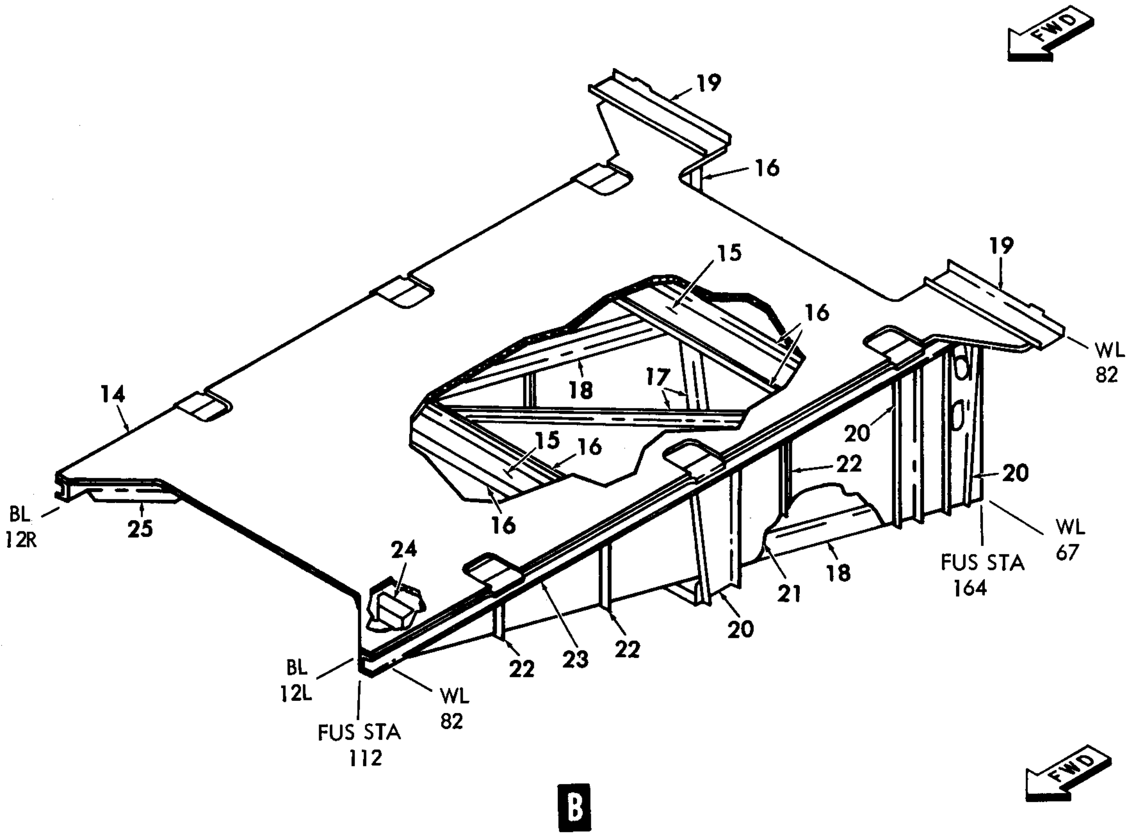
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	STRUT		2024-T4 EXTRUSION
2	ANGLE	0.040	2024-T3 CLAD
3	ANGLE		2024-T42 EXTRUSION
4	WEB	0.020	2024-T3 CLAD
5	ANGLE		2024-T4 EXTRUSION
6	CHANNEL	0.050	2024-T4 EXTRUSION
7	ZEE	0.050	2024-T3 CLAD
8	WEDGE PLATE	0.375	2024-T351
9	PLATE	0.040	2024-T4 CLAD
10	BRACKET	0.050	2024-T4 CLAD
11	CHANNEL	0.040	2024-T3 CLAD
12	ANGLE		7075-T6 EXTRUSION
13	SUPPORT		7075-T6 EXTRUSION
14	PLATE ASSEMBLY		
	FACE PLATE	0.012	7075-T6 CLAD
	HONEYCOMB CORE	0.166	3003 FOIL
15	PLATE	0.050	7075-T6 CLAD
16	ANGLE		7075-T6 EXTRUSION
17	CROSS		7075-T6 EXTRUSION
18	CAP		7075-T6 EXTRUSION
19	CHANNEL	0.063	7075-T6 CLAD
20	CHANNEL	0.050	7075-T6 CLAD
21	WEB	0.040	7075-T6 CLAD
22	ANGLE	0.050	7075-T6 CLAD
23	TRACK		7075-T6 EXTRUSION
24	BRACKET	0.050	7075-T6 CLAD
25	CHANNEL		7075-T6 EXTRUSION
26	RING ASSEMBLY		
	RING SEGMENT	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.050	7075-T6 CLAD
	BRACKET	0.050	7075-T6 CLAD
27	DOUBLER	0.040	7075-T6 CLAD
28	LONGERON		7075-T6 EXTRUSION
29	LONGERON		7075-T6 EXTRUSION
30	RING ASSEMBLY		
	RING SEGMENT	0.040	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	STRAP	0.016	6AL-4V TITANIUM
31	CHANNEL	0.040	7075-T6 CLAD
32	RING ASSEMBLY		
	RING SEGMENT	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	STRAP	0.016	6AL-4V TITANIUM
33	RING ASSEMBLY		
	RING SEGMENT	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.016	6AL-4V TITANIUM
34	DOUBLER	0.050	7075-T6 CLAD
35	RING ASSEMBLY		
	RING SEGMENT	0.050	7075-T6 CLAD
	SPLICE ANGLE	0.050	7075-T6 CLAD
	STRAP	0.063	7075-T6 CLAD
36	INTERCOSTAL	0.050	7075-T6 CLAD
37	FITTING		356-T6 CASTING
38	ANGLE		7075-T6 EXTRUSION
39	WEB	0.032	7075-T6 CLAD
40	ANGLE	0.050	7075-T6 CLAD
41	ANGLE		2024-T4 EXTRUSION

JS-2-2-5301-1

Figure 53-1. Nose and Equipment Support Structure (Sheet 1 of 2)

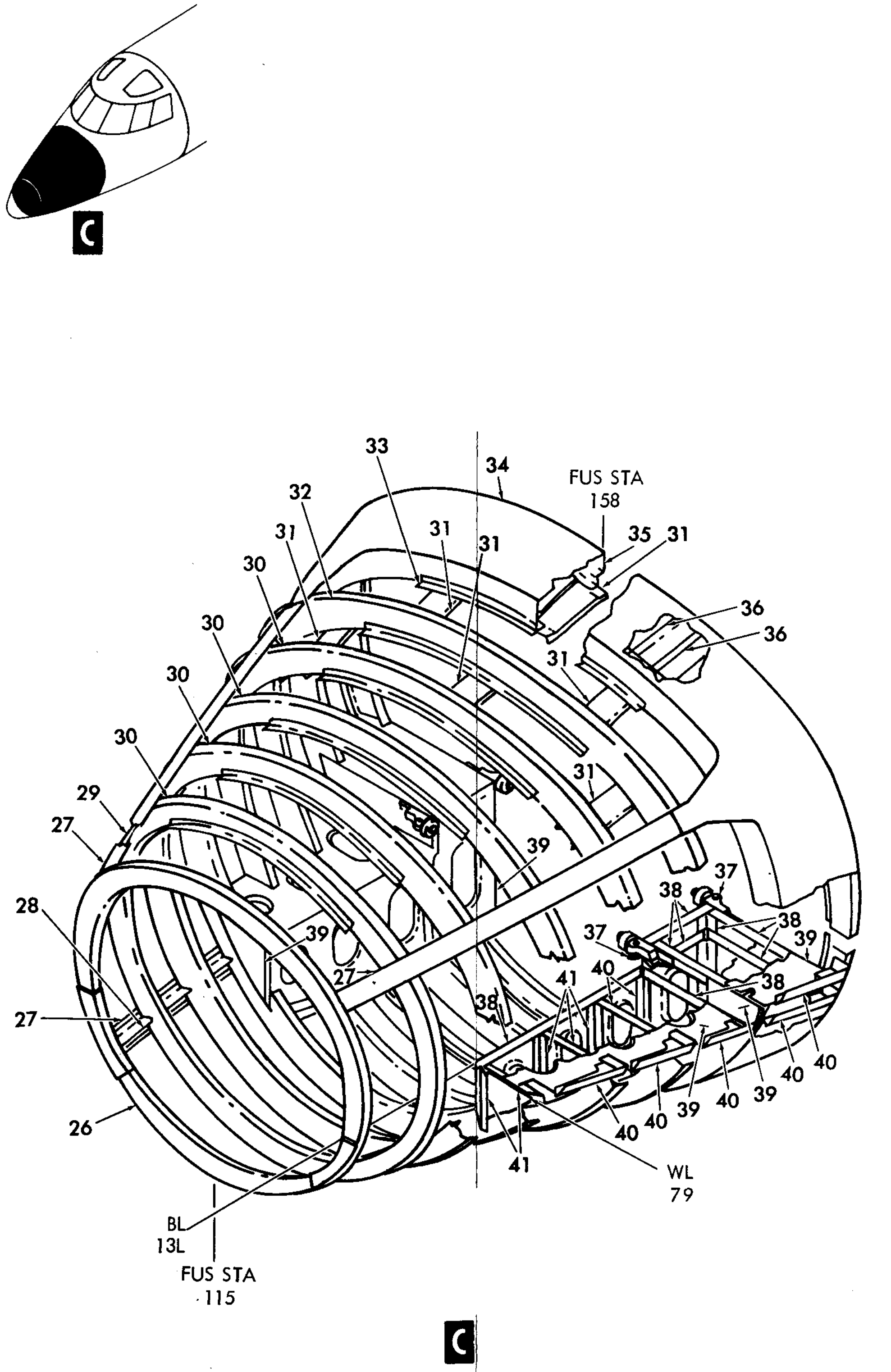


NOSE ELECTRONICS UPPER SUPPORT SHELF ASSEMBLY



ELECTRONIC GEAR SUPPORT STRUCTURE

C140-3-0-01-1



PRESSURE NOSE SHELL STRUCTURE

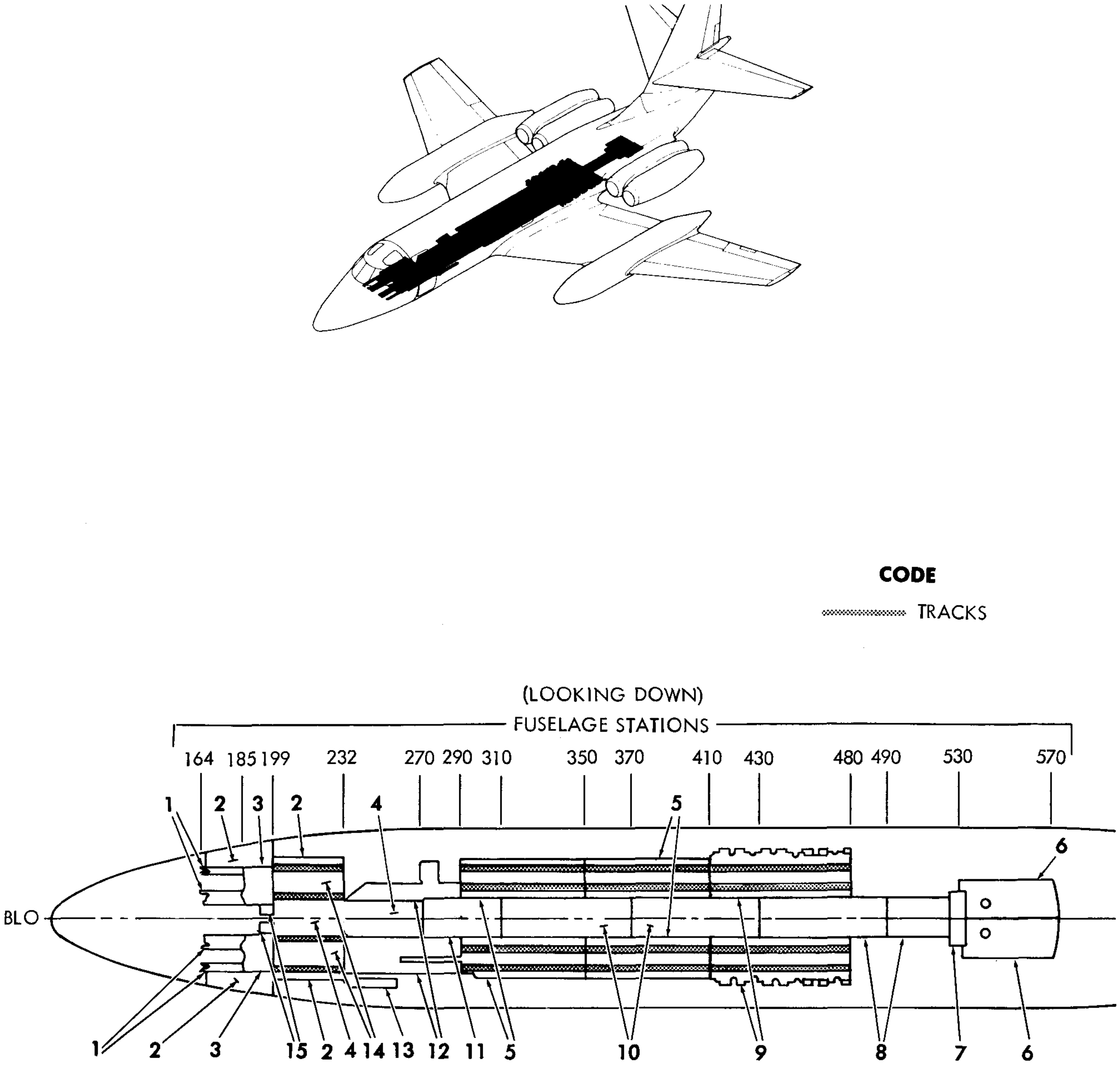
JS-2-2-5301-2

Figure 53-1. Nose and Equipment Support Structure (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	FLOOR	0.032	2024-T3 CLAD
2	COVER	0.040	2024-T4 CLAD
3	COVER	0.032	2024-T3 CLAD
4	FLOOR	0.040	2024-T3 CLAD
5	SEAT DECK PANEL ASSEMBLY		
	PANEL	0.020	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	PLATE	0.040	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	CHANNEL	0.050	2024-T3 CLAD
	TRACK		7075-T6 EXTRUSION
6	WALKWAY PANEL ASSEMBLY		
	PANEL	0.012	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	COVER	0.063	7075-T6 CLAD
7	COVER	0.063	7075-T6 CLAD
8	WALKWAY PANEL ASSEMBLY		
	PANEL	0.012	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
9	SEAT DECK PANEL ASSEMBLY		
	PANEL	0.020	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	PLATE	0.040	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	CHANNEL	0.050	2024-T3 CLAD
	TRACK		7075-T6 EXTRUSION
	DOUBLER	0.040	7075-T6 CLAD
10	WALKWAY PANEL ASSEMBLY		
	PANEL	0.012	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
11	FLOOR PANEL ASSEMBLY		
	PANEL	0.040	7075-T6 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
	PAN	0.037	GLASS LAMINATE
12	FLOOR PANEL ASSEMBLY		
	PANEL	0.016	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
13	FLOOR PANEL ASSEMBLY		
	PANEL	0.012	7075-T6 CLAD
	PANEL	0.025	7075-T6 CLAD
	HONEYCOMB CORE		5052-H38
14	PILOTS FLOOR PANEL ASSEMBLY		
	PANEL	0.032	2024-T3 CLAD
	CHANNEL	0.032	2024-T4
	ZEE	0.032	2024-T4
	ZEE		7075-T6 EXTRUSION
	TRACK		7075-T6 EXTRUSION
15	COVER	0.032	7075-T6 CLAD

JS-2-2-5302-1

Figure 53-2. Fuselage Floor Panels and Covers (Sheet 1 of 2)



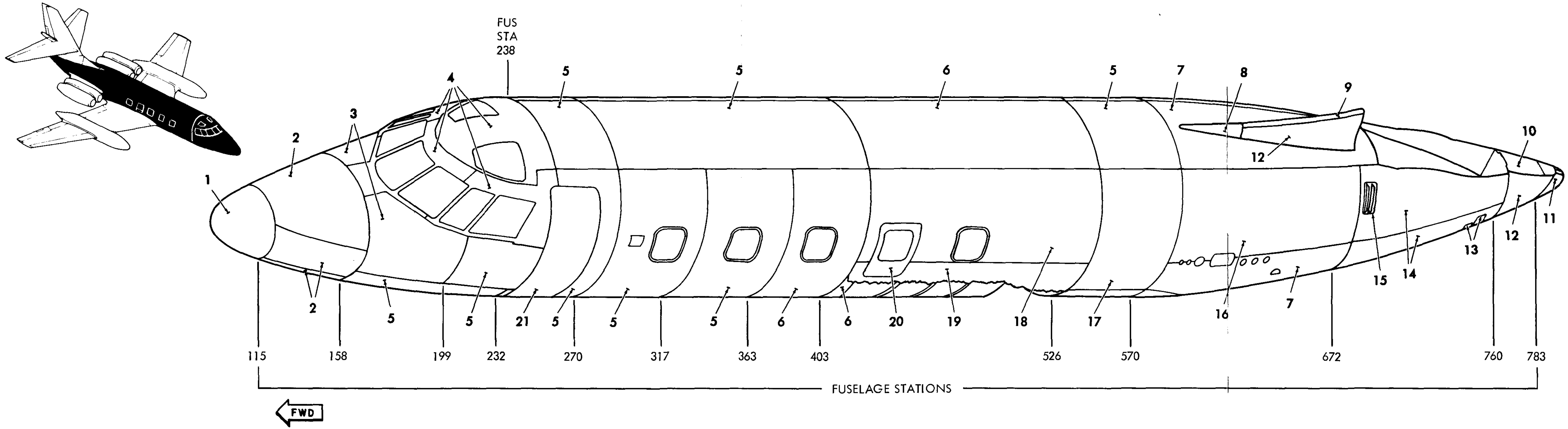
JS-2-2-5302-2

Figure 53-2. Fuselage Floor Panels and Covers (Sheet 2 of 2)

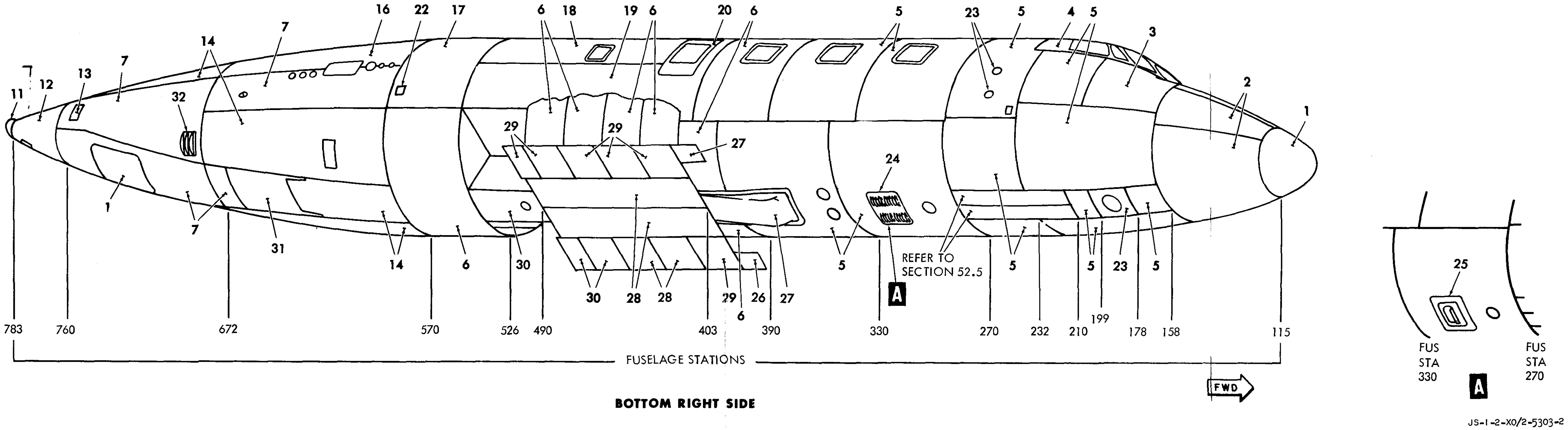
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	RADOME		FIBERGLAS
2	SKIN	0.032	2024-T4 CLAD
3	SKIN	0.032	2024-T3 CLAD
3	SKIN	0.032	2024-T4 CLAD
4	SKIN ASSEMBLY		
	OUTER SKIN	0.040	2024-T4 CLAD
	INNER SKIN	0.040	2024-T4 CLAD
5	SKIN	0.032	2024-T3 CLAD
6	SKIN	0.040	2024-T3 CLAD
7	SKIN	0.032	2024-T4 CLAD
8	FAIRING	0.032	2024-T4 CLAD
9	DORSAL CAP	0.032	2024-T4 CLAD
10	SKIN	0.025	7075-T6 CLAD
11	DOOR	0.032	2024-T4 CLAD
12	SKIN	0.025	2024-T4 CLAD
13	DOOR	0.050	2024-T4 CLAD
14	SKIN	0.040	2024-T4 CLAD
15	LOUVER DOOR	0.040	2024-T4 CLAD
16	SKIN	0.036	2024-T4 CLAD
17	SKIN	0.036	2024-T3 CLAD
18	SKIN	0.050	2024-T3 CLAD
19	FILLET SKIN	0.040	2024-T4 CLAD
20	DOOR SKIN	0.032	2024-T4 CLAD
21	MAIN PASSENGER DOOR		
	OUTER SKIN	0.032	2024-T4 CLAD
	INNER SKIN	0.032	2024-T4 CLAD
	INNER SKIN	0.032	2024-T3 CLAD
22	DOOR	0.040	2024-T4 CLAD
23	DOOR PANEL	0.071	2024-T4 CLAD
24	LOUVER	0.071	6061-T6
25.....5041 & UP	DUCT ASSEMBLY		
	DUCT		LAMINATED GLASS CLOTH
	FLANGE		LAMINATED GLASS CLOTH
26	ACCESS DOOR		
	OUTER SKIN	0.020	2024-T4 CLAD
	INNER SKIN	0.020	2024-T4 CLAD
27	COVER		LAMINATED GLASS CLOTH
28	ACCESS DOOR		
	OUTER SKIN	0.032	2024-T4 CLAD
	INNER SKIN	0.025	2024-T4 CLAD
29	ACCESS DOOR		
	OUTER SKIN	0.032	2024-T4 CLAD
	INNER SKIN	0.032	2024-T4 CLAD
30	FAIRING	0.032	2024-T3 CLAD
31	SPEED BRAKE DOOR		AZ91C-T6 CASTING
32....5001-5023	LOUVER DOOR	0.040	2024-T4 CLAD

JS-2-2-5303-1

Figure 53-3. Fuselage Skins (Sheet 1 of 2)



TOP LEFT SIDE



BOTTOM RIGHT SIDE

Figure 53-3. Fuselage Skins (Sheet 2 of 2)

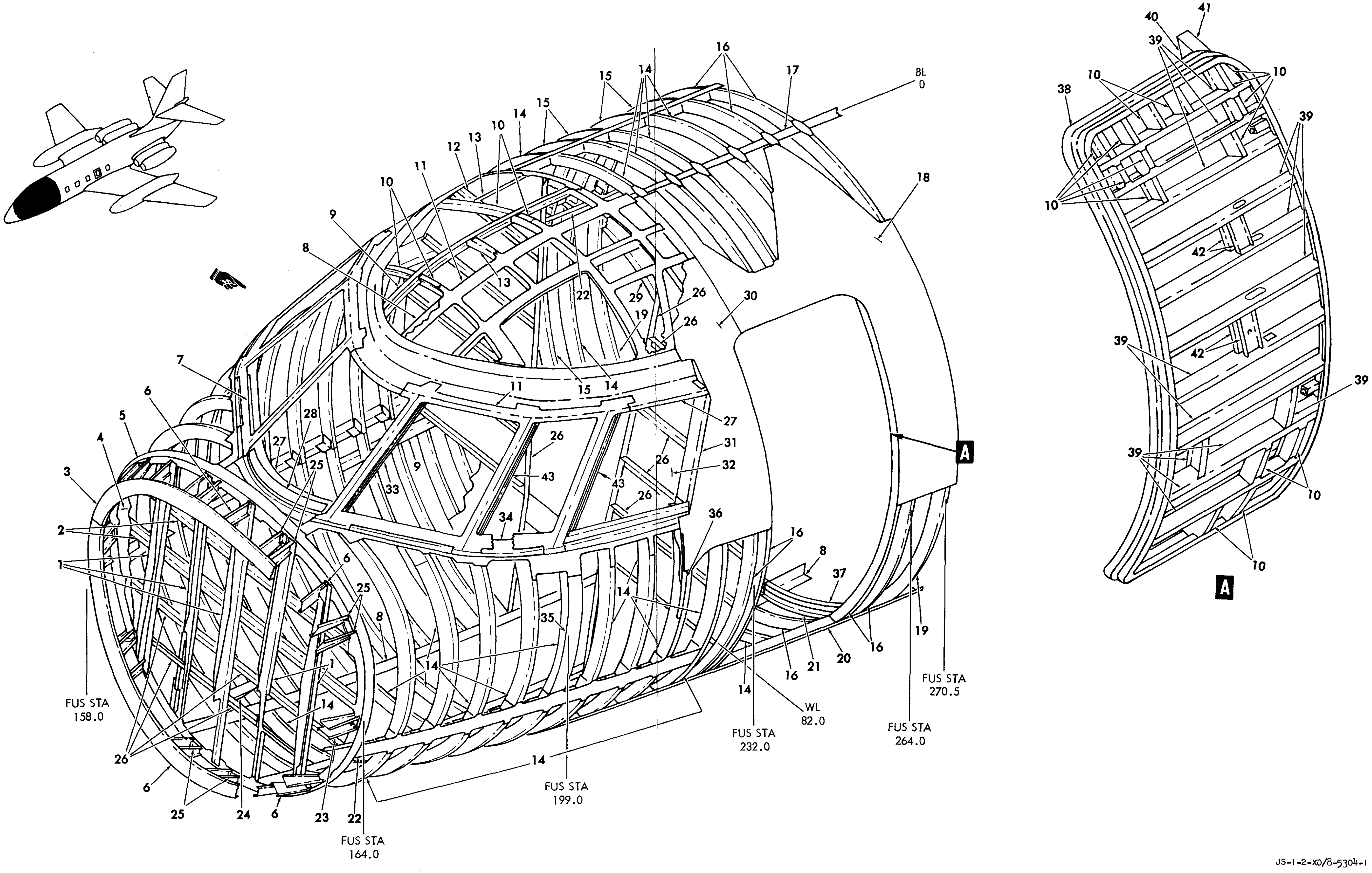


Figure 53-4. Forward Fuselage (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	BEAM		7075-T6 EXTRUSION
2	ZEE		7075-T6 EXTRUSION
3	RING	0.050	7075-T6 CLAD
4	WEB	0.032	2024-T3 CLAD
5	RING	0.090	7075-T6 CLAD
6	CHANNEL	0.040	7075-T6 CLAD
7	ANGLE	0.080	7075-T6 CLAD
8	LONGERON		7075-T6 EXTRUSION
9	LONGERON	0.063	7075-T6 CLAD
10	INTERCOSTAL	0.063	7075-T6 CLAD
11	DOUBLER	0.080	2024-T4
12	CHANNEL		7075-T6 EXTRUSION
13	STIFFENER	0.050	7075-T6 CLAD
14	RING	0.040	7075-T6 CLAD
15	RING	0.032	7075-T6 CLAD
16	RING	0.063	7075-T6 CLAD
17	STRINGER		7075-T6 EXTRUSION
18	DOUBLER	0.025	2024-T3 CLAD
19	RING	0.071	7075-T6 CLAD
20	SILL	0.125	7075-T6 CLAD
21	RING		7075-T6 EXTRUSION
22	LONGERON	0.050	7075-T6 CLAD
23	ANGLE	0.040	7075-T6 CLAD
24	TEE		7075-T6 EXTRUSION
25	CHANNEL	0.063	7075-T6 CLAD
26	ANGLE		7075-T6 EXTRUSION
27	SILL	0.090	2024-T4
28	CHANNEL	0.080	2024-T4 CLAD
29	FRAME	0.040	7075-T6 CLAD
30	DOUBLER	0.080	2024-T4 CLAD
31	CHANNEL	0.080	7075-T6 CLAD
32	WEB	0.025	7075-T6 CLAD
33	POST		2014-T6 FORGING.
34	INTERCOSTAL		7075-T6 EXTRUSION
35	SPLICE	0.040	7075-T6 CLAD
36	ANGLE (LEFT SIDE ONLY)	0.050	7075-T6 CLAD
37	RING CAP		7075-T6 EXTRUSION
38	FRAME	0.063	2024-T4 CLAD
39	FORMER	0.063	7075-T6 CLAD
40	WEB	0.063	7075-T6 CLAD
41	CHANNEL		2024-T4 EXTRUSION
42	BRACKET	0.063	2024-T4 CLAD
43	POST		4340 STEEL FORGING

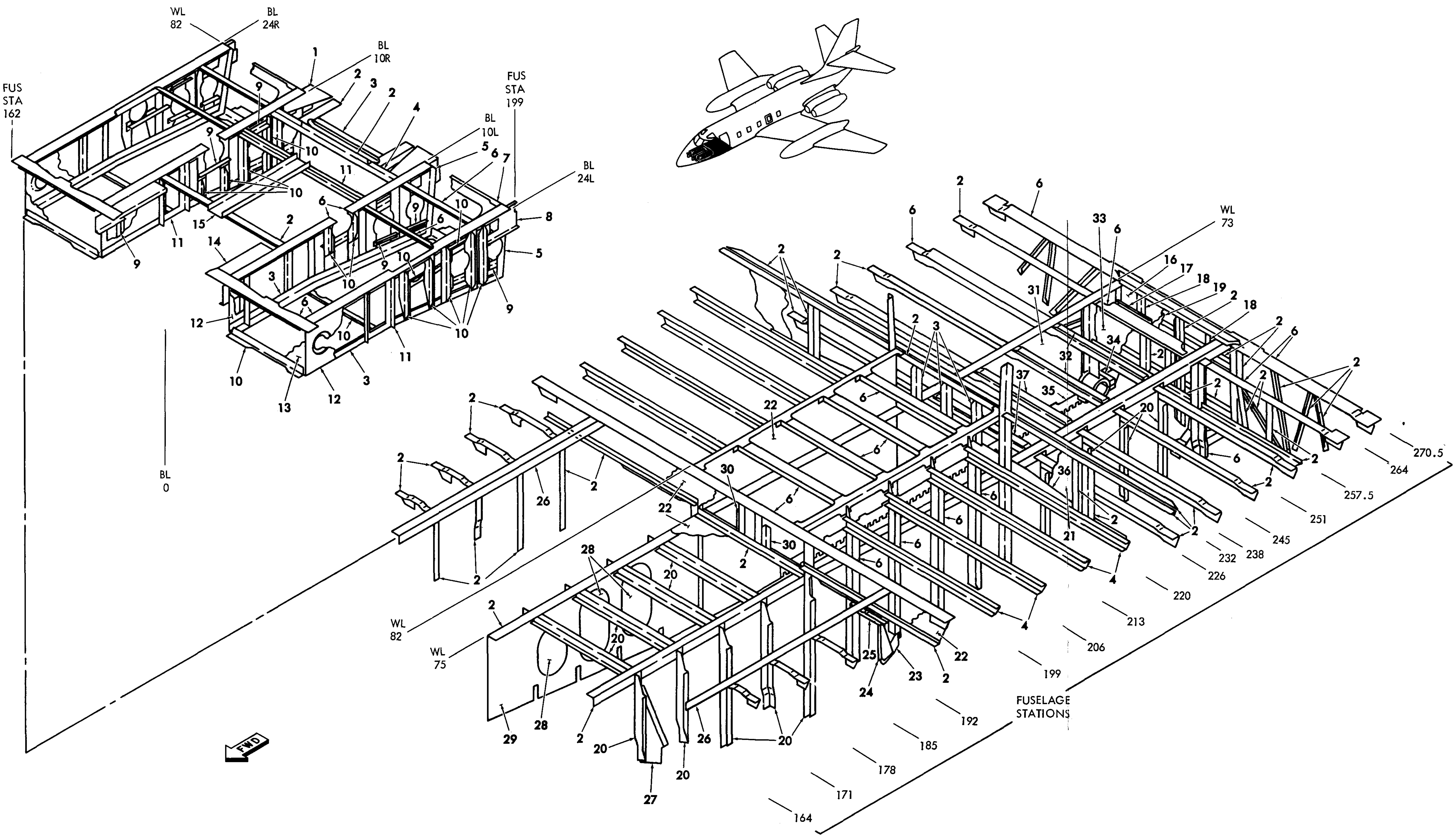
JS-1-2-X0/8-5304-2

Figure 53-4. Forward Fuselage (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	SUPPORT	0.080	2024-T4 CLAD
2	ANGLE		7075-T6 EXTRUSION
3	ANGLE		2024-T4 EXTRUSION
4	CHANNEL		2024-T4 EXTRUSION
5	ANGLE	0.050	2024-T3 CLAD
6	TEE		7075-T6 EXTRUSION
7	PANEL	0.032	2024-T3
8	CHANNEL	0.050	2024-T3 CLAD
9	ANGLE	0.040	2024-T3 CLAD
10	TEE		2024-T4 EXTRUSION
11	CHANNEL	0.040	2024-T3 CLAD
12	WEB	0.032	2024-T3 CLAD
13	WEB	0.040	2024-T4 CLAD
14	CHANNEL	0.063	2024-T3 CLAD
15	CHANNEL	0.040	2024-T4 CLAD
16	WEB	0.063	7075-T6 CLAD
17	DEFLECTOR	0.063	2024-T4
18	ANGLE		7075-T6 EXTRUSION
19	DOUBLER	0.071	7075-T6
20	ZEE		7075-T6 EXTRUSION
21	WEB	0.063	2024-T3 CLAD
22	WEB	0.040	2024-T3 CLAD
23	ANGLE (LEFT HAND ONLY)	0.050	7075-T6 CLAD
24	ANGLE (LEFT HAND ONLY)		7075-T6 EXTRUSION
25	ANGLE (LEFT HAND ONLY)		2024-T4 EXTRUSION
26	ANGLE	0.040	2024-T4 CLAD
27	GUSSET	0.040	7075-T6 CLAD
28	COVER	0.080	2024-T3 CLAD
29	WEB ASSEMBLY		
	WEB	0.040	2024-T3 CLAD
	DOUBLER	0.032	2024-T3 CLAD
30	STIFFENER	0.063	2024-T3 CLAD
31	WEB	0.050	2024-T3 CLAD
32 5042 & UP	CHANNEL		8740 ANNEALED STEEL
32 5001-5041	CHANNEL		7075-T6 EXTRUSION
33	WEB	0.072	7075-T6 CLAD
34	FITTING		7075-T6 FORGING
35	HINGE		7075-T6 EXTRUSION
36	SUPPORT		7075-T6 EXTRUSION
37	CHANNEL		7075-T6 EXTRUSION

JS-2-2-5305-1

Figure 53-5. Forward Fuselage Floor Support Structure (Sheet 1 of 2)



JS-2-2-5305-2

Figure 53-5. Forward Fuselage Floor Support Structure (Sheet 2 of 2)

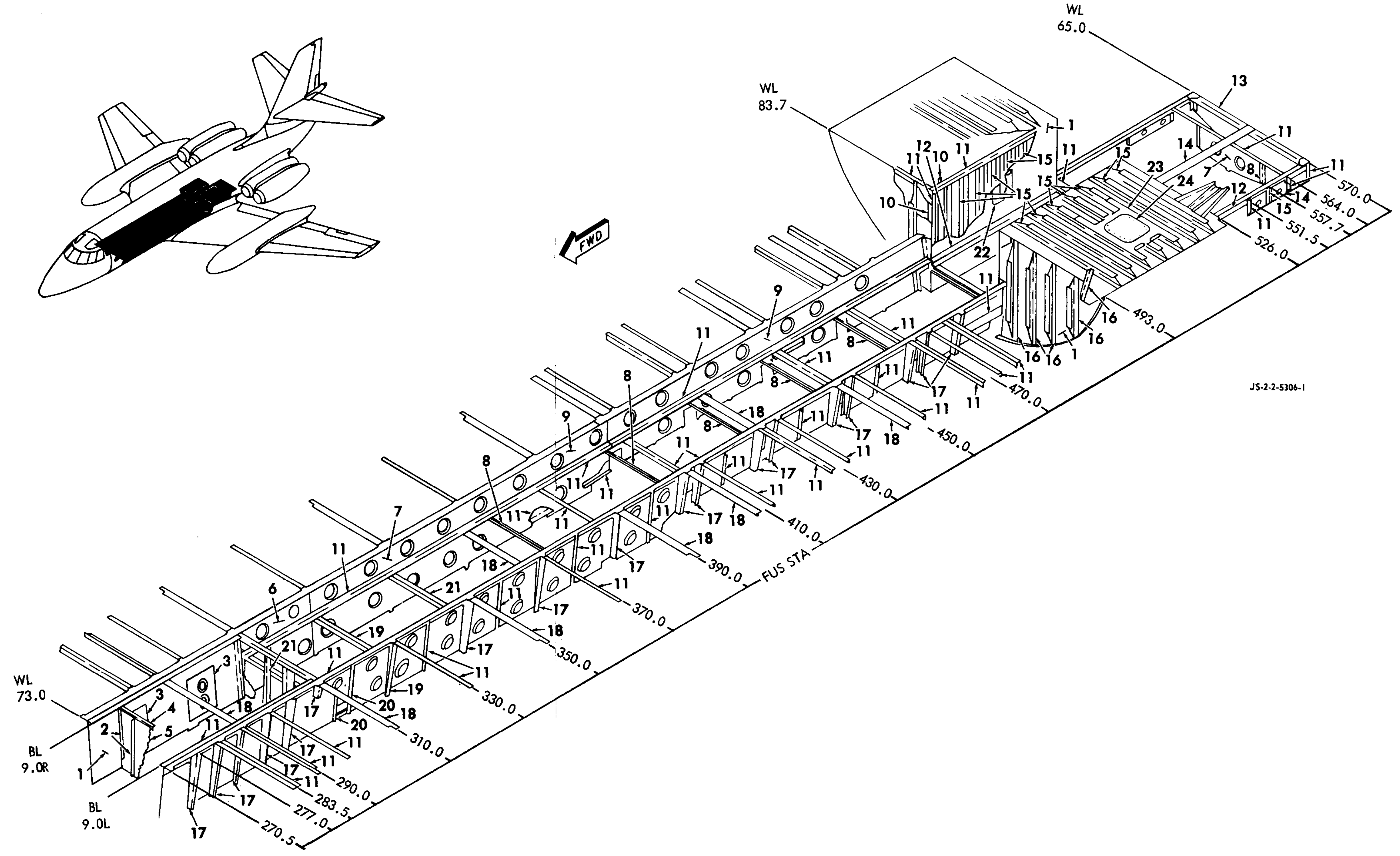


Figure 53-6. Center Fuselage Floor Support Structure and Wheel Well Box (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	WEB	0.040	2024-T3 CLAD
2....5041 & UP	ANGLE		2024-T4 EXTRUSION
3	DOUBLER (RIGHT SIDE ONLY)	0.040	2024-T3 CLAD
4....5041 & UP	CHANNEL	0.040	2024-T4
5....5041 & UP	BULKHEAD	0.040	2024-T4
6	WEB	0.025	2024-T3 CLAD
7	WEB	0.025	2024-T3 CLAD
8	ANGLE	0.040	2024-T3 CLAD
9	WEB	0.025	7075-T6 CLAD
10	SUPPORT		7075-T6 EXTRUSION
11	ANGLE		7075-T6 EXTRUSION
12	ANGLE		2024-T4 EXTRUSION
13	CHANNEL		7075-T6 EXTRUSION
14	INTERCOSTAL	0.032	2024-T3 CLAD
15	TEE		2024-T4 EXTRUSION
16	STIFFENER		7075-T6 EXTRUSION
17	CHANNEL	0.050	7075-T6 CLAD
18	TEE		7075-T6 EXTRUSION
19	CHANNEL	0.050	7075-T6 CLAD
20	ANGLE		7075-T6 EXTRUSION
21	BULKHEAD ASSEMBLY		
	WEB	0.040	2024-T3 CLAD
	ANGLE		7075-T6 EXTRUSION
	ANGLE		2024-T4 EXTRUSION
	TEE		7075-T6 EXTRUSION
	ZEE		7075-T6 EXTRUSION
22....5036 & UP	DOUBLER	0.063	7075-T6 CLAD
23	PLATE	0.100	2024-T3 CLAD
24	COVER PLATE	0.080	2024-T3 CLAD

JS-2-2-5306-2

Figure 53-6. Center Fuselage Floor Support Structure and Wheel Well Box (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	RING SEGMENT		7075-T6 EXTRUSION
2	RING SEGMENT	0.032	7075-T6 CLAD
3	RING SEGMENT	0.040	7075-T6 CLAD
4	FRAME ASSEMBLY		
	CAP	0.080	7075-T6 CLAD
	CHANNEL	0.040	7075-T6 CLAD
	DOUBLER	0.080	7075-T6
	SPLICE	0.040	2024-T4
	TEE		7075-T6 EXTRUSION
	CAP		7075-T6 EXTRUSION
5	FRAME ASSEMBLY		
	CHEMICALLY MILLED CAP	0.125	7075-T6
	CHANNEL	0.040	7075-T6 CLAD
	DOUBLER	0.091	7075-T6
	DOUBLER	0.080	7075-T6
	TEE		7075-T6 EXTRUSION
	CAP		7075-T6 EXTRUSION
6	RING SEGMENT		7075-T6 EXTRUSION
7	RING SEGMENT	0.050	7075-T6 CLAD
8	FRAME ASSEMBLY		
	SEGMENT	0.063	7075-T6 CLAD
	CAP		7075-T6 CLAD
	STRAP	0.160	7075-T6 CLAD
	PLATE	0.050	7075-T6 CLAD
9	CHANNEL	0.063	7075-T6 CLAD
10	CHANNEL	0.050	7075-T6 CLAD
11	PRESSURE BULKHEAD ASSEMBLY		
	SKIN SPLICE	0.063	2024-T4 CLAD
	SPLICE	0.090	2024-T4 CLAD
	FORWARD CAP ANGLE	0.090	2024-T4 CLAD
	AFT CAP ANGLE	0.090	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
	SPLICE ANGLE	0.050	7075-T6 CLAD
	CIRCUMFERENTIAL STIFFENER		7075-T6 EXTRUSION
	RADIAL STIFFENER		7075-T6 EXTRUSION
	DIAPHRAM	0.032	2024-T3 CLAD
	SPIDER	0.063	2024-T4 CLAD
12	CHANNEL	0.040	7075-T6 CLAD
13	LONGERON ASSEMBLY		
	LONGERON	0.063	7075-T6 CLAD
	SPLICE	0.040	2024-T3 CLAD
	SILL		7075-T6 EXTRUSION
14	FRAME ASSEMBLY		
	FRAME		7075-T6 PLATE
	ANGLE	0.080	7075-T6 CLAD
	CHANNEL	0.080	7075-T6 CLAD
15	CHEMICALLY MILLED FRAME		7075-T6 FORGING
16	WINDOW FRAME		7075-T6 FORGING
17	LONGERON ASSEMBLY		
	PLATE	0.063	2024-T3 CLAD
	CHANNEL	0.050	7075-T6 CLAD
18	LONGERON ASSEMBLY		
	PLATE	0.063	2024-T3 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	SKIN SPLICE	0.040	2024-T3 CLAD
	CHANNEL	0.063	7075-T6 CLAD
19	BEAM	0.050	7075-T6 CLAD
20	INTERCOSTAL	0.050	7075-T6 CLAD
21	CHANNEL		2024-T4 EXTRUSION
22	WING MATING PLANE.		
	CAP	0.063	2024-T4 CLAD
	ANGLE	0.063	2024-T4 CLAD
	STIFFENER		7075-T6 EXTRUSION
	STIFFENER		2024-T4 EXTRUSION
	NOSE RIB	0.050	2024-T4 CLAD
23	BEAM		7075-T6 EXTRUSION
24	CORNER	0.050	2024-T4 CLAD
25	SPLICE	0.050	2024-T4 CLAD
26	STRIKER		2024-T4 EXTRUSION
27	CORNER		2014-T6 FORGING
28	(NOT USED)		
29 ..... 5024 & UP	INTERCOSTAL	0.040	2024-T3 CLAD
29 5023-5024	INTERCOSTAL	0.040	7075-T6 CLAD
30	LONGERON ASSEMBLY		
	ANGLE		7075-T6 EXTRUSION
	CAP		7075-T6 EXTRUSION
	WEB	0.050	7075-T6 CLAD
31	LONGERON	0.050	7075-T6 CLAD
32	DOUBLER	0.025	2024-T3 CLAD
33	DOUBLER	0.040	2024-T3 CLAD
34	DOUBLER	0.016	6AL-4V TITANIUM
35	LONGERON ASSEMBLY		
	LONGERON		7075-T6 EXTRUSION
	SILL	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
36	LONGERON ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	STRIP	0.125	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION

JS-2-2-5307-1

Figure 53-7. Center Fuselage Upper Structure (Sheet 1 of 2)

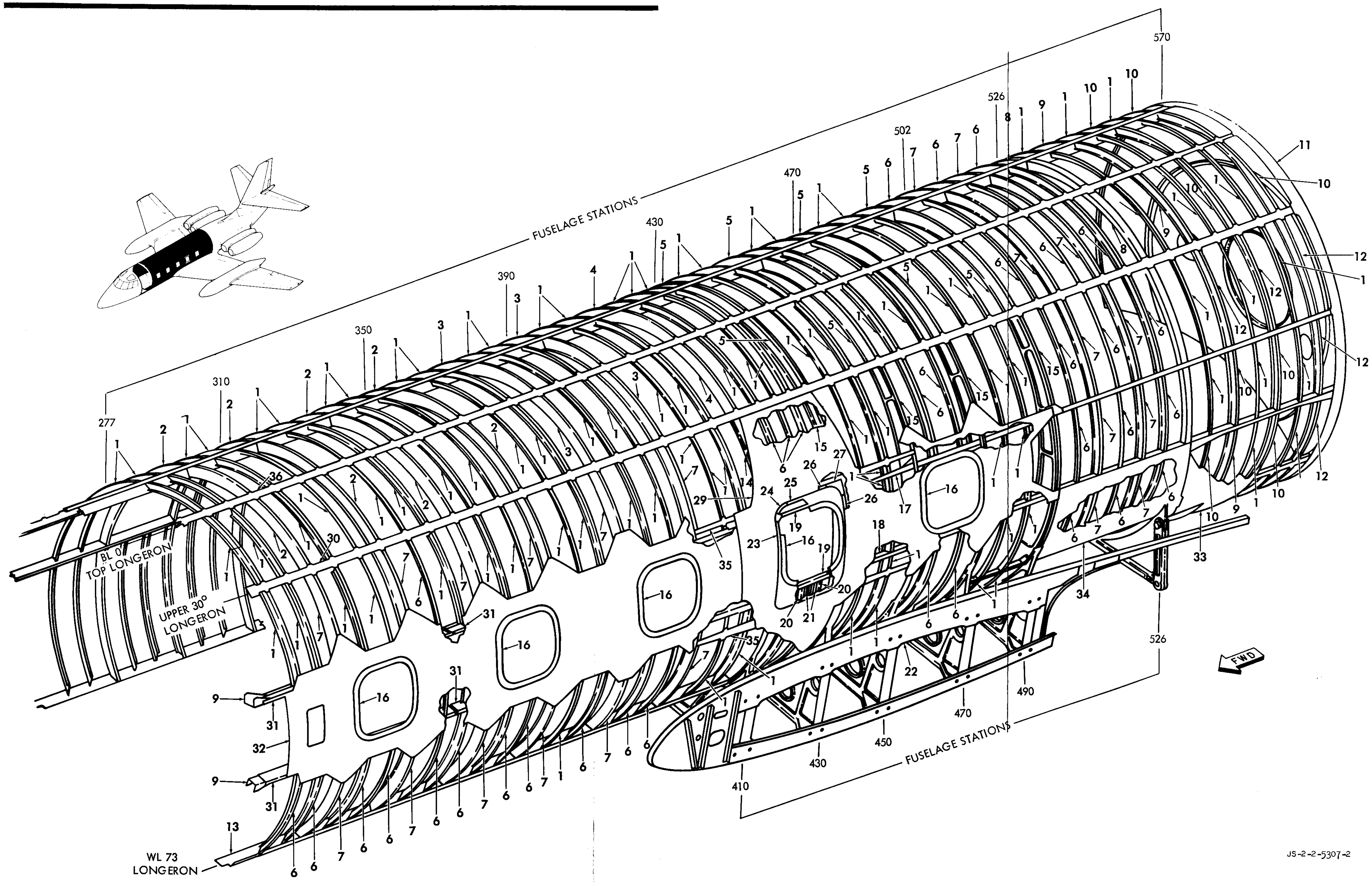


Figure 53-7. Center Fuselage Upper Structure (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1.	RING SEGMENT		7075-T6 EXTRUSION
2.	RING SEGMENT	0.040	7075-T6 CLAD
3.	RING SEGMENT		7075-T6 EXTRUSION
4.	CHANNEL	0.050	7075-T6 CLAD
5.	CHANNEL	0.040	7075-T6 CLAD
6.	MAIN FRAME		7075 FORGING
7...5041 & UP	INTERCOSTAL	0.050	2024-T4 CLAD
8.	ANGLE		7075-T6 EXTRUSION
9.	TEE		7075-T6 EXTRUSION
10.	BULKHEAD	0.032	7075-T6 CLAD
11.	ANGLE		2024-T4 EXTRUSION
12...5001-5045	INTERCOSTAL	0.040	2024-T3 CLAD
12...5046 & UP	INTERCOSTAL	0.040	7075-T6 CLAD
13.	RING SEGMENT	0.063	7075-T6 CLAD
14.	BULKHEAD ASSEMBLY		
	BULKHEAD	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
15.	LONGERON	0.071	7075-T6 CLAD
16.	LONGERON ASSEMBLY		
	LONGERON	0.050	7075-T6 CLAD
	DOUBLER	0.125	7075-T6 CLAD
	SPLICE PLATE	0.040	2024-T3 CLAD
17.	BULKHEAD ASSEMBLY		
	BULKHEAD	0.032	7075-T6 CLAD
	TEE		7075-T6 EXTRUSION
18.	INTERCOSTAL	0.040	7075-T6 CLAD
19.	INTERCOSTAL	0.032	7075-T6 CLAD
20.	LONGERON		7075-T6 EXTRUSION
21...5001-5045	INTERCOSTAL	0.040	2024-T3 CLAD
21...5046 & UP	INTERCOSTAL	0.032	7075-T6 CLAD
22...5001-5040	RING SEGMENT	0.063	7075-T6 CLAD
23.	RING SEGMENT		7075-T6 EXTRUSION
24.	LONGERON ASSEMBLY		
	BASE ANGLE		7075-T6 EXTRUSION
	CAP ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.063	2024-T3 CLAD
	BEARING PLATE		7075-T6 BARE
	SHEAR TIE		7075-T6 EXTRUSION
25.	CAN ASSEMBLY		
	CAN	0.040	2024-T3 CLAD
	STIFFENER	0.040	2024-T3 CLAD
	CAP	0.063	6061-T6
	ANGLE	0.040	2024-T3 CLAD
	FLANGE	0.032	347A STAINLESS STEEL
	BASE	0.032	347A STAINLESS STEEL
26.	LONGERON ASSEMBLY		
	LONGERON		7075-T6 EXTRUSION
	CAP		7075-T6 EXTRUSION
27.	WEB	0.040	2024-T4 CLAD
28.	ANGLE	0.040	7075-T6 CLAD
29.	CLIP	0.040	7075-T6 CLAD
30.	WEB	0.032	2024-T3 CLAD
31.	WEB	0.032	7075-T6 CLAD
32.	WEB	0.040	7075-T6 CLAD
33.	ANGLE	0.063	7075-T6
34.	LONGERON ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	LONGERON		7075-T6 EXTRUSION
	DOUBLER	0.063	7075-T6 CLAD
35...5041 & UP.	FRAME ASSEMBLY		
	FRAME	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	CHANNEL		7075-T6 EXTRUSION
36...5041 & UP	FRAME ASSEMBLY		
	ANGLE	0.040	2024-T4 CLAD
	FITTING	0.190	2024-T4
	BRACKET	0.050	2024-T4 CLAD
37.	BULKHEAD ASSEMBLY		
	BULKHEAD	0.032	7075-T6 CLAD
	TEE		7075-T6 EXTRUSION
	BRACKET	0.032	7075-T6 CLAD
	SPLICE	0.063	7075-T6 CLAD

Figure 53-8. Center Fuselage Lower Structure (Sheet 1 of 2)

Changed 1 April 1968

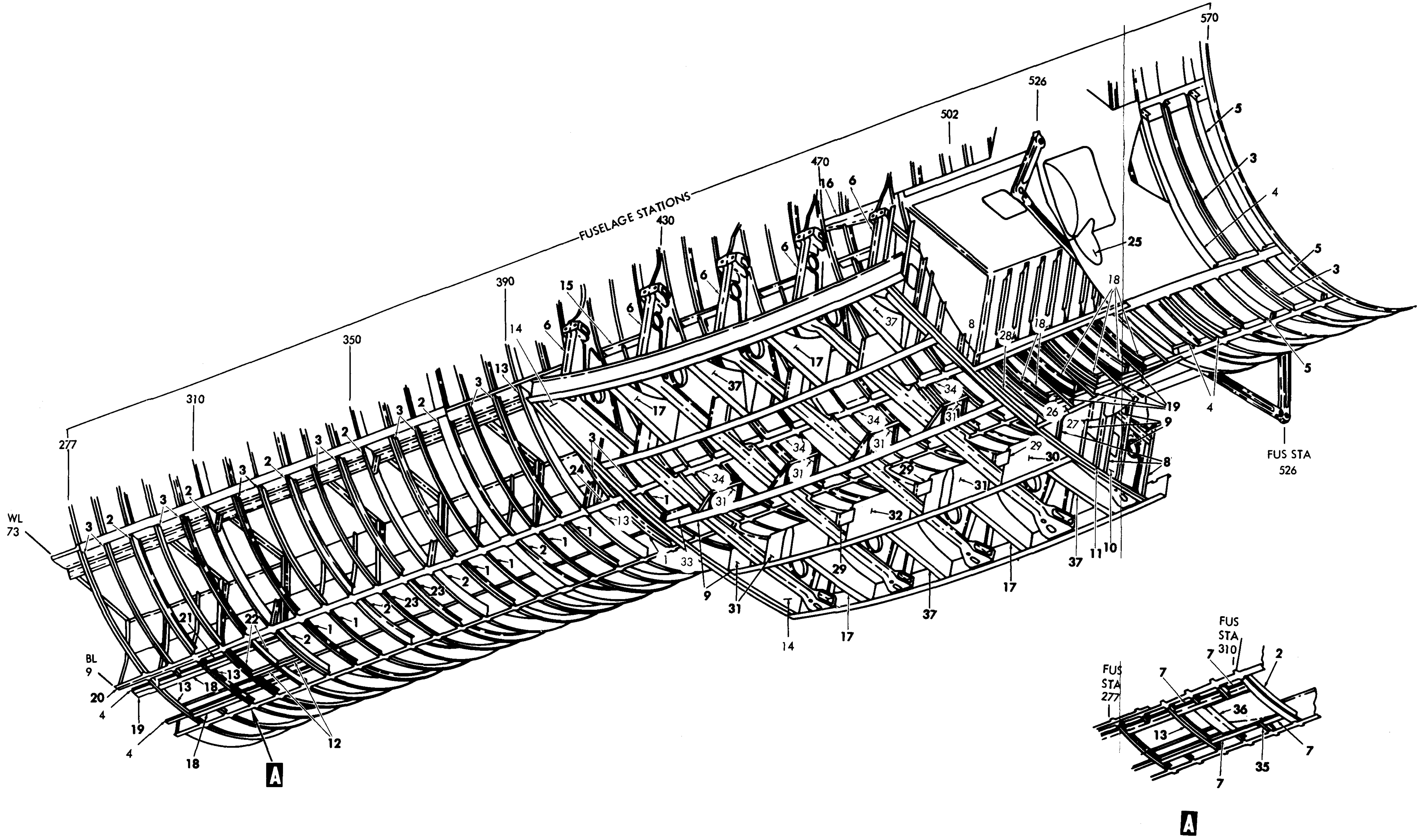


Figure 53-8. Center Fuselage Lower Structure (Sheet 2 of 2)

Changed 31 August 1983

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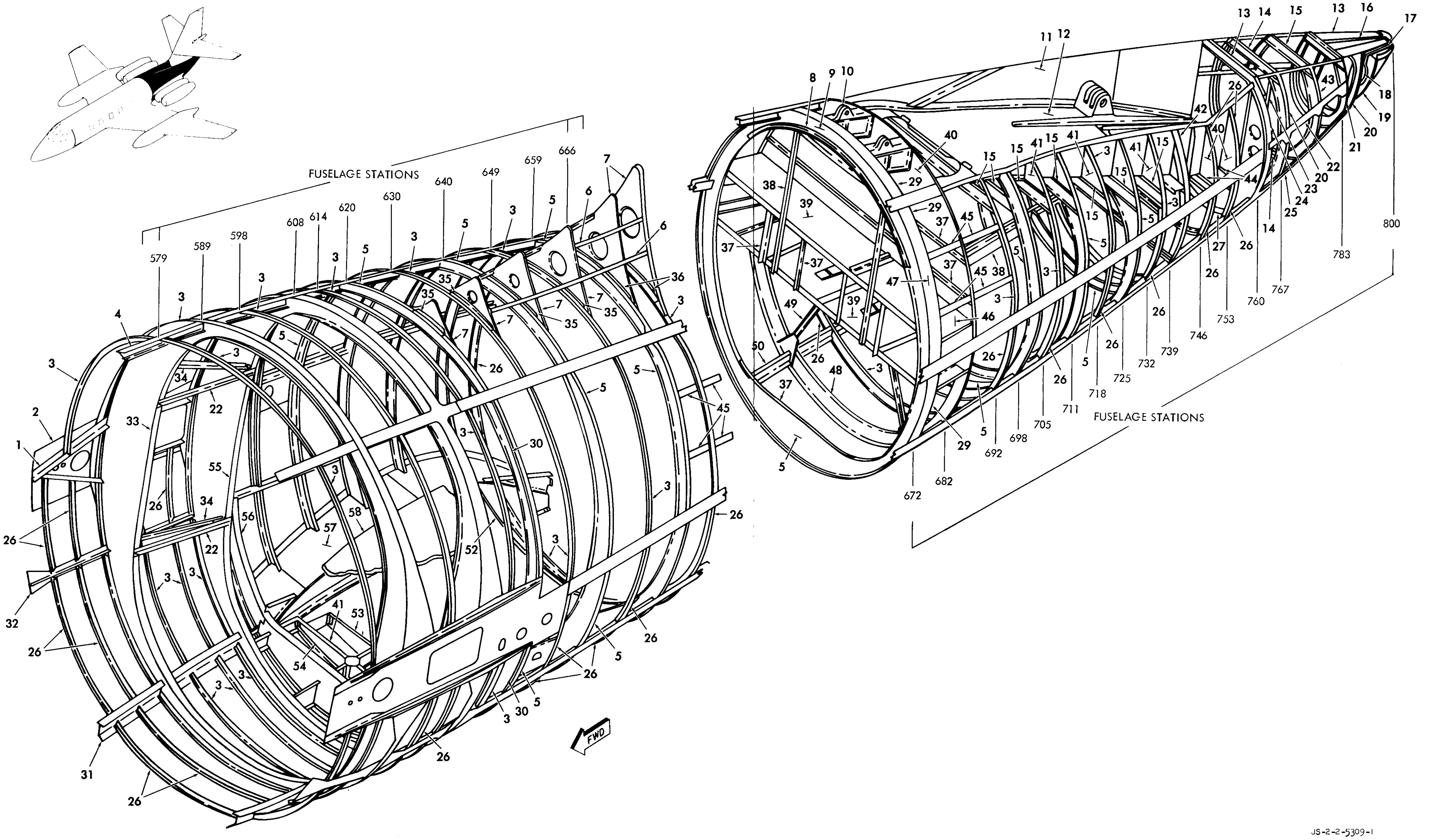


Figure 53-9. Aft Fuselage Structure (Sheet 1 of 4)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	LONGERON ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	DOUBLER	0.050	2024-T4 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.040	2024-T4 CLAD
2	LONGERON ASSEMBLY		
	LONGERON		7075-T6 EXTRUSION
	DOUBLER	0.032	2024-T4 CLAD
3	RING SEGMENT		7075-T6 EXTRUSION
4	LONGERON ASSEMBLY		
	CHANNEL	0.063	7075-T6 CLAD
	CHANNEL	0.071	7075-T6 CLAD
	BAR	0.250	7075-T6 CLAD
	SPLICE STRAP	0.050	2024-T3 CLAD
	DOUBLER	0.040	7075-T6 CLAD
5	BULKHEAD	0.050	7075-T6 CLAD
6	STRINGER		7075-T6 EXTRUSION
7	RIB	0.040	2024-T4 CLAD
8	ANGLE	0.070	7075-T6 CLAD
9	FRAME	0.063	7075-T6 CLAD
10	FITTING		2014-T6 FORGING
11	LONGERON ASSEMBLY		
	ANGLE	0.090	7075-T6 CLAD
	WEB	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	SPLICE PLATE	0.063	7075-T6 CLAD
	SPLICE PLATE	0.070	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	DOUBLER	0.040	7075-T6 CLAD
	WEB	0.032	7075-T6 CLAD
12	SHELF ASSEMBLY		
	DOUBLER	0.063	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.040	7075-T6 CLAD
	WEB	0.032	7075-T6 CLAD
	DOOR	0.040	7075-T6 CLAD
13	FORMER	0.032	7075-T6 CLAD
14	CHANNEL	0.040	7075-T6 CLAD
15	CHANNEL	0.050	7075-T6 CLAD
16 5041 & UP	CHANNEL	0.020	2024-T4 CLAD
17	RIB	0.040	7075-T6 CLAD
18	CAN	0.032	2024-T4 CLAD
19	FORMER	0.040	7075-T6 CLAD
20	RING	0.032	7075-T6 CLAD
21	RING	0.080	7075-T6 CLAD
22	LONGERON		7075-T6 EXTRUSION
23	ANGLE	0.050	7075-T6 CLAD
24	BULKHEAD ASSEMBLY		
	BULKHEAD	0.040	7075-T6 CLAD
	CHANNEL	0.032	7075-T6 CLAD
25	LONGERON	0.080	7075-T6 CLAD
26	RING SEGMENT		7075-T6 EXTRUSION
27	RING SEGMENT		7075-T6 EXTRUSION
28	OMITTED		
29	SPLICE	0.050	7075-T6 CLAD
30	ANGLE	0.040	7075-T6 CLAD
31	LONGERON ASSEMBLY		
	ANGLE	0.063	7075-T6 CLAD

JS-2-2-5309-2

Figure 53-9. Aft Fuselage Structure (Sheet 2 of 4)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
	TEE		7075-T6 EXTRUSION
	LONGERON	0.063	7075-T6 CLAD
	STOP	0.100	7075-T6 CLAD
	DOUBLER	0.050	7075-T6 CLAD
32	LONGERON ASSEMBLY		
	LONGERON		7075-T6 EXTRUSION
	DOUBLER	0.040	2024-T4 CLAD
33	BULKHEAD ASSEMBLY		
	CHANNEL	0.063	7075-T6 CLAD
	CHANNEL CAP	0.080	7075-T6 CLAD
	OUTER CAP	0.080	7075-T6 CLAD
	CHANNEL CAP	0.071	7075-T6 CLAD
	INNER CAP	0.071	7075-T6 CLAD
	STRAP	0.071	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.071	7075-T6 CLAD
	SUPPORT	0.040	2024-T3 CLAD
	FITTING		2014-T6 FORGING
34	SUPPORT		7075-T6 EXTRUSION
35	ANGLE	0.040	2024-T4 CLAD
36	ANGLE	0.062	2024-T4 CLAD
37	ANGLE		7075-T6 EXTRUSION
38	STIFFENER		2024-T4 EXTRUSION
39	BEAM ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	CHANNEL	0.063	7075-T6 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	WEB	0.032	7075-T6 CLAD
40	WEB	0.032	7075-T6 CLAD
41	WEB	0.050	7075-T6 CLAD
42	BULKHEAD ASSEMBLY		
	CAP	0.072	7075-T6 CLAD
	WEB	0.072	7075-T6 CLAD
	CHANNEL	0.090	7075-T6 CLAD
	CHANNEL	0.063	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	DOUBLER	0.080	7075-T6 CLAD
	FITTING		2014-T6 FORGING
43	RING	0.040	7075-T6 CLAD
44	CHANNEL	0.032	7075-T6 CLAD
45	INTERCOSTAL		7075-T6 EXTRUSION
46	PAN	0.063	2024-T4 CLAD
47	FRAME	0.040	7075-T6 CLAD
48	ZEE	0.063	7075-T6 CLAD
49	HAT	0.040	2024-T3 CLAD
50	LONGERON ASSEMBLY		
	LONGERON	0.063	7075-T6 CLAD
	DOUBLER	0.063	7075-T6 CLAD
	SKIN SPLICE	0.063	2024-T4 CLAD
51	OMITTED		
52	BULKHEAD ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.071	7075-T6 CLAD
	STRAP	0.125	7075-T6 CLAD
53	ZEE	0.050	7075-T6 CLAD
54	TEE		7075-T6 EXTRUSION
55	BULKHEAD ASSEMBLY		
	CHANNEL	0.071	7075-T6 CLAD

JS-2-2-5309-3

Figure 53-9. Aft Fuselage Structure (Sheet 3 of 4)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
	OUTER CAP	0.102	7075-T6 CLAD
	INNER CAP	0.125	7075-T6 CLAD
	SPLICE PLATE	0.125	7075-T6 CLAD
	ANGLE	0.080	7075-T6 CLAD
	DOUBLER	0.070	7075-T6 CLAD
	FITTING		2014-T6 FORGING
56	BULKHEAD ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	TEE		2024-T42 EXTRUSION
	WEB	0.050	7075-T6 CLAD
	WEB	0.063	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	SPLICE	0.050	7075-T6 CLAD
	FITTING		4130 STEEL
	SUPPORT		7075-T6 EXTRUSION
57	TORQUE BOX ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	BRACLET	0.063	7075-T6 CLAD
	BRACKET	0.081	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	TOP WEB	0.050	7075-T6 CLAD
	TOP WEB	0.063	7075-T6 CLAD
	SIDE WEB	0.063	7075-T6 CLAD
58	PRESSURE PANEL AND DOOR ASSEMBLY		
	CHANNEL	0.063	7075-T6 CLAD
	CHANNEL	0.071	7075-T6 CLAD
	WEB CAP	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.063	2024-T3 CLAD
	PANEL SKIN	0.020	7075-T6 CLAD
	HONEYCOMB CORE		5052-438
	DOUBLER	0.020	7075-T6 CLAD
	ZEE SUPPORT	0.040	2024-T4 CLAD
	DOOR SKIN	0.020	7075-T6 CLAD
	DOUBLER	0.040	7075-T6 CLAD
	HINGE		2024-T4 EXTRUSION

JS-2-2-5309-4

Figure 53-9. Aft Fuselage Structure (Sheet 4 of 4)

CHAPTER 54
NACELLES AND PYLONS

Contents

Page

DESCRIPTION 54-1
General 54-1
Nose Cowl Assembly 54-1
Fixed Forward Nacelle Structure 54-1
Box Beams 54-1
Engine Mounting 54-1
Fixed Aft Nacelle Structure 54-2
Engine Removal Doors 54-2
Longitudinal Firewall 54-2
Thrust-Reverser and Stang Assemblies 54-2
Pylon Fairings 54-2
Minor Repair of Firewall Seal 54-2

FIGURES
54-1. Nacelle Group 54-3
54-2. Nacelle Nose Lip, Forward Inlet Duct, Stang, and Thrust-Reverser Skin and Structure 54-4
54-3. Nacelle and Pylon Skin 54-6
54-4. Nacelle Structure 54-8
54-5. Screws Used in Box Beam Skins 54-10

54.1 DESCRIPTION.

54.1.1 GENERAL. (See figure 54-1.) The nacelles are of aluminum, titanium, and steel construction. The left and right nacelles, each containing compartments for two engines, are bolted to the fuselage frames at fuselage stations 589.5 and 608 with external wrenching bolts.

54.1.2 NOSE COWL ASSEMBLY. The nose cowl assembly is of aluminum alloy construction and consists of a removable nose lip assembly from nacelle stations 31 to 43 and a fixed cowling assembly from nacelle stations 43 to 66. The nose lip is a welded assembly. The fixed cowling consists of frames, longerons, skin, and removable panels. The forward inlet duct assembly is housed inside the fixed cowling.

54.1.3 FIXED FORWARD NACELLE STRUCTURE. The three-beam structure from nacelle stations 66 to 100 consists of the two side skin panels and the longitudinal firewall. The skin panels are made of aluminum and steel frames covered with aluminum and titanium skins. The longitudinal firewall is made of titanium and steel webs and stiffeners.

54.1.4 BOX BEAMS. The two steel box beam frames at nacelle stations 81.5 and 100 are I-sections and are in the shape of a double yoke as viewed looking aft or forward. These frames are covered with steel skins on the top and bottom. The screws used for attaching the top box beam skins are shown in figure 54-5. There is no structural requirement that the engine be removed prior to removal of the top box beam skins.

54.1.5 ENGINE MOUNTING. Each engine has three support points: two rear mounts and one forward mount. The forward engine mount consists of a short, machined steel fitting which is canti-levered from the forward box beam frame at nacelle station 81.5. The aft engine mounts consists of steel fittings bolted to the box beam assembly at nacelle station 100. Diagonal steel tie members extend from each of the aft mounts forward to fittings on the forward box beam frame, except at the inboard-most fitting, where a forward shear tie is used.

54.1.6 FIXED AFT NACELLE STRUCTURE. The fixed aft nacelle structure from nacelle station 100 to 148 forms a structural unit with the engine removal doors. The fixed transverse firewall is at nacelle station 117. The firewall and the frame at nacelle station 148 are of steel construction. The rest of the aft nacelle consists of aluminum longerons and frames covered with aluminum and titanium skins.

54.1.7 ENGINE REMOVAL DOORS. There are four access doors on the bottom of each nacelle for engine removal (two for each engine). The forward doors are hinged on the side opposite the longitudinal firewall, they extend from nacelle station 66 to 112, and each is held closed by three latch assemblies. The two main latches are located along the longitudinal firewall. There is also a secondary latch along the forward edge to prevent door deflection. The forward doors are made primarily of aluminum allow frame and longeron construction (aircraft 5001 thru 5045). Steel cross frames are used between hinge and latch points (aircraft 5046 and up). The skin covering is of aluminum and steel. The two aft doors extend from nacelle station 112 to 148. These doors are primarily of aluminum skin and stiffener construction with quarter turn fasteners on three sides.

The forward (hinged) engine removal doors should be latched as follows: First close the forward main latch assuring positive engagement with local compression of the seal along the longitudinal firewall. Secondly, close the aft main latch assuring positive engagement with seal compression along the full length of the door. The latch on the forward edge is closed last. When this latch is closed, a local tightening of the door against the nacelle should be observed and there should be no forward facing step on the leading edge of the door. All latch assemblies are adjustable in order that the amount of door compression may be regulated to meet these requirements.

CAUTION

With the anti-deflector latch assembly improperly adjusted, there is a possibility that the engine access door assembly may be lost in flight if the forward latch assembly (Figure 52-5, Item 40) is not latched securely.

To check for this condition, release the forward latch and check the forward edge of the door assembly to see if it droops below a "faired in" position. If it does not fair properly adjust the anti-deflector latch until the door is faired. The latch assembly is adjusted by rotating the grooved nut on the latch mechanism with a common screwdriver. After the anti-deflector latch is properly adjusted close the forward latch assembly and check for a slight compression of the door. If a slight compression is not noted, the forward latch should be adjusted in the same manner as the anti-deflector latch in order to relieve the load on the anti-deflector latch.

54.1.8 LONGITUDINAL FIREWALL. A longitudinal titanium and steel firewall which separates the engine compartments extends from the inlet duct assembly at nacelle station 66 to the stang assembly at nacelle station 160. The firewall is primarily made of titanium and steel webs and stiffeners, with steel skins covering the top.

54.1.9 THRUST-REVERSER AND STANG ASSEMBLIES. A thrust-reverser assembly for each engine is installed aft of nacelle station 148. This assembly consists of two steel clamshell-type doors, a steel longeron assembly, and a steel ejector assembly. A steel stang assembly is installed at the longitudinal firewall area at nacelle station 160. This stang contains the engine tailpipe assemblies and the aft portion of the longitudinal firewall.

54.1.10 PYLON FAIRINGS. Fairing structure is provided between each nacelle assembly and the fuselage to maintain smooth contour at the leading and trailing edges of the nacelles. The fairing consists of aluminum, titanium, and steel fillet assemblies.

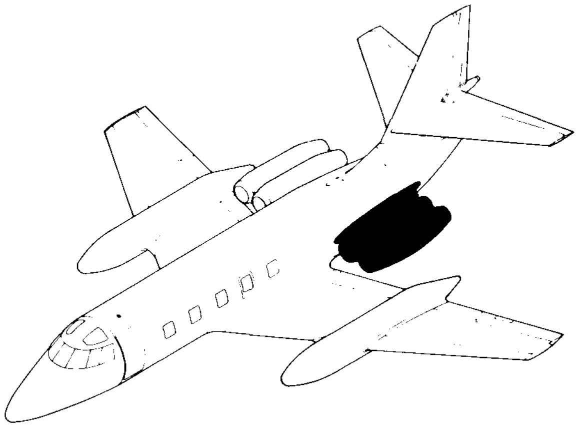
54.1.11 MINOR REPAIR OF FIREWALL SEAL. The teflon covered inconel mesh fire seals mounted on the fixed aft nacelle and the aft engine removal door at nacelle station 148 sometimes develop minor damage in the teflon cover due to normal wear and tear. This damage can be satisfactorily repaired as outlined below but is should be noted that this repair is not completely fuel and oil resistant.

a. Remove the seal locally at the damaged area.

b. Wrap the seal bulb and tail completely with two wraps of Connecticut Hand Rubber "Temp-R-Tape TV" or equal (MIL-I-23594, Type 1, .0065 thick). Two-inch width tape is suggested.

c. Reinstall the seal with monel rivets or small screws.

INDEX NO.	COMPONENT	FIG. NO.
1	NOSE LIP	54-2
2	PYLON	54-3
3	STANG	54-2
4	THRUST REVERSER	54-2
5	NACELLE STRUCTURE	54-4
6	BOX BEAM SKIN	54-3
7	NACELLE SKIN	54-3
8	FORWARD INLET DUCT	54-2



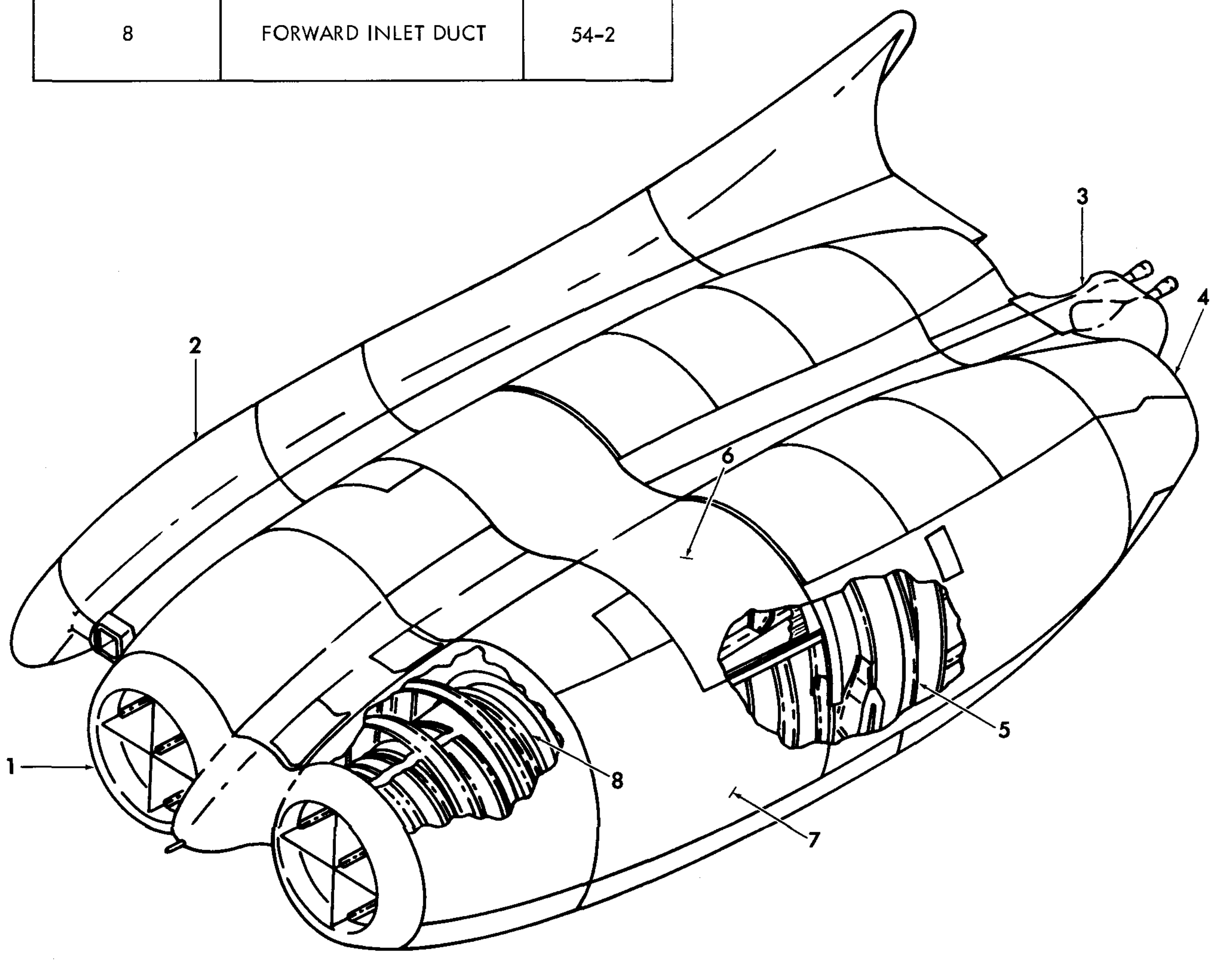


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Figure 54-1. Nacelle Group

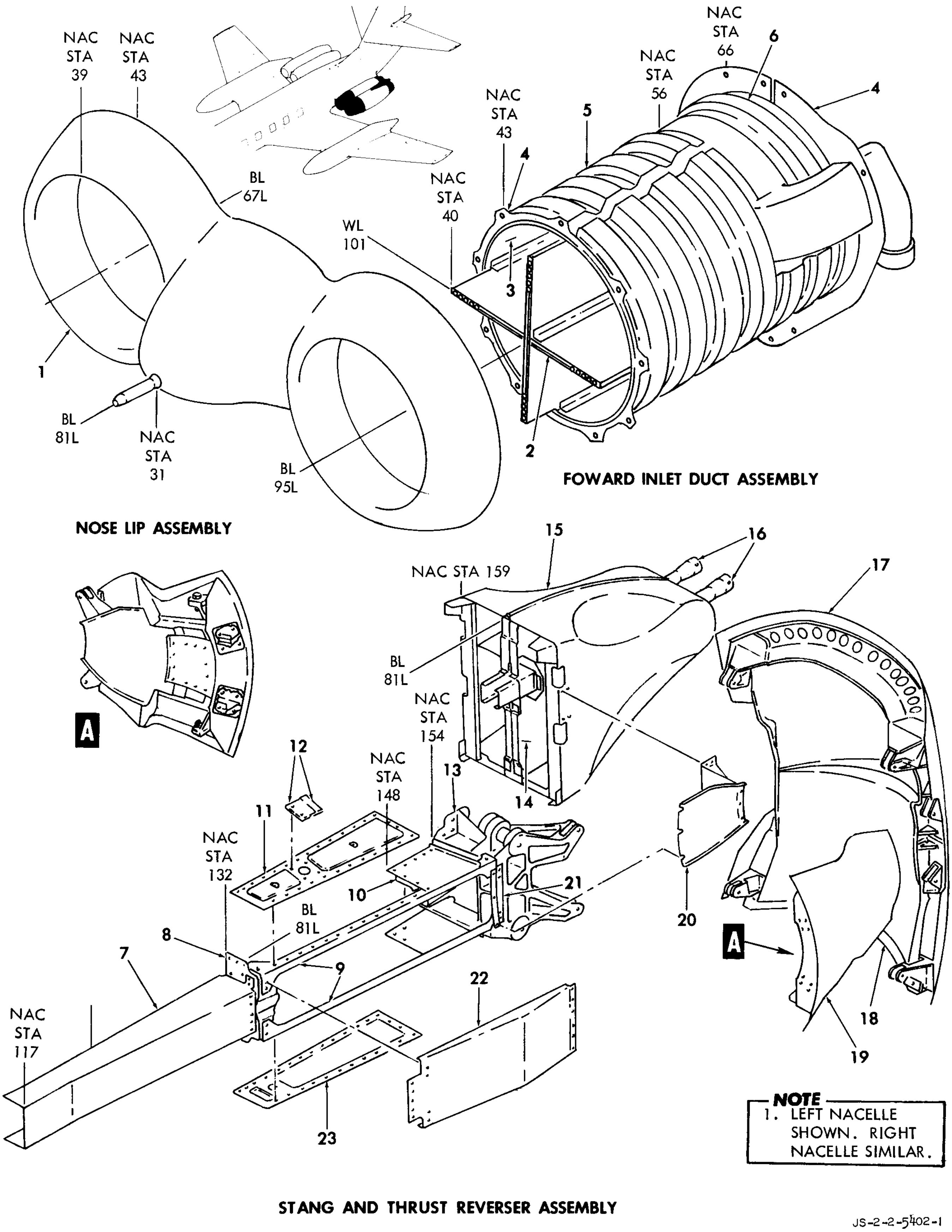

Figure 54-2. Nacelle Nose Lip, Forward Inlet Duct, Stang, and Thrust-Reverser Skin and Structure (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	NOSE LIP ASSEMBLY		
	LIP	0.040	6061-T6
	BAFFLE (INNER LINER)	0.025	6061-T6
	DUCT	0.050	6061-T6
	ANGLE	0.050	6061-T6
	TUBE	0.049	6061-T6
	BULLET	0.050	6061-T6
2	CRUCIFORM ASSEMBLY		
	VANE, HORIZONTAL	0.100	6061-T6
	VANE, VERTICAL	0.125	6061-T6
	ANGLE	0.050	6061-T6
3	SKIN	0.032	6061-T6
4	RING	0.032	6061-T6
5	BEADED SKIN	0.020	6061-T6
6	ZEE	0.032	6061-T6
7	TUNNEL ASSEMBLY		
	TUNNEL	0.016	347 CRES
	FOIL	0.002	302 CRES
	VENTILATOR	0.012	302 CRES
	DOUBLER	0.016	347 CRES
8	FITTING		7075-T651 EXTRUSION
9	BEAM ASSEMBLY		
	ANGLE	0.072	301 CRES - 1/4 HARD
	WEB	0.032	301 CRES - 1/4 HARD
10	TRACK		A-286 CRES
11	COVER ASSEMBLY		
	COVER	0.032	301 CRES - 1/4 HARD
	FOIL	0.004	302 CRES
	VENT	0.012	302 CRES ANNEALED
12	FAIRLEAD	0.025	MELAMINE GLASS FIBER.
13	FITTING		AM355 CRES CASTING
14	FIREWALL ASSEMBLY		
	DUCT	0.016	347 CRES ANNEALED
	ANGLE	0.032	321 CRES ANNEALED
	FIREWALL	0.032	321 CRES ANNEALED
	WEB	0.032	321 CRES ANNEALED
	DOUBLER	0.032	321 CRES ANNEALED
	SPLICE	0.080	321 CRES - 1/4 HARD
15	SHELL	0.040	321 CRES ANNEALED
16	TAILPIPE ASSEMBLY		
	TAILPIPE	0.035	347 CRES ANNEALED
	FLANGE	0.040	321 CRES ANNEALED
17	UPPER DOOR ASSEMBLY		
	SKIN, OUTER	0.032	A-286 CRES
	SKIN, INNER	0.040	A-286 CRES
	LONGERON	0.050	A-286 CRES
	BRACKET	0.040	A-286 CRES
	PLATE	0.063	A-286 CRES
	ZEE	0.040	A-286 CRES
	FRAME	0.040	A-286 CRES
	FITTING		A-286 CRES
	FITTING		AM355 CRES CASTING
	FILLET	0.040	A-286 CRES
18	LOWER DOOR ASSEMBLY		
	SKIN, OUTER	0.032	A-286 CRES
	SKIN, INNER	0.040	A-286 CRES
	LONGERON	0.050	A-286 CRES
	BRACKET	0.040	A-286 CRES
	PLATE	0.063	A-286 CRES
	ZEE	0.040	A-286 CRES
	FRAME	0.040	A-286 CRES
	FITTING		A-286 CRES
	FITTING		AM355 CRES CASTING
	FILLET	0.040	A-286 CRES
	LONGERON	0.040	A-286 CRES
	SUPPORT	0.040	A-286 CRES
	ANGLE	0.040	A-286 CRES
19	LONGERON ASSEMBLY		
	CHANNEL	0.040	A-286 CRES
	CHANNEL	0.050	A-286 CRES
	STIFFENER	0.040	A-286 CRES
	ZEE	0.032	A-286 CRES
	COVER	0.032	A-286 CRES
	OUTER SKIN	0.032	A-286 CRES
	INNER SKIN	0.040	A-286 CRES
	GUSSET	0.032	A-286 CRES
	FITTING		AM355 CRES CASTING
	FITTING		4340 STEEL
	COVER (INBOARD LONGERON ONLY)	0.040	A-286 CRES
	FAIRING (INBOARD LONGERON ONLY)	0.025	347A CRES ANNEALED
20	EJECTOR ASSEMBLY		
	SKIN	0.040	A-286 CRES
	CHANNEL	0.040	A-286 CRES
	ANGLE	0.040	A-286 CRES
21	ZEE	0.032	A-286 CRES
22	TUNNEL ASSEMBLY.		
	TUNNEL	0.016	347 CRES
	FOIL	0.002	302 CRES
	VENTILATOR	0.012	302 CRES
23	COVER ASSEMBLY		
	COVER	0.032	301 CRES - 1/4 HARD
	FOIL	0.004	302 CRES

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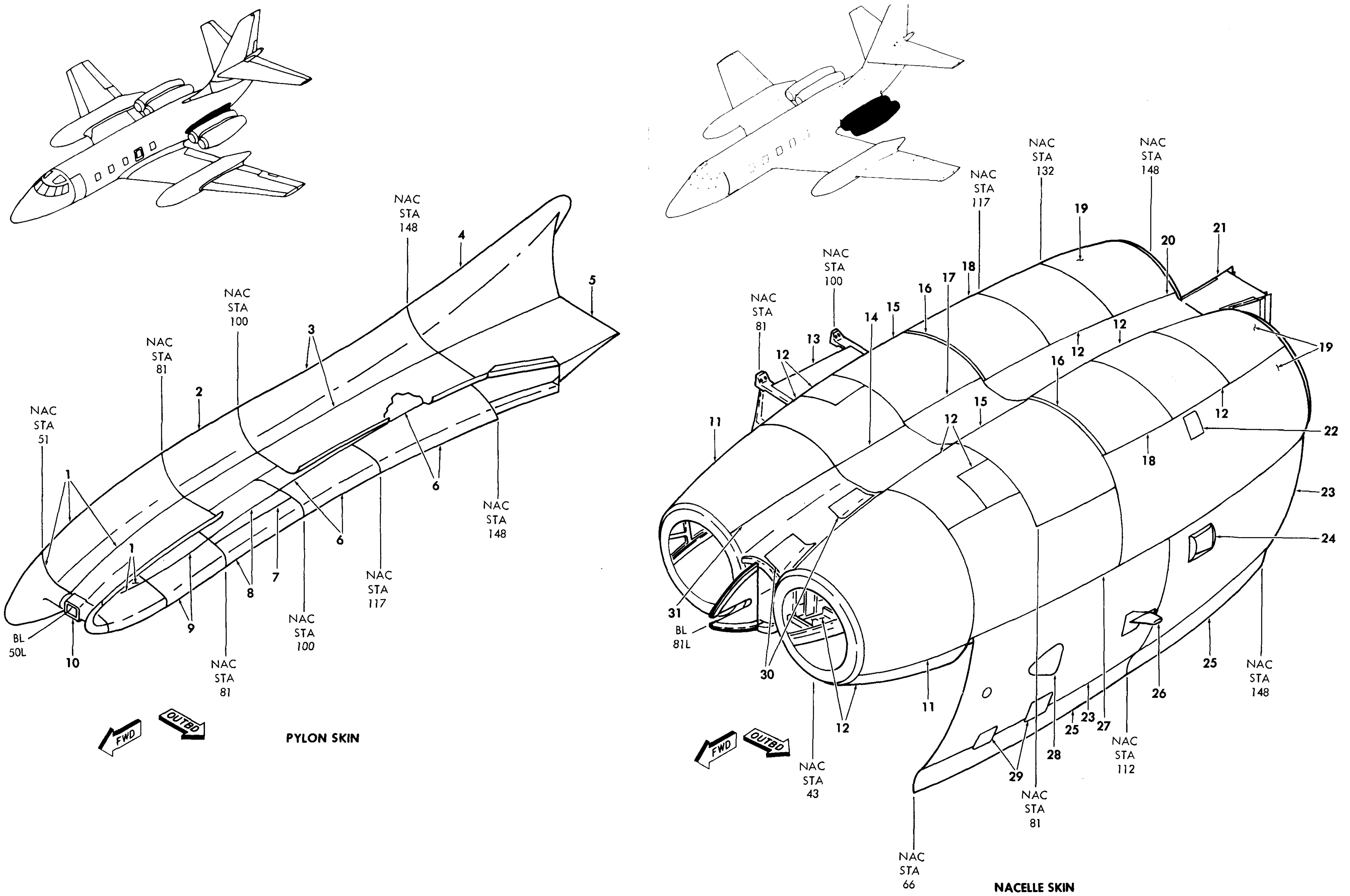
NOTE

1. REFER TO CHAPTER 51 FOR NEGLIGIBLE DAMAGE LIMITS AND TYPICAL REPAIRS.
2. REFER TO CHAPTER 78 FOR MINOR REPAIR OF THRUST REVERSER
3. MINOR CRACKS IN 321 CRES ANNEALED SHEET MAY BE REPAIRED BY INERT-ARC WELDING WITH 347 CRES ROD.

Figure 54-2. Nacelle Nose Lip, Forward Inlet Duct, Stang, and Thrust-Reverser Skin and Structure (Sheet 2 of 2)



PYLON SKIN

NACELLE SKIN

Figure 54-3. Nacelle and Pylon Skin (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	FILLET	0.028	2024-T4 CLAD
2	FILLET	0.032	2024-T4 CLAD
3	FILLET	0.025	2024-T4 CLAD
4	FILLET ASSEMBLY		
	FILLET	0.025	347-A CRES ANNEALED
	STIFFENER	0.025	347-A CRES ANNEALED
5	FILLET ASSEMBLY		
	FILLET	0.025	347-A CRES ANNEALED
	CLOSING RIB	0.025	347-A CRES ANNEALED
	STIFFENER	0.025	347-A CRES ANNEALED
6	FILLET	0.020	AMS 4900 TITANIUM
7	DOOR	0.040	AM 350 CRES
8	FAIRING	0.025	AMS 4900 TITANIUM
9	FILLET	0.020	AMS 4901 TITANIUM
10	DUCT ASSEMBLY		
	DUCT	0.045	6061-T6 BARE
	FAIRING	0.032	6061-T6 BARE
11	SKIN	0.028	2024-T4 CLAD
12	SKIN	0.025	2024-T4 CLAD
13	DOOR	0.050	AM 350 CRES
14	SKIN	0.020	347-A CRES ANNEALED
15	SKIN ASSEMBLY		
	DOUBLER	0.040	301 CRES FULL HARD
	SKIN	0.020	AM 350 CRES
16	STRAP		2024-T4 EXTRUSION
17	SKIN	0.020	AM 350 CRES
18	SKIN	0.032	2024-T3 CLAD
19	SKIN (OUTBOARD SIDE, OUTBOARD NACELLE)	0.028	2024-T6 CLAD
	SKIN (INBOARD SIDE, INBOARD NACELLE)	0.020	AMS 4901 TITANIUM
20	SKIN	0.025	302 CRES
21	SKIN	0.025	347 CRES
22	DOOR ASSEMBLY		
	FILLER	0.032	2024-T6 CLAD
	SKIN (OUTBOARD SIDE, OUTBOARD NACELLE)	0.028	2024-T6 CLAD
	SKIN (INBOARD SIDE, INBOARD NACELLE)	0.020	AMS 4901 TITANIUM
23	SKIN	0.025	2024-T6 CLAD
24	DUCT ASSEMBLY		
	DUCT	0.06	LAMINATED GLASS FABRIC
	SCOOP	0.025	2024-T6 CLAD
	LIP		LAMINATED GLASS FABRIC
25	SKIN	0.020	302 CRES ANNEALED
26	MAST ASSEMBLY		
	MAST	0.080	6061-T4 BARE
	FLANGE	0.080	6061-T4 BARE
27	SKIN (OUTBOARD SIDE, OUTBOARD NACELLE)	0.028	2024-T6 CLAD
	SKIN (INBOARD SIDE, INBOARD NACELLE)	0.020	AMS 4901 TITANIUM
28	FLAP ASSEMBLY		
	DEFLECTOR	0.025	2024-T4 CLAD
	FLAP	0.025	2024-T4 CLAD
29	DOOR	0.020	302 CRES ANNEALED
30	ACCESS PANEL	0.032	2024-T4 CLAD
31	SKIN	0.032	2024-T4 CLAD

NOTE: REFER TO CHAPTER 51 FOR NEGLIGIBLE DAMAGE LIMITS AND TYPICAL REPAIRS

JS-2-2-5403-2

Figure 54-3. Nacelle and Pylon Skin (Sheet 2 of 2)

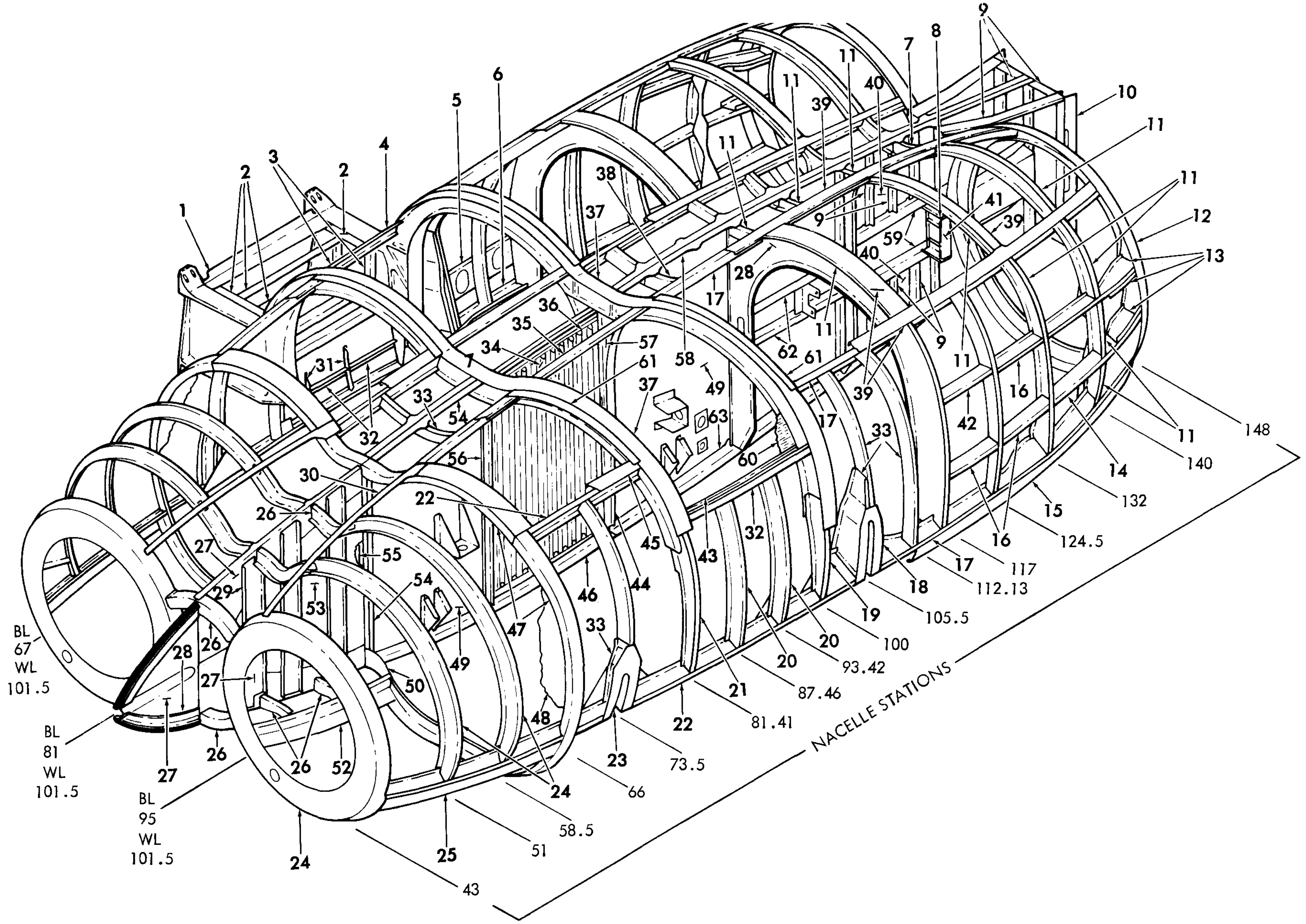


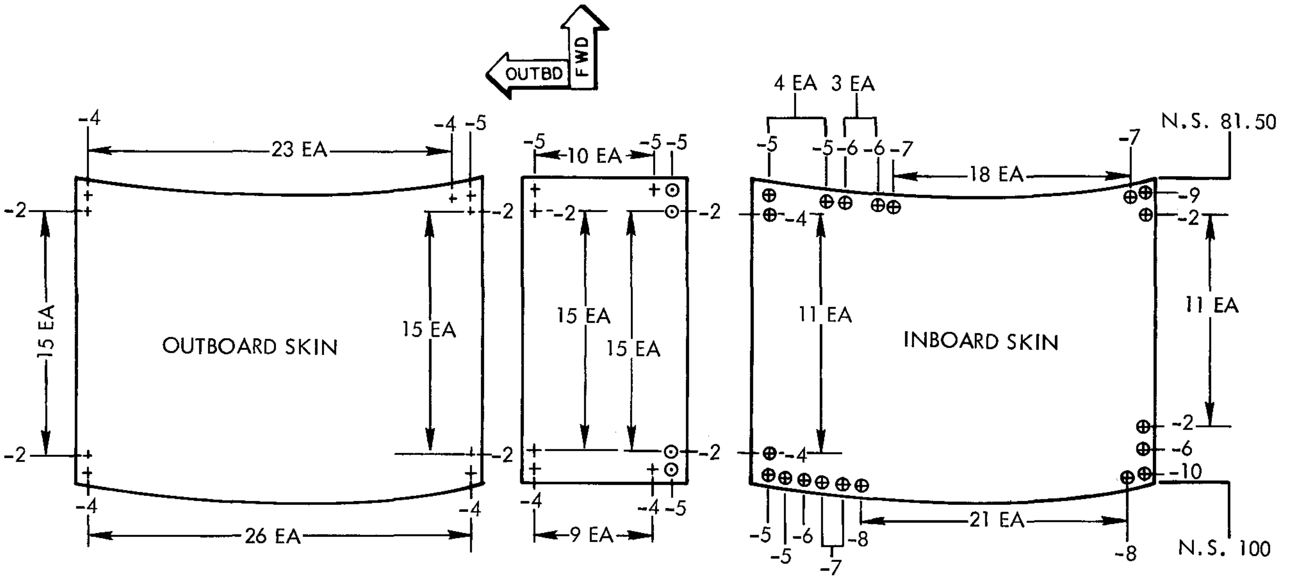
Figure 54-4. Nacelle Structure (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	ANGLE	0.040	AM350 CRES
2	ANGLE	0.063	AM350 CRES
3	ANGLE	0.090	AM350 CRES
4	HAT	0.063	AM350 CRES
5	SUPPORT	0.032	AMS4901 TITANIUM
6	HAT	0.050	AM350 CRES
7	FRAME	0.040	2024-T6 BARE
8	CHANNEL	0.050	301 CRES - 1/4 HARD
9	ANGLE	0.032	347 CRES
10	ANGLE	0.050	347 CRES
11	FRAME	0.032	2024-T6 BARE
12	FRAME	0.040	AM350 CRES
13	CHANNEL	0.050	2024-T6 BARE
14	INTERCOSTAL	0.040	2024-T6 BARE
15	CHANNEL	0.063	2024-T6 CLAD
16	INTERCOSTAL	0.032	2024-T6 CLAD
17	LONGERON	0.040	2024-T6 BARE
18	DOUBLER	0.032	2024-T3 CLAD
19	FRAME	0.063	2024-T4 BARE
20	FRAME	0.028	2024-T4 BARE
21	FRAME	0.050	AM350 CRES
22	LONGERON	0.040	2024-T4 BARE
23	DOUBLER	0.025	2024-T3 CLAD
24	FRAME	0.032	2024-T4 CLAD
25	SUPPORT	0.050	2024-T4 CLAD
26	FORMER	0.032	2024-T4 CLAD
27	WEB	0.020	AMS4900 TITANIUM
28	RETAINER	0.020	302 CRES
29	CHANNEL	0.025	347 CRES ANNEALED
30	DOUBLER	0.028	2024-T3 CLAD
31	ANGLE	0.032	2024-T4 BARE
32	ANGLE	0.040	2024-T4 CLAD
33	FRAME	0.032	2024-T4 BARE
34	WEB	0.016	AMS4900 TITANIUM
35	HAT	0.040	AM350 CRES
36	ANGLE	0.040	301 CRES FULL HARD
37	FRAME		4340 STEEL, HT TR 180,000 TO 200,000 PSI.
38	FRAME	0.040	20240T4 BARE
39	SPLICE	0.040	2024-T6 BARE
40	WEB	0.036	AMS4901 TITANIUM
41	WEB	0.040	17-7PH CRES, HT TR 18,000 TO 215,000 PSI
42	CHANNEL	0.040	2024-T6 CLAD
43	CHANNEL	0.063	AM350 CRES
44	CHANNEL	0.056	AMS4901 TITANIUM
45	ANGLE	0.040	2024-T4 BARE
46	ANGLE	0.032	AMS4901 TITANIUM
47	CAP	0.040	2024-T4 CLAD
48	WEB	0.032	2024-T3 CLAD
49	WEB	0.016	AMS4901 TITANIUM
50	FORMER	0.040	2024-T4 CLAD
51	LONGERON	0.040	2024-T4 CLAD
52	LONGERON	0.050	2024-T4 CLAD
53	BULKHEAD	0.020	AMS4900 TITANIUM
54	ANGLE	0.020	AMS4901 TITANIUM
55	ANGLE	0.025	AMS4900 TITANIUM
56	CHANNEL	0.040	AM350 CRES ANNEALED
57	CHANNEL	0.032	AM350 CRES
58	CAP	0.040	AMS4901 TITANIUM
59	ANGLE	0.080	301 CRES - 1/4 HARD
60	LOWER SKIN ASSEMBLY		
	SKIN	0.020	AM350 CRES
	BEADED SKIN	0.020	AM350 CRES
61	STRAP		2024-T4 EXTRUSION
62	ANGLE	0.020	301 CRES 1/4 HARD
63	CAP	0.032	AMS4901 TITANIUM

NOTE: REFER TO CHAPTER 51 FOR NEGLIGIBLE DAMAGE LIMITS AND TYPICAL REPAIRS

JS-2-2-5404-2

Figure 54-4. Nacelle Structure (Sheet 2 of 2)



L/H NACELLE SHOWN, R/H OPPOSITE

	SCREWS △1		NUTPLATES	
LEGEND	PREFERRED	ALTERNATE	FLOATING △2	SELF-ALIGNING △3
+	NAS 1623	LS 5900-3	NAS686A3	52LHA3022-02
⊙	NAS 1624	LS 5900-4	NAS686A4	F1967-4 △4
⊕	NAS 1625	LS 5900-5	NAS686A5	12LHA3022-054

△1 GRIP LENGTH DASH NUMBERS SHOWN (ACTUAL LENGTHS USED MAY BE INCREASED ONE OR TWO DASH NUMBERS AS REQUIRED).

△2 MS21059 SERIES NUTPLATES ARE INTERCHANGEABLE WITH NAS 686.

△3 SELF-ALIGNING NUTPLATES ARE USED ONLY WHERE INSTALLED ON THE ANGULAR SURFACE OF THE MAIN FRAME FORGINGS.

△4 USED ONLY WITH NAS 1624-5 SCREWS (2 PLACES)

Figure 54-5. Screw Used In Box Beam Skins

JS-2-0-5405

CHAPTER 55
STABILIZERS

Contents Page

GENERAL 55-1

VERTICAL STABILIZER BOX STRUCTURE 55-1

HORIZONTAL STABILIZER BOX STRUCTURE 55-2

VERTICAL AND HORIZONTAL STABILIZER LEADING EDGES 55-2

VERTICAL AND HORIZONTAL STABILIZER TRAILING EDGES 55-2
  Vertical Stabilizer Trailing Edge - Lower Section 55-2

VERTICAL AND HORIZONTAL STABILIZER TIPS 55-2

FAIRINGS 55-2

RUDDER 55-2

ELEVATOR 55-2

DETAIL STABILIZER COMPONENT MATERIAL AND IDENTIFICATION 55-2

FIGURES
  55-1. Empennage Group 55-3
  55-2. Lower Trailing Edge and Trailing Edge Tee Assembly 55-4
  55-3. Empennage Tips and Fairing Skin 55-6
  55-4. Leading Edge Skin and Structure 55-8
  55-5. Rudder and Elevator Skin and Structure 55-9
  55-6. Horizontal Stabilizer Box Section Skin and Structure 55-10
  55-7. Vertical Stabilizer Box Section Skin and Structure 55-11

55.1 GENERAL. (See figure 55-1.) The empennage group consists of the vertical stabilizer, rudder, rudder tab, horizontal stabilizer, elevator, fuel jettison nozzle, stabilizer tips, and fairings. The empennage is attached to the fuselage by a scissors fitting, two pitch trim actuator fittings, and a pivot fitting at the bottom of the vertical stabilizer. This is a flexible joint which permits the entire empennage to be rotated by the pitch trim actuator to trim the airplane. The bolts at these attach fittings may be removed to permit removal of the entire empennage from the airplane. On the lower vertical stabilizer, a sheet of stainless steel shields the skin from damage due to contact with the fuselage-to-empennage fairing during pitch trim actuation of the empennage.

55.2 VERTICAL STABILIZER BOX STRUCTURE. The vertical stabilizer box is an aluminum alloy structure and consists of two major sections, the upper section (that part above the horizontal stabilizer), and the lower section (that part below the horizontal stabilizer). The box section consists of front and rear beams, ribs, skin and, in the lower section only, skin stiffeners. The pivot fitting which

55.2 (Cont).

provides for pivoting the empennage is attached to the lower portion of the rear beam in the lower box section. The pitch trim actuator attaching fittings and scissors attaching fittings are attached to the lower portion of the front beam in the lower box section. All of the box structure skins are butt jointed and attached by flush rivets.

55.3 HORIZONTAL STABILIZER BOX STRUCTURE. The horizontal stabilizer box is a full cantilever, aluminum alloy structure, and consists of front and rear beams, ribs, stringers, and chemically milled skins. Flush rivets are used throughout for attaching the skin to the structure. The left and right horizontal stabilizer box sections are tied together through the vertical stabilizer by the horizontal stabilizer center section. This is a small box section consisting of ribs, webs, and heavy, chemically milled plates. This box section also serves to tie the horizontal stabilizer to the vertical stabilizer.

55.4 VERTICAL AND HORIZONTAL STABILIZER LEADING EDGES. Both the vertical and horizontal stabilizer leading edges are aluminum alloy structures consisting of ribs and skin. The vertical stabilizer leading edge is continuous through the junction with the horizontal stabilizer. The horizontal leading edges butt-join against the vertical leading edge. All leading edges are afforded ice protection by de-icing boots. All of the leading edges are assembled by flush rivets, are attached to their respective box structures by flush screws, and are removable.

55.5 VERTICAL AND HORIZONTAL STABILIZER TRAILING EDGES. A hollow open box section of aluminum alloy skin and ribs is attached to the rear beams of both the horizontal stabilizer box structure and the upper section of the vertical stabilizer box structure. These trailing edges provide fairing over the leading edges of the elevator and rudder respectively. They are assembled with flush rivets, and are permanently attached to their respective box sections with flush rivets.

55.5.1 VERTICAL STABILIZER TRAILING EDGE LOWER SECTION. The trailing edge for the lower section of the vertical stabilizer is an aluminum alloy structure consisting of skin, ribs, and an auxiliary rear beam. The aft-most part of this trailing edge is of double skin (inner skin beaded) construction, assembled by bonding and attached to the main part of the trailing edge structure at the auxiliary beam by flush rivets. All of the skin and rib structure is assembled by flush rivets, is attached to the vertical stabilizer by flush screws, and is removable.

55.6 VERTICAL AND HORIZONTAL STABILIZER TIPS. The tips for the vertical and horizontal stabilizers are aluminum alloy welded shell assemblies. They are attached to their respective stabilizers with flush screws, and are removable. The vertical stabilizer tip houses the upper anti-collision light.

55.7 FAIRINGS. Aluminum alloy fairings are used at the intersection of the horizontal and vertical stabilizers, at the junction of the vertical stabilizer and fuselage, and around the fuel jettison nozzle. The fairings are of skin and rib construction using flush rivets and spot-welds. All of the fairings are installed by flush screws and are removable. The fuselage-to-vertical-stabilizer fairing is installed on the fuselage with no mechanical connection with the vertical stabilizer. It fits closely around the vertical stabilizer, with a nylon rub strip attached around the upper inside edge of the fairing to prevent scuffing of the stainless steel shield due to contact with the fairing during pitch trim movement of the empennage.

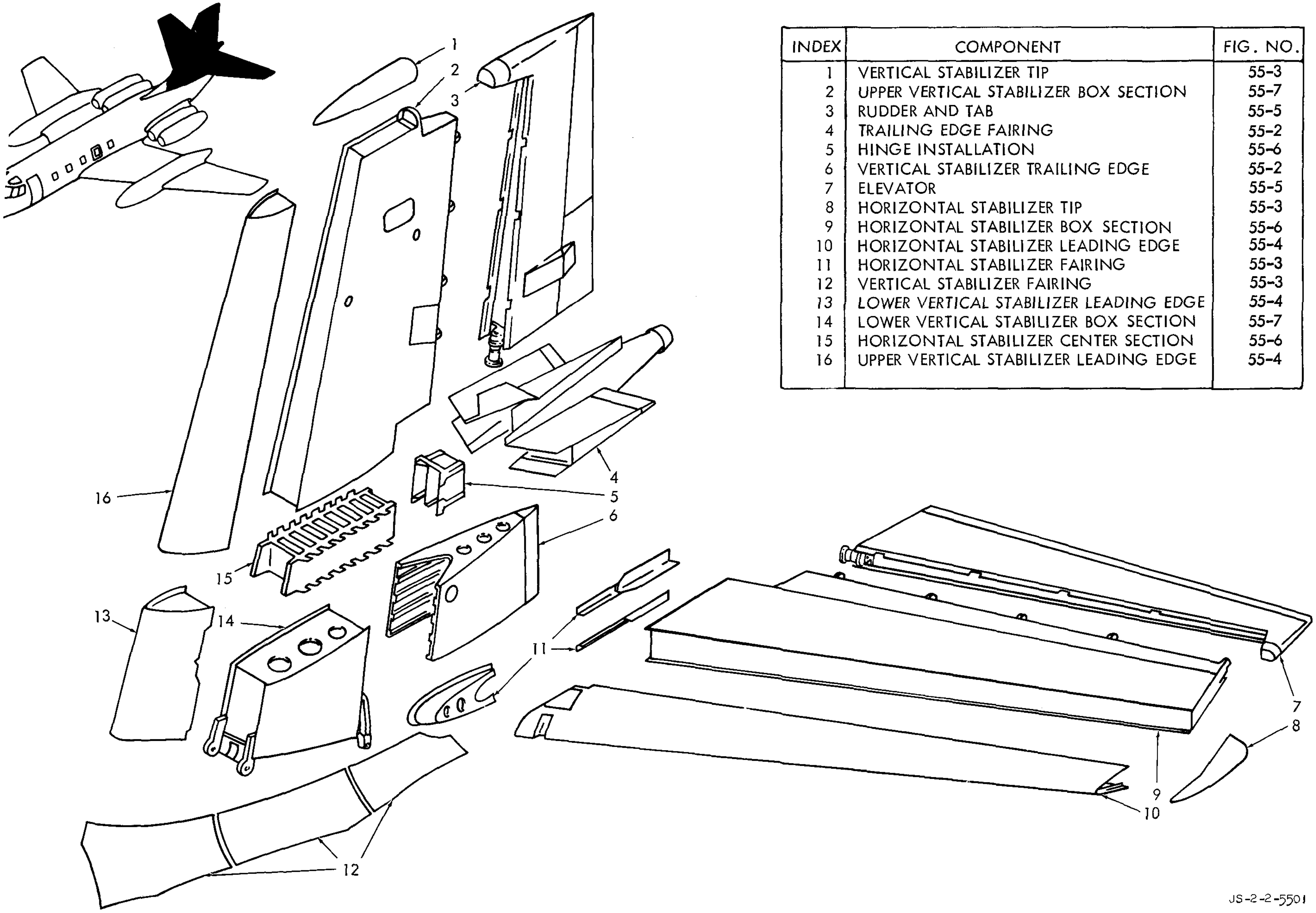
55.8 RUDDER. The rudder is an aluminum alloy structure consisting of front and rear beam, ribs, and unstiffened skin. The rudder is supported at four hinge points which tie to the vertical stabilizer, and at two bearing points which support the rudder torque tube. A horn balance mounting a depleted uranium counterweight forms the rudder tip. Another uranium counterweight is attached at the lower end of the rudder torque tube. The rudder skin is attached by flush rivets. The trailing edge is of double skin (inner skin beaded) construction, and is assembled by spot-welding or bonding. The trailing edge is attached to the rudder rear beam by flush rivets. A trim tab is attached to the rudder at the rear beam, beginning where the trailing edge ends. The trim tab is of aluminum alloy skin and rib construction, and is assembled by both rivets and spot-welds or bonding. The tab is connected to the rudder by a piano hinge, and the tab actuator is mounted on the rear beam of the vertical stabilizer.

55.9. ELEVATOR. The elevator is an aluminum alloy structure consisting of front and rear beams, ribs, and unstiffened skin. The skin is attached by flush rivets. The trailing edge is of double skin (inner skin beaded) construction and is assembled by spot-welding or bonding. The trailing edge is attached to the elevator rear beam by flush rivets. The elevator is supported by five hinge points which tie to the horizontal stabilizer, and by a support bearing near the centerline of the airplane. Static balance is obtained by counterweights forward of the hingeline at the outboard tip and at the inboard end on the torque tube. The counterweights are of depleted uranium.

55.10 DETAIL STABILIZER COMPONENT MATERIAL AND IDENTIFICATION. For material and identification of detail stabilizer components, refer to figures 55-1 through 55-7 at the end of this section. For typical repairs, refer to Chapter 51.

WARNING

Refer to the precautions in section 51.9 before working on or near the uranium counterweights.



INDEX	COMPONENT	FIG. NO.
1	VERTICAL STABILIZER TIP	55-3
2	UPPER VERTICAL STABILIZER BOX SECTION	55-7
3	RUDDER AND TAB	55-5
4	TRAILING EDGE FAIRING	55-2
5	HINGE INSTALLATION	55-6
6	VERTICAL STABILIZER TRAILING EDGE	55-2
7	ELEVATOR	55-5
8	HORIZONTAL STABILIZER TIP	55-3
9	HORIZONTAL STABILIZER BOX SECTION	55-6
10	HORIZONTAL STABILIZER LEADING EDGE	55-4
11	HORIZONTAL STABILIZER FAIRING	55-3
12	VERTICAL STABILIZER FAIRING	55-3
13	LOWER VERTICAL STABILIZER LEADING EDGE	55-4
14	LOWER VERTICAL STABILIZER BOX SECTION	55-7
15	HORIZONTAL STABILIZER CENTER SECTION	55-6
16	UPPER VERTICAL STABILIZER LEADING EDGE	55-4

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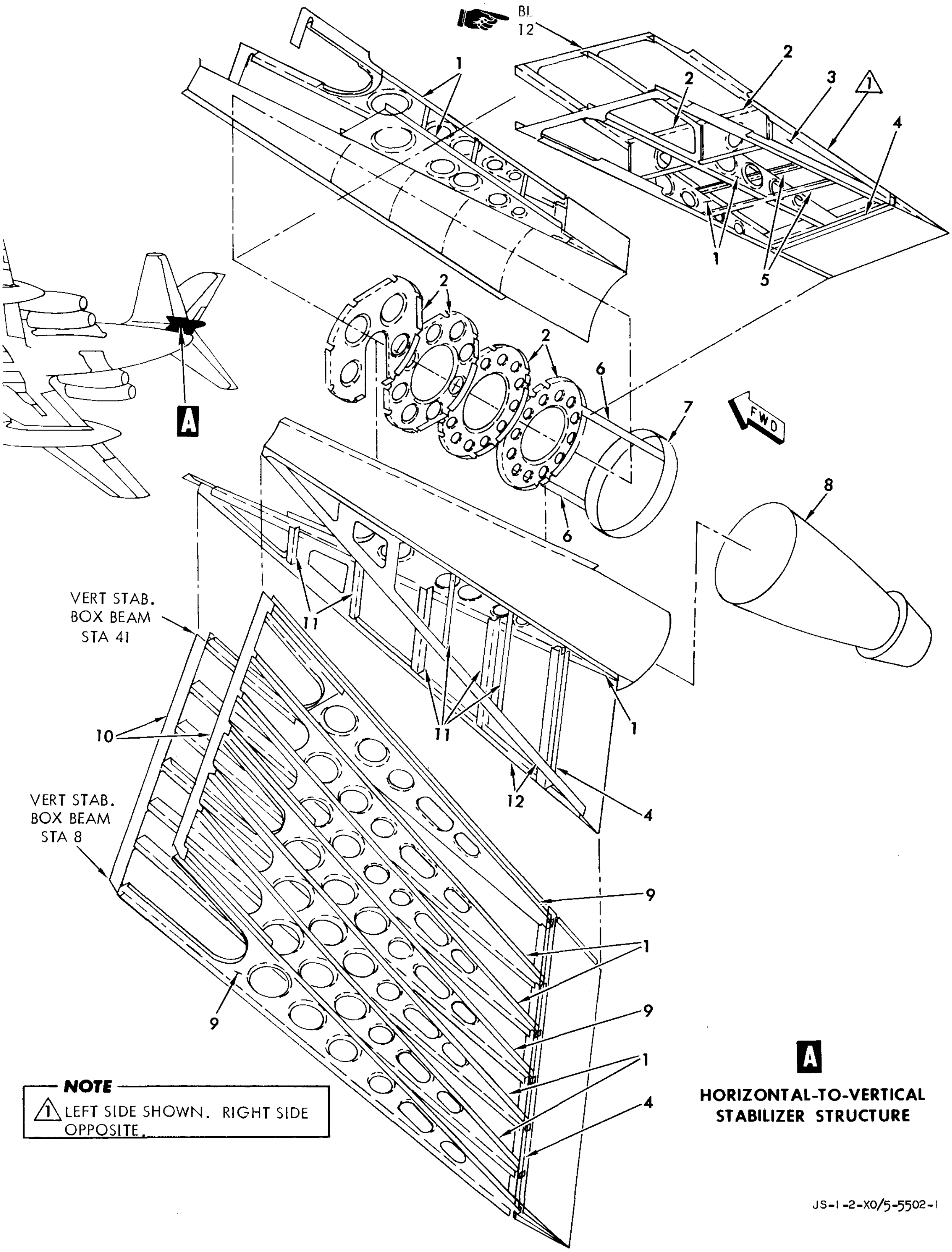
Figure 55-1. Empennage Group


BL 12
1
2
2
3
△1
4
1
5
A
2
2
6
7
FWD
8
6
VERT STAB. BOX BEAM STA 41
11
10
11
1
12
4
VERT STAB. BOX BEAM STA 8
9
1
9
1
4
9
A
HORIZONTAL-TO-VERTICAL STABILIZER STRUCTURE
JS-1-2-X0/5-5502-1

NOTE

△1 LEFT SIDE SHOWN. RIGHT SIDE OPPOSITE.

Figure 55-2. Lower Trailing Edge and Trailing Edge Tee Assembly (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	RIB	0.032	2024-T4 CLAD
2	FORMER	0.032	2024-T4 CLAD
3	RIB ASSEMBLY		
	CAP		2024-T4 EXTRUSION
	WEB	0.032	2024-T4 CLAD
4	CHANNEL	0.032	2024-T3 CLAD
5	STIFFENER	0.032	2024-T4 CLAD
6	STRAP	0.032	2024-T3 CLAD
7	DOUBLER	0.032	2024-T4 CLAD
8	NOZZLE FAIRING ASSEMBLY		
	TIP	0.080	6061-T4
	EJECTOR	0.063	6061-T6
9	RIB	0.040	2024-T4 CLAD
10	DOUBLER	0.063	7075-T6 CLAD
11	ANGLE	0.032	2024-T4 CLAD
12	DOUBLER	0.040	2024-T4 CLAD

JS-2-2-5502-2

Figure 55-2. Lower Trailing Edge and Trailing Edge Tee Assembly (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	FORMER	0.040	2024-T4 CLAD
2	ZEE	0.032	2024-T4 CLAD
3	FORMER	0.051	2024-T4 CLAD
4	ZEE	0.040	2024-T4 CLAD
5	SKIN	0.032	2024-T4 CLAD
6	DOOR	0.040	2024-T4 CLAD
7	FILLET	0.040	2024-T4 CLAD
8	FILLET	0.063	2024-T3 CLAD
9	DOOR	0.025	2024-T4 CLAD
10	SKIN	0.025	2024-T3 CLAD
11	SKIN	0.025	2024-T4 CLAD
12	TRAILING EDGE BOND ASSEMBLY		
	SKIN	0.016	2024-T4 CLAD
	INNER SKIN	0.016	2024-T4 CLAD
13	SHIELD	0.020	302 STEEL ANNEALED
14	SKIN	0.032	2024-T3 CLAD
15	DOOR	0.025	2024-T3 CLAD
16	SPLICE ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	DOUBLER	0.040	2024-T4 CLAD
17	DOOR	0.032	2024-T4 CLAD
18	TIP, VERTICAL STABILIZER	0.040	6061-T6
19	ANTICOLLISION LIGHT		
20	TIP, HORIZONTAL STABILIZER	0.032	6061-T4

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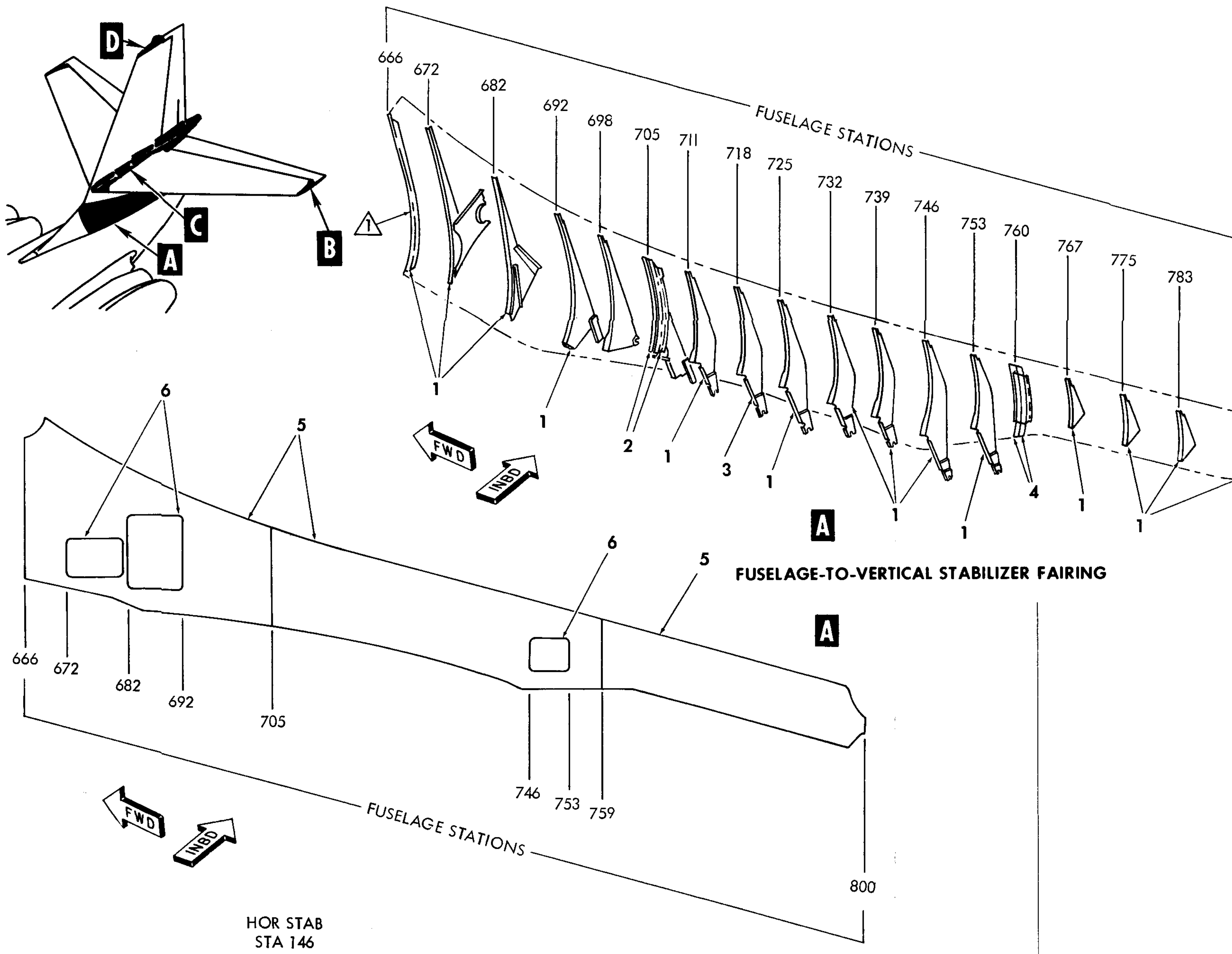
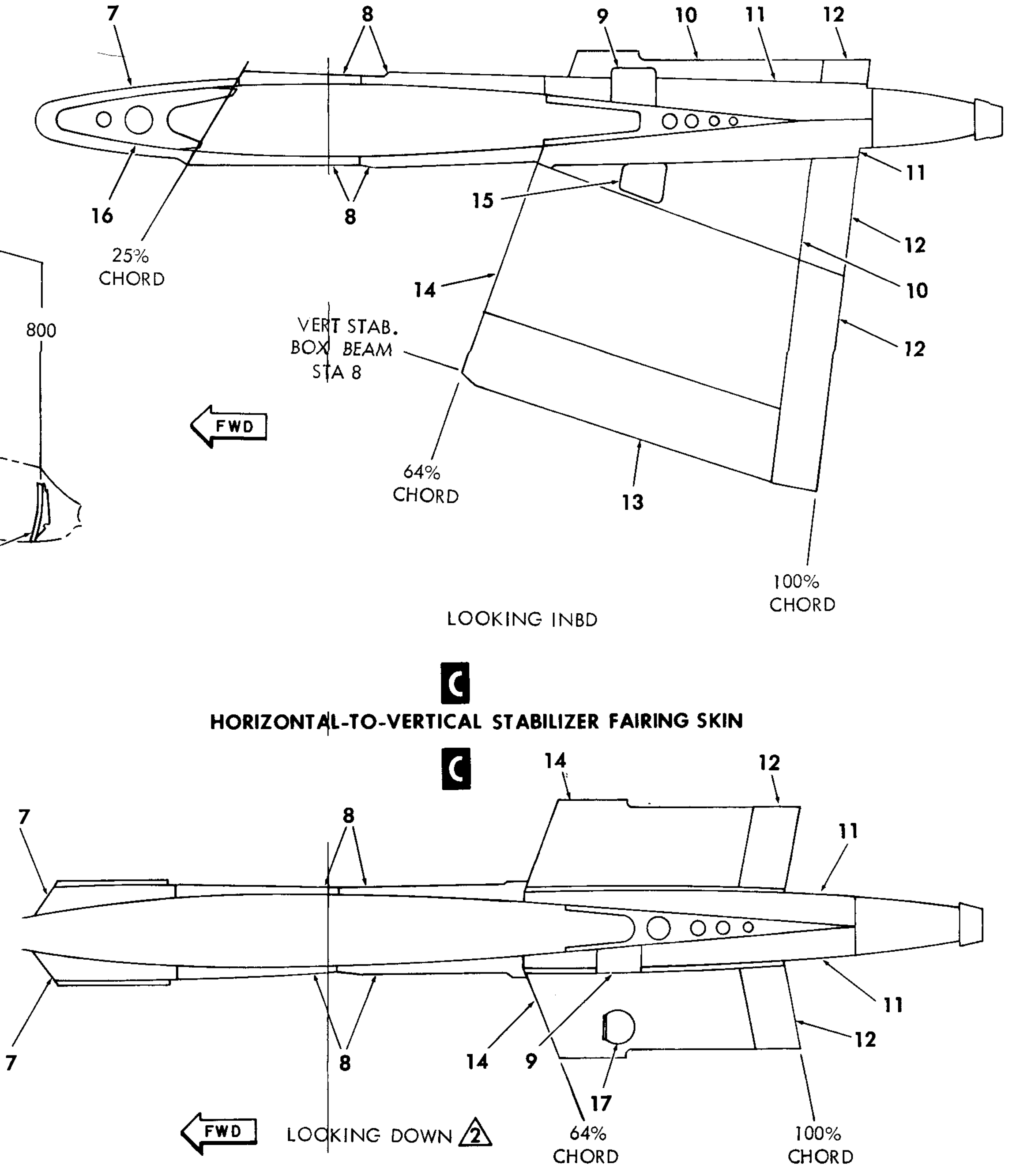
Figure 55-3. Empennage Tips and Fairing Skin (Sheet 1 of 2)



A

FUSELAGE-TO-VERTICAL STABILIZER FAIRING

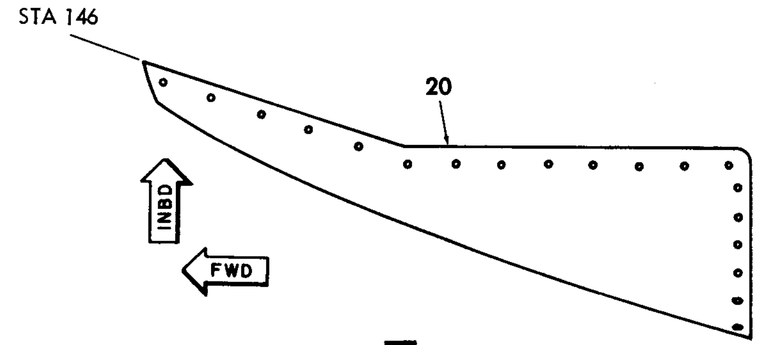
A



C

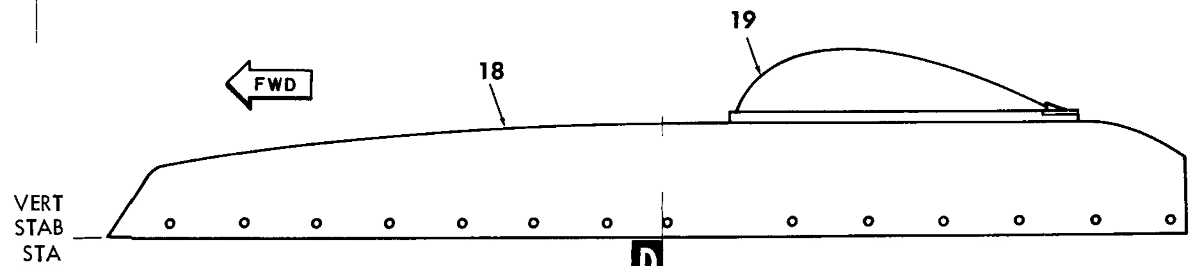
HORIZONTAL-TO-VERTICAL STABILIZER FAIRING SKIN

C



B

HORIZONTAL STABILIZER TIP



D

VERTICAL STABILIZER TIP

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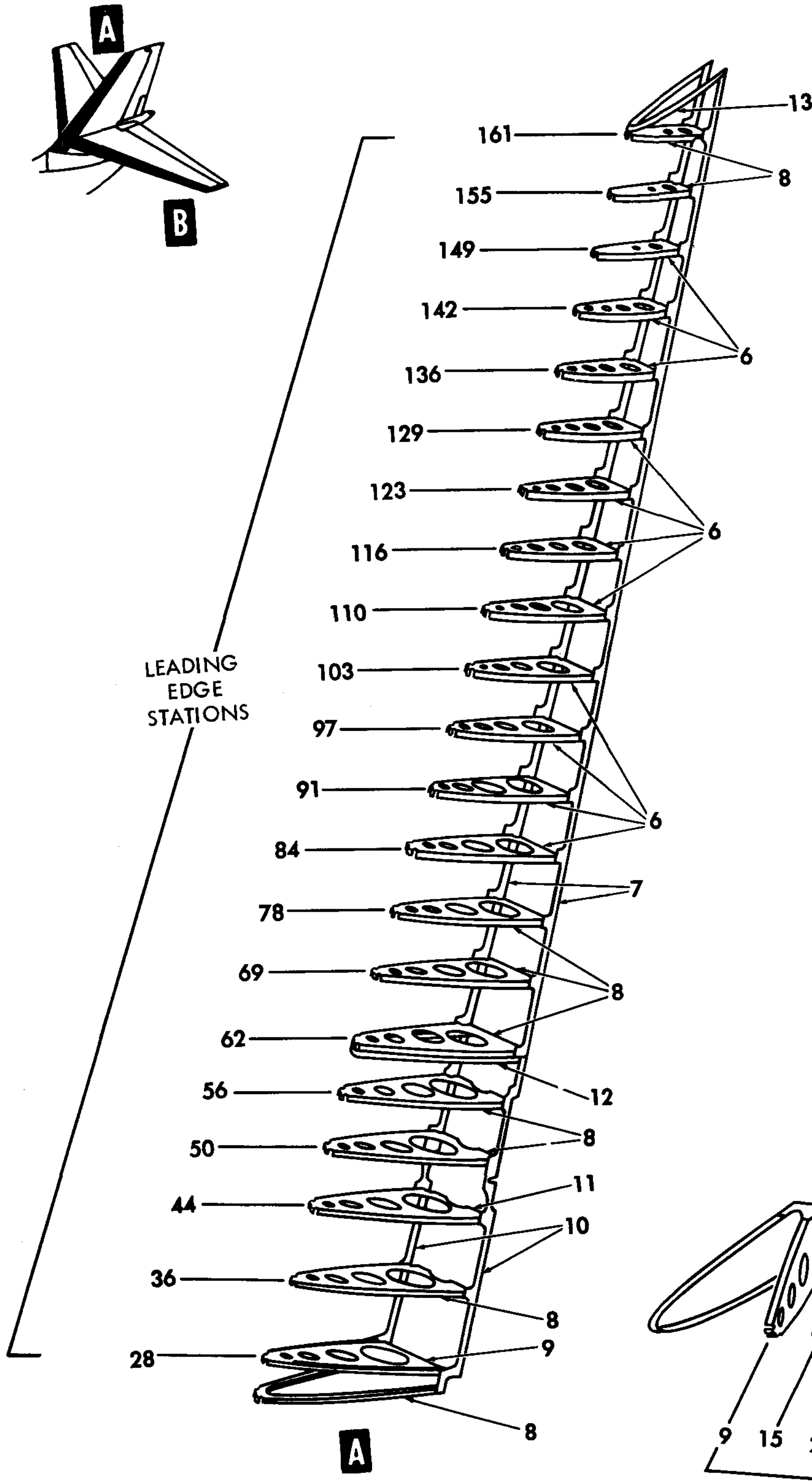
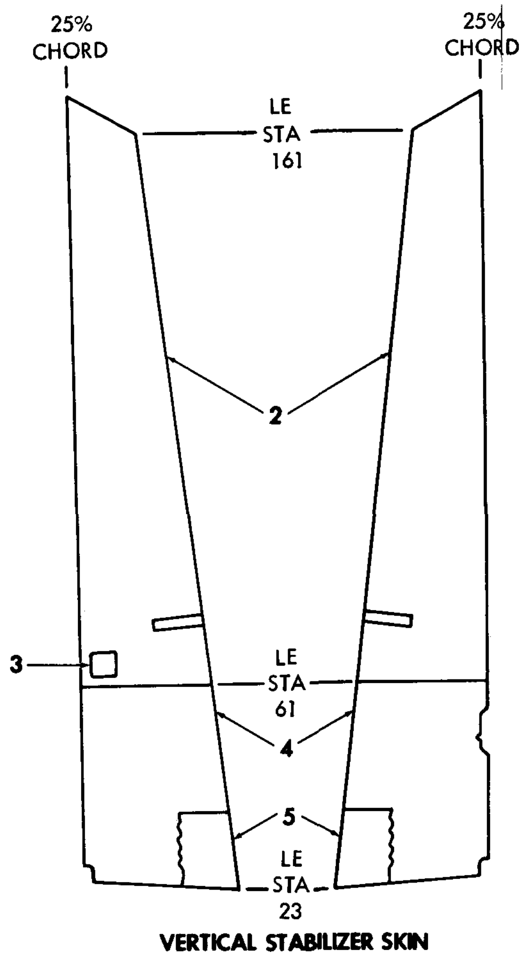
Figure 55-3. Empennage Tips and Fairing Skin (Sheet 2 of 2)

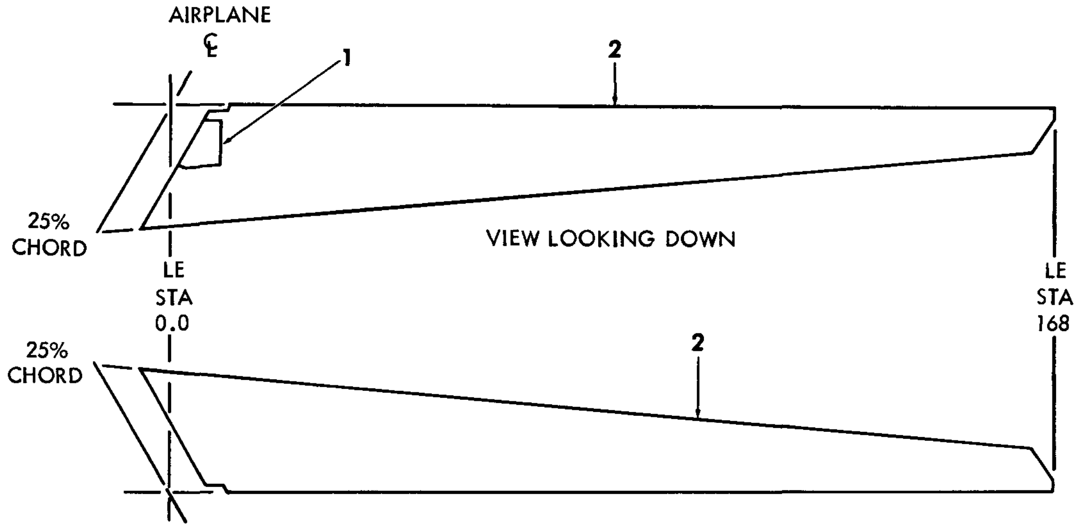
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	DOOR	0.040	7075-T6 CLAD
2	SKIN	0.032	7075-T6 CLAD
3	DOOR	0.040	2024-T3 CLAD
4	SKIN	0.032	2024-T3 CLAD
5	SHIELD	0.020	302 STEEL, ANNEALED
6	RIB	0.032	2024-T4 CLAD
7	DOUBLER	0.063	7075-T6 CLAD
8	RIB	0.040	2024-T4 CLAD
9	RIB ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	TEE		2024-T4 EXTRUSION
10	DOUBLER	0.063	2024-T3 CLAD
11	RIB	0.040	2024-T3 CLAD
12	DOUBLER	0.063	2024-T4 CLAD
13	SPLICE	0.040	2024-T3 CLAD





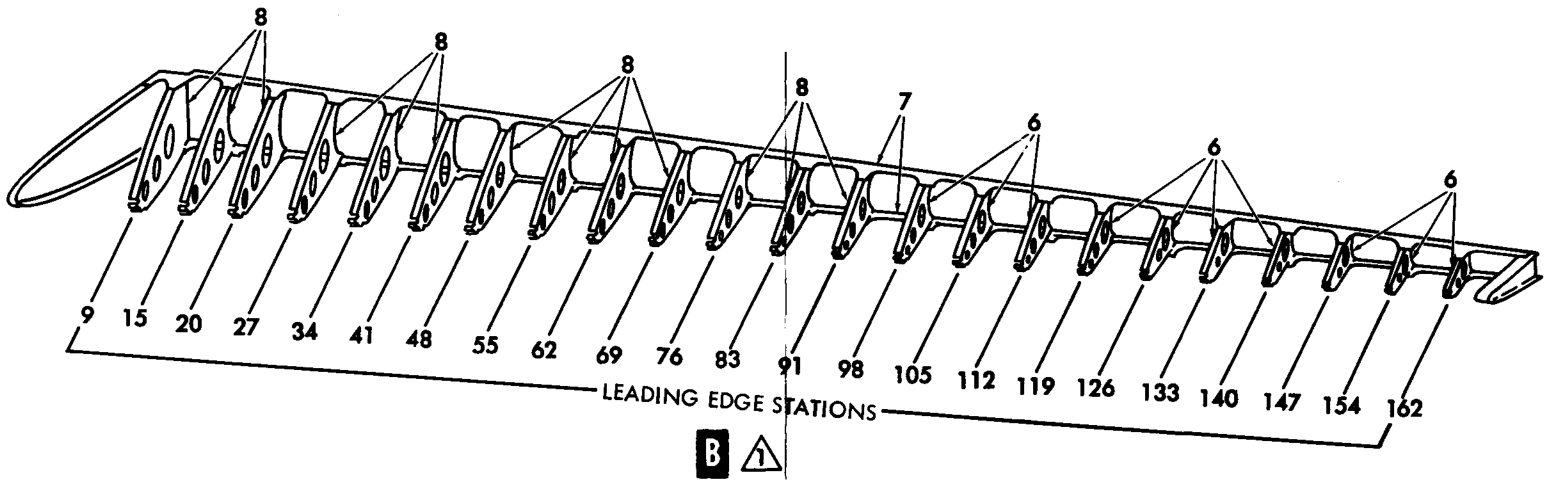
VERTICAL STABILIZER SKIN

A



HORIZONTAL STABILIZER SKIN

B ⚠1



NOTE

⚠1 LEFT SIDE SHOWN

RIGHT SIDE OPPOSITE

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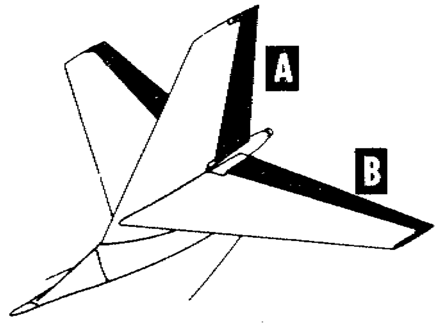
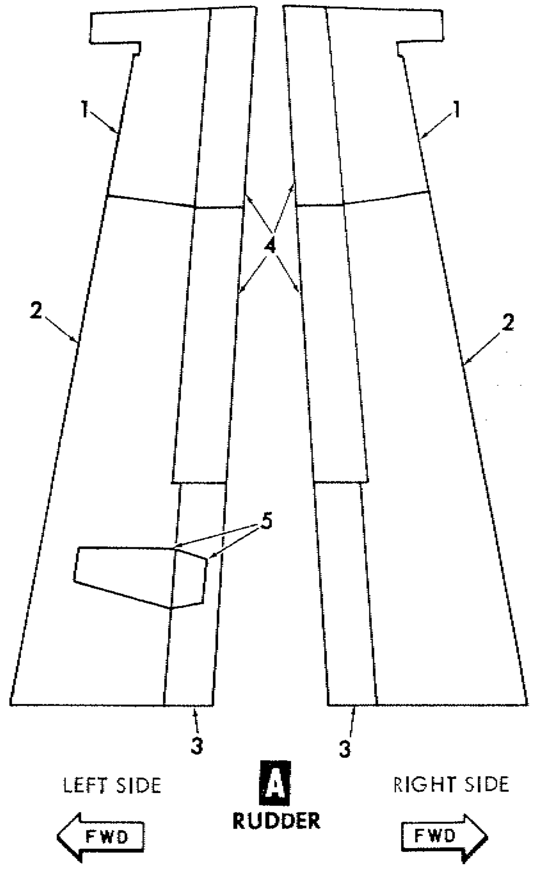
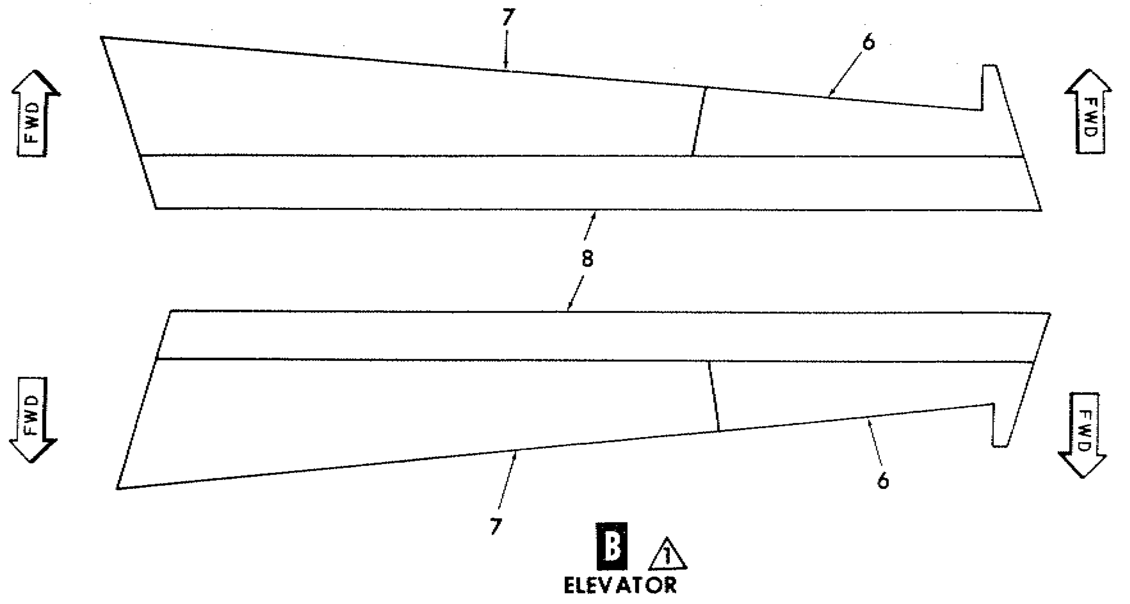
Figure 55-4. Leading Edge Skin and Structure

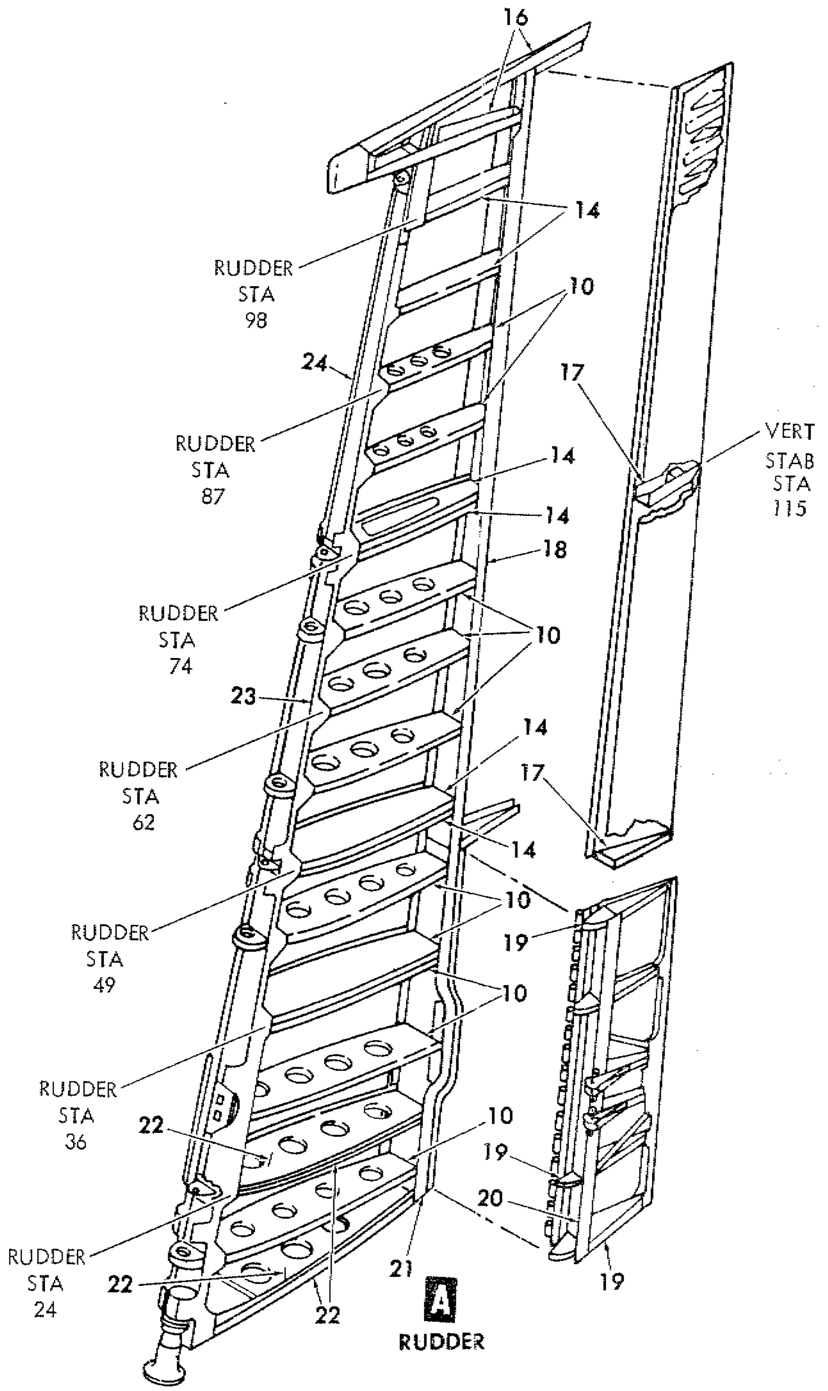




NOTE

⚠ LEFT ELEVATOR SHOWN
RIGHT ELEVATOR OPPOSITE





INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	SKIN	0.040	7075-T6 CLAD
2	SKIN	0.025	2024-T4 CLAD
3	SKIN	0.020	2024-T4 CLAD
4	SKIN ASSEMBLY		
	OUTER SKIN	0.016	2024-T4 CLAD
	INNER SKIN	0.016	2024-T4 CLAD
5	COVER (LEFT SIDE ONLY)	0.032	2024-T4 CLAD
6	SKIN	0.050	7075-T6 CLAD
7	SKIN	0.025	7075-T6 CLAD
8	SKIN ASSEMBLY		
	INNER SKIN	0.012	2024-T4 CLAD
	OUTER SKIN	0.012	2024-T4 CLAD
9	RIB	0.063	7075-T6 CLAD
10	RIB	0.032	2024-T4 CLAD
11	CHANNEL	0.032	7075-T6 CLAD
12	DOUBLER	0.040	2024-T3 CLAD
13	CHANNEL	0.050	7075-T6 CLAD
14	RIB	0.040	2024-T4 CLAD
15	SPAR	0.040	2024-T4 CLAD
16	RIB		2014-T6 FORGING
17	RIB	0.025	2024-T4 CLAD
18	REAR BEAM	0.040	2024-T4 CLAD
19	TAB RIB	0.032	2024-T4 CLAD
20	BEAM	0.032	2024-T4 CLAD
21	DOUBLER	0.032	2024-T4 CLAD
22	RIB	0.050	2024-T4 CLAD
23	DOUBLER	0.040	7075-T6 CLAD
24	FRONT BEAM	0.040	7075-T6 CLAD

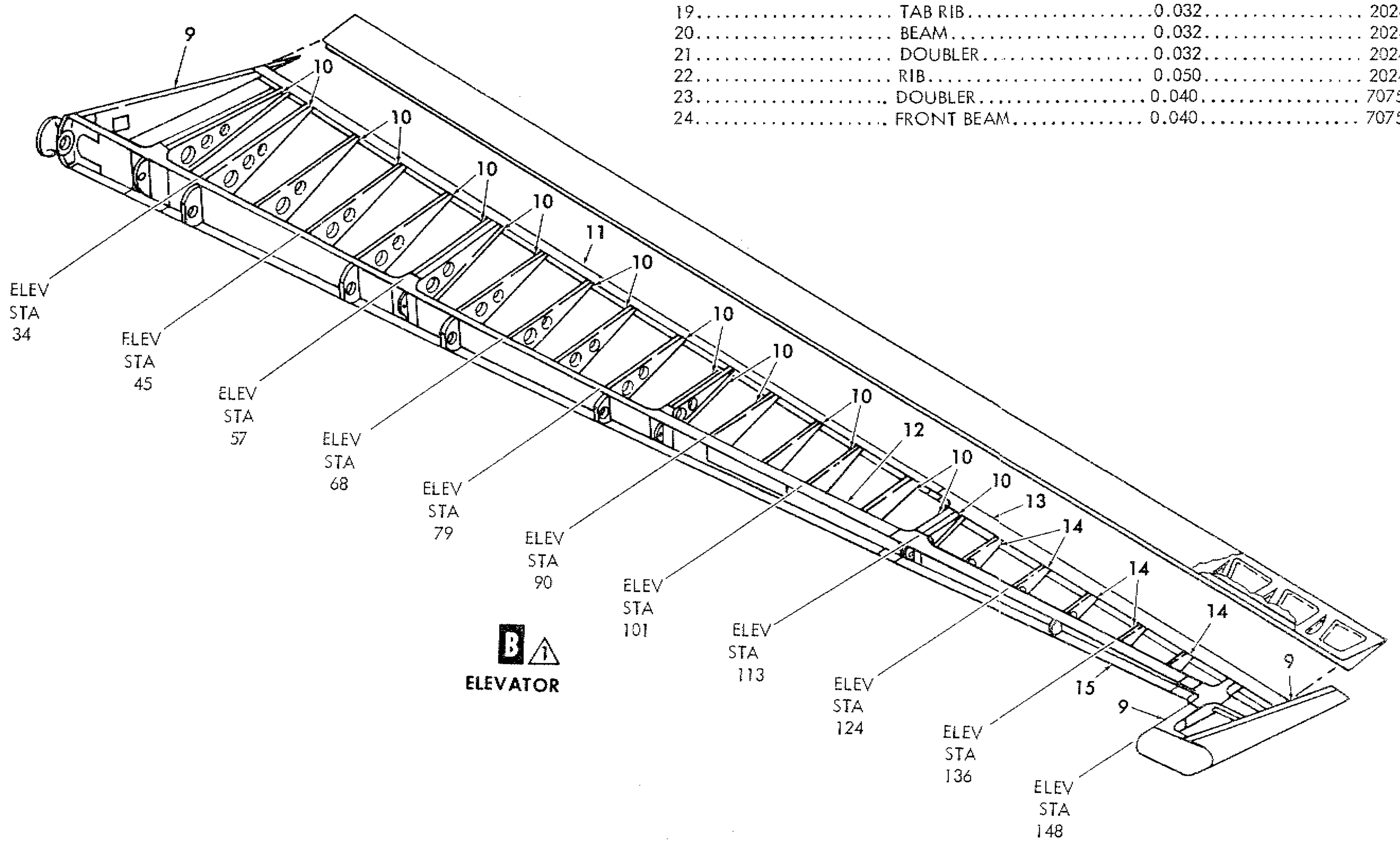
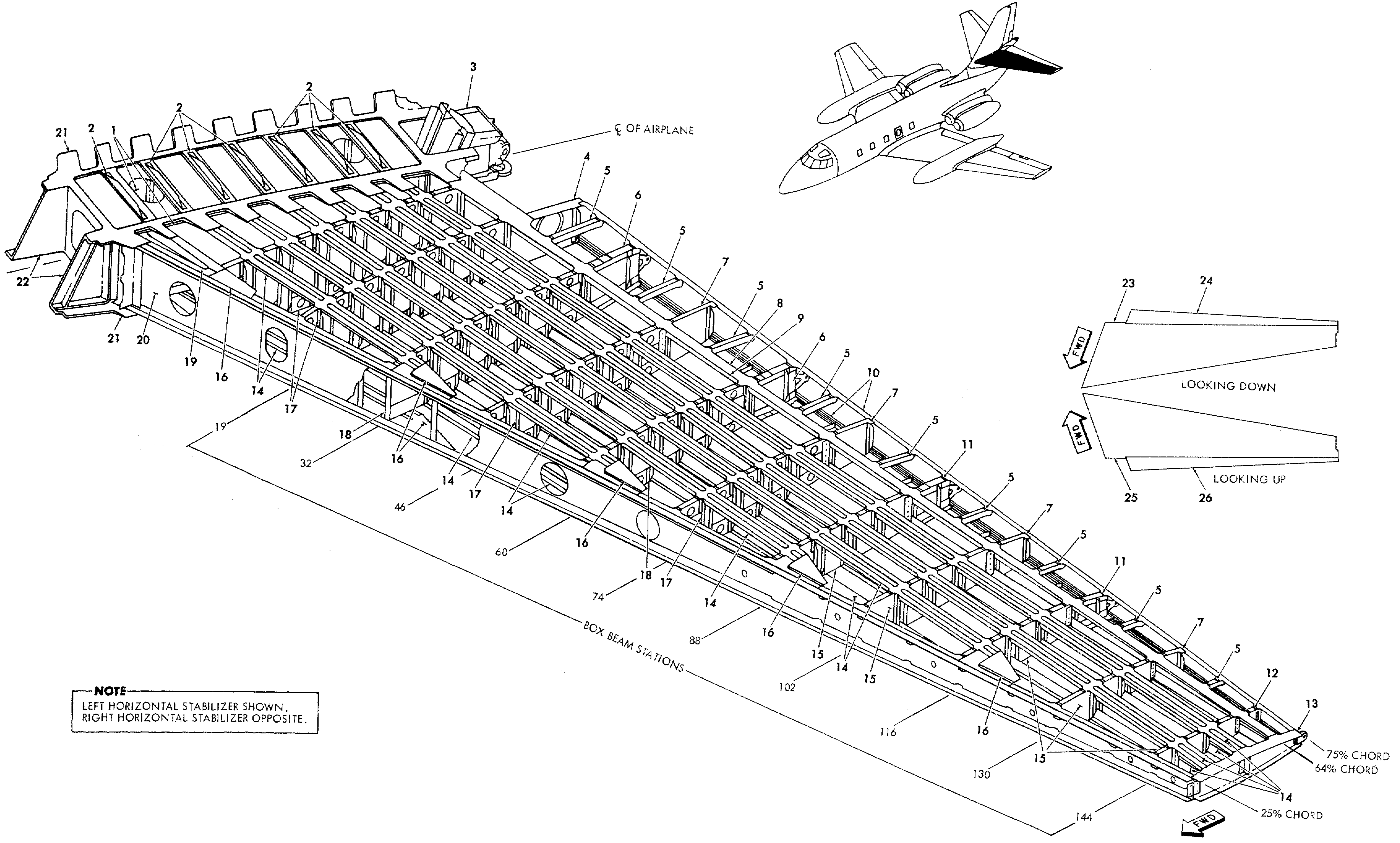


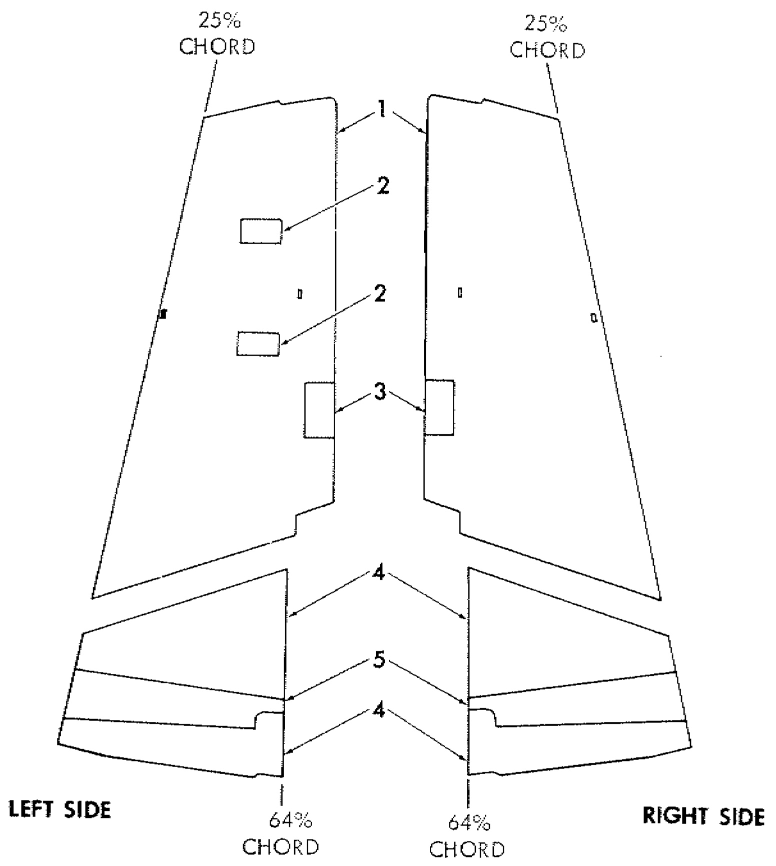
Figure 55-5. Rudder and Elevator Skin and Structure



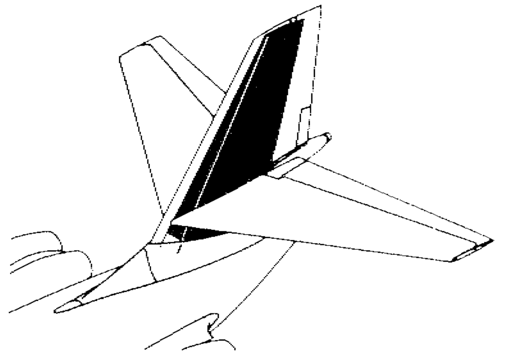
NOTE
LEFT HORIZONTAL STABILIZER SHOWN.
RIGHT HORIZONTAL STABILIZER OPPOSITE.

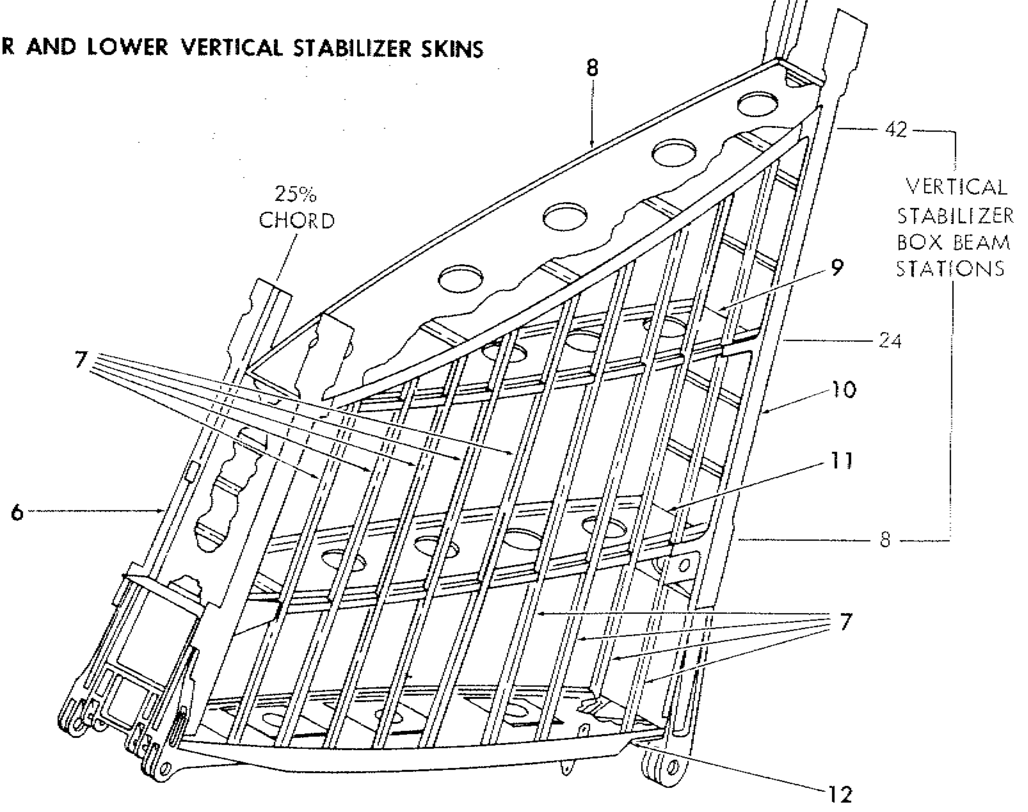
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1.	RIB ASSEMBLY		
	CAP, UPPER & LOWER		7075-T6 EXTRUSION
	WEB, FRONT	0.050	7075-T6 CLAD
	WEB, AFT	0.090	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
2.	STIFFENER		7075-T6 EXTRUSION
3.	ELEVATOR HINGE INSTALLATION		
4.	SPLICE RIB	0.050	2024-T4 EXTRUSION
5.	STIFFENER		2024-T4 EXTRUSION
6.	RIB ASSEMBLY		
	CAP, UPPER AND LOWER	0.065	2024-T4 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER	0.070	2024-T4 EXTRUSION
	STIFFENER	0.040	2024-T4 CLAD
	SPACER	0.063	2024-T3 CLAD
	FITTING		7075-T6 EXTRUSION
7.	RIB ASSEMBLY		
	CAP, UPPER AND LOWER	0.050	2024-T4 EXTRUSION
	WEB	0.040	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
8.	DOUBLER	0.063	7075-T6 CLAD
9.	REAR BEAM ASSEMBLY		
	CAP, UPPER & LOWER		7075-T6 EXTRUSION
	WEB	0.050	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	ANGLE	0.050	2024-T4 CLAD
10.	TEE		7075-T6 EXTRUSION
11.	RIB ASSEMBLY		
	CAP, UPPER AND LOWER	0.065	2024-T4 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER	0.070	2024-T4 EXTRUSION
	SPACER	0.063	2024-T3 CLAD
	FITTING		7075-T6 EXTRUSION
12.	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	2024-T4 CLAD
13.	RIB		2014-T6 AL. FORGING
14.	STRINGER		7075-T6 EXTRUSION
15.	RIB ASSEMBLY		
	RIB	0.071	2024-T4 CLAD
	STIFFENER	0.062	2024-T4 EXTRUSION
16.	GUSSET	0.040	7075-T6 CLAD
17.	RIB ASSEMBLY		
	CHANNEL	0.071	2024-T4 CLAD
	STIFFENER	0.062	2024-T4 EXTRUSION
18.	RIB ASSEMBLY		
	CHANNEL	0.071	2024-T4 CLAD
	WEB	0.032	2024-T3 CLAD
	FILLER	0.040	2024-T3 CLAD
	STIFFENER	0.062	2024-T4 EXTRUSION
19.	STIFFENER		2024-T4 EXTRUSION
20.	FRONT BEAM		
	CAP, UPPER & LOWER		7075-T6 EXTRUSION
	WEB	0.063	7075-T6 EXTRUSION
	RIB ATTACHMENT TEE		7075-T6 EXTRUSION
	RIB ATTACHMENT ANGLE		7075-T6 EXTRUSION
	LEADING EDGE ATTACHMENT ANGLE		7075-T6 EXTRUSION
21.	PLATE (CHEMICALLY MILLED)	0.312	7075-T6 BARE
22.	FITTING		7075-T6
23.	SKIN (CHEMICALLY MILLED)	0.050	7075-T6 CLAD
24.	SKIN	0.032	7075-T6 CLAD
25.	SKIN	0.050	7075-T6 CLAD
26.	SKIN (CHEMICALLY MILLED)	0.032	2024-T3 CLAD

Figure 55-6. Horizontal Stabilizer Box Section Skin and Structure

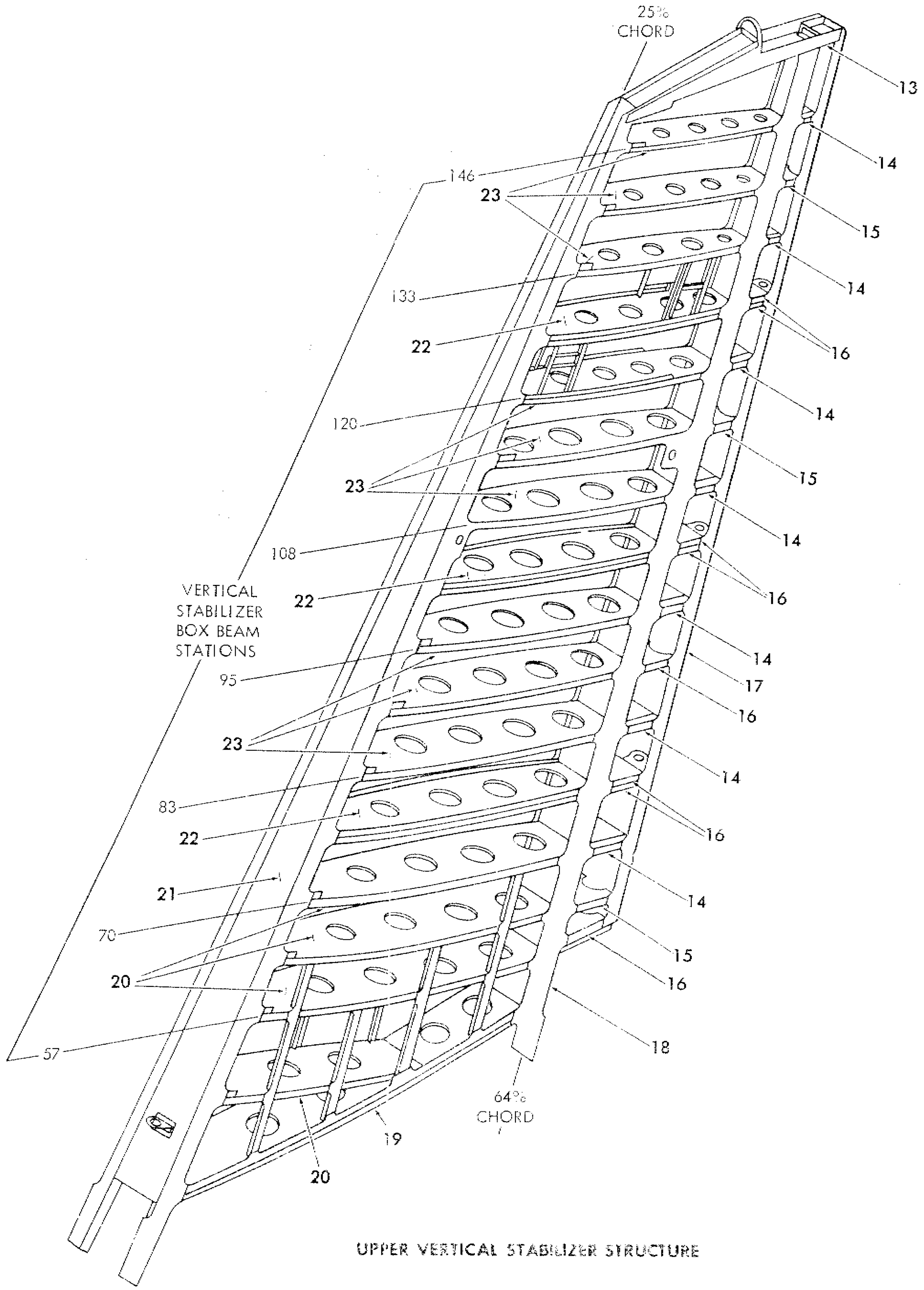


UPPER AND LOWER VERTICAL STABILIZER SKINS





LOWER VERTICAL STABILIZER STRUCTURE



UPPER VERTICAL STABILIZER STRUCTURE

Figure 55-7. Vertical Stabilizer Box Section Skin and Structure

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	SKIN	0.025	2024-T4 CLAD
2	DOOR	0.040	2024-T4 CLAD
3	DOOR	0.025	2024-T3 CLAD
4	SKIN	0.063	7075-T6 CLAD
5	SHIELD	0.020	302 STEEL ANNEALED
6	FRONT BEAM ASSEMBLY		
	CAP		2024-T4 EXTRUSION
	WEB	0.050	7075-T6 CLAD
	ANGLE	0.040	2024-T4 CLAD
	STIFFENER		7075-T6 EXTRUSION
	DOUBLER	0.160	7075-T6 CLAD
	ACTUATOR FITTING		8630 STEEL COND E-Z
	SCISSORS FITTING		7075-T6 FORGING
7	STRINGER	0.032	2024-T4 CLAD
8	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.050	2024-T3 CLAD
	STIFFENER		7075-T6 EXTRUSION
9	RIB ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	STIFFENER	0.050	2024-T4 CLAD
10	REAR BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	DOUBLER	0.250	7075-T6 BARE
	WEB	0.050	7075-T6 BARE
	STIFFENER		7075-T6 EXTRUSION
	PIVOT FITTING		2014-T6 AL ALLOY
11	RIB ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	ANGLE		7075-T6 EXTRUSION
	STIFFENER	0.063	2024-T4 CLAD
12	RIB ASSEMBLY		
	WEB	0.050	2024-T3 CLAD
	CAP		7075-T6 EXTRUSION
	STIFFENER		7075-T6 EXTRUSION
13	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.040	7075-T6 CLAD
	DOUBLER	0.050	7075-T6 CLAD
14	STIFFENER		7075-T6 EXTRUSION
15	RIB	0.032	2024-T4 CLAD
16	RIB	0.040	2024-T4 CLAD
17	TEE		2024-T4 EXTRUSION
18	REAR BEAM		
	CAP		7075-T6 EXTRUSION
	WEB, UPPER	0.063	7075-T6 CLAD
	WEB, LOWER	0.040	7075-T6 CLAD
19	RIB ASSEMBLY		
	CAP		2024-T4 EXTRUSION
	WEB	0.050	2024-T4 CLAD
	STIFFENER		2024-T4 EXTRUSION
20	RIB	0.040	2024-T4 CLAD
21	FRONT BEAM		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
22	RIB ASSEMBLY		
	RIB	0.040	2024-T4 CLAD
	FORMER, CAP	0.040	2024-T4 CLAD
23	RIB	0.032	2024-T4 CLAD

* REFER TO SECTION VIII FOR NEGLIGIBLE DAMAGE LIMITS AND TYPICAL REPAIRS

CHAPTER 56
WINDOWS

Contents Page

DESCRIPTION AND OPERATION 56-1
- Windshields 56-1
- Skylights 56-1
- Cabin Windows 56-1
- Observation and Inspection Windows 56-2

INSPECTION 56-2
- Windshields 56-2
- Delamination 56-2
- Visibility 56-2
- Limits for Delamination 56-2
- Arcing 56-2
- Scratches 56-3
- Cracks and Vents 56-3
- Chips 56-3
- Conchoidal Edge Chips 56-3
- Skylight Panel Assembly 56-3
- Cabin Window Assemblies 56-3
- Glass Vinyl Seal 56-4
- Steel Frame Windshields 56-4

REMOVAL AND REPLACEMENT 56-5
- Removal of Windows 56-5
- Window Sealing 56-5
- Window Replacement 56-5
- Replacement of Clear Vision Windshield Molded Seal 56-5
- Replacement of Thermostat Aft Side Windshield 56-6

FIGURES
- 56-1 Delamination Limits for Windshields 56-7
- 56-2 Types of Windshields 56-8
- 56-3 Sealing of Windshields and Windows 56-9
- 56-4 Clear Vision Windshield Screw and Bolt Length Diagram 56-10
- 56-5 Center Windshield Screw and Bolt Length Diagram 56-10
- 56-6 Aft Side Windshield Screw and Bolt Length Diagram 56-11
- 56-7 Forward Side Windshield Screw and Bolt Length Diagram 56-11
- 56-8 Inspection and Repair of Windshield Sealant 56-12
- 56-9 Windshield Installation Instructions 56-13
- 56-10 Skylight and Cabin Window Installation Instructions 56-16
- 56-11 Cabin Window Rabbet Visual Inspection 56-18

56.1 DESCRIPTION AND OPERATION.

56.1.1 WINDSHIELDS. The windshields consist of seven individual panels extending around the front of the pilot's compartment. The second panel from the rear on each side of the compartment is a clear vision panel and can be unlatched and slid aft, clear of the opening. All windshields are electrically anti-iced and are provided with a de-fogging system. The electrical heating also keeps the vinyl layers on the windshields in a flexible condition, and improves the shock load absorbency. This also improves the bird strike resistance quality of the windshields.

Note

For electrical details of the windshield panels, refer to Section 30.2.

56.1.2 SKYLIGHTS. Two windows at the top of the pilot's compartment provide vertical vision. No anti-icing or de-fogging is provided for these panels, but they are provided with a thin, replaceable inner liner for protection and/or glare resistance.

56.1.3 CABIN WINDOWS. Five windows flank the cabin on each side. The fourth window aft on each side of the cabin is contained in an emergency exit. Beneath each of the cabin windows, behind the interior trim, is a small desiccant-filled vial connected to the window by a small tube. The desiccant absorbs the moisture from the air in the open space between the window laminations and prevents window misting. When the windows begin to mist, the desiccant should be replaced. (Refer to Section 30.6.)

56.1.4 OBSERVATION AND INSPECTION WINDOWS. There is an observation window on the aft pressure bulkhead at approximately the center of the aircraft and approximately 52 inches above the floor, to enable the crew to view the aft fuselage compartment. There are also two observation windows, one in each main wheel well, in the walkway area approximately nine inches above the floor, to enable the crew to observe whether the main landing gear is properly locked. These windows are provided on aircraft 5035 and up, and on all aircraft that have incorporated Service Bulletin 329-66. An inspection window exists in the tail cone assembly on aircraft 5046 and up, and all previous aircraft that have incorporated Service Bulletin 329-86. This window is for inspection of the locking mechanism of the drag chute.

56.2 INSPECTION.

56.2.1 WINDSHIELDS. The windshield panels in the pilot's compartment must be free of defects which seriously weaken them or that impair vision.

Note

> Occasionally, during close inspection of windshield panels, small dimples may be noticeable close to the edge of the panel, and sometimes partially obscured by the frame. These small dimples may generally be found on the edge of the panel which contains the vendor identification data, and may be found in quantities of two or four. The dimples will generally be of the same diameter but may be of varying depths. Presence of these marks in the glass surface does not impair the quality of the glass and should not be cause for concern or rejection. These marks are caused by tongs used to hold the glass during the heat tempering process. The glass is heated and becomes semi-molten prior to the air cooling/quenching process. It is impossible to hold temperatures identical throughout the panel during this process, which accounts for difference in depth between dimples.

Small defects such as the following are permitted:

a. Small transparent glass chips (cullet) adhering to the interior plastic (vinyl) sheet.

b. Very slight traces of crayon.

c. Irregularities in the plastic sheets which give slight change in light reflection.

d. Small bits of fabric embedded or entrapped in the laminations.

e. Small irregularities (glass sleeks) on the glass surfaces, which show as a slight change in light reflection.

56.2.1.1 DELAMINATION. Delamination of window and windshield panels is a visible, physical separation of the glass and vinyl plies in the panel sandwich. A delaminated panel is identified by the following characteristics:

a. The delaminated area is smooth in appearance in both direct and indirect light.

b. Visual distortion through the delaminated area is slight except along the edges.

c. Delamination generally starts near the edge of the glass and progresses inward and along the edge of the panel.

d. The delaminated area may be difficult to detect unless the window is viewed at an angle.

Note

> Do not mistake the normal parting medium between the outer glass ply and the plastic inner layer for a delamination. This normally separated area will be noticed around the entire periphery of the window.

56.2.1.2 VISIBILITY. Delamination does not seriously affect the strength of the panel, but it may affect visibility, or damage the electrically conductive coating. Electrical anti-icing and pressurized flight can continue safely regardless of nature, extent, or configuration of the delamination, provided arcing does not occur. However, the panel should be removed at the earliest convenience or prior to unsafe impairment of vision, if the delamination limits listed below are exceeded.

56.2.1.3 LIMITS FOR DELAMINATION. (See figure 56-1 for explanation of dimensions "L" and "W.")

a. Edge delamination is permitted when "L" is less than 1/2 inch.

b. Edge delamination up to 1.0 inch in the "L" direction is permitted provided "W" is at least twice as much as "L."

c. Corner delamination is permitted when "L" is less than 1-1/2 inches and the corner angle is 60 degrees or greater.

d. Corner delamination is permitted when "L" is less than 2 inches and the corner angle is less than 60 degrees.

56.2.1.4 ARCING. If any electrically heated panel delaminates, it may cause arcing or loss of electrical continuity. If such a condition is evident, the windshield anti-icing system should be turned off as soon as possible.

Note

> Heated windshield limitations in the Airplane Flight Manual must be observed when operating without windshield anti-icing.

Arcing of electrically heated panels may or may not occur before delamination or cracking. However, this condition is unsatisfactory and necessitates replacement of the panel.

CAUTION

Delaminated panels should not be confused with chipped panels. Chipped panels are seriously weakened, and the panel must be replaced if the chipping exceeds the limits noted in paragraphs 56.2.1.7 and 56.2.1.8.

56.2.1.5 SCRATCHES. Panels that exceed the following limits should be replaced:

a. Scratches containing small cracks are not permissible.

b. Scratches should not exceed 0.020 inch in depth.

c. The length of a scratch should not exceed one third of the width of the glass, measured parallel to the scratch, and should not extend to the edge of the glass.

56.2.1.6 CRACKS AND VENTS. Panels that exceed the following limits should be replaced.

a. Cracks extending completely through a glass ply are not acceptable.

b. Vents are internal folds extending from the underside, but not to, the glass surface. Small occasional vents are permissible only in the window inner ply near the edge, provided they are not directed toward the center of the panel and do not impair vision.

56.2.1.7 CHIPS. Panels that exceed the following limits should be replaced:

a. Chips containing small cracks are not permissible.

b. Circular chips should not exceed 1.0 inch in diameter, should not exceed one fourth of the glass thickness in depth, and should not extend to the edge of the glass.

56.2.1.8 CONCHOIDAL EDGE CHIPS. Panels that exceed the following limits should be replaced:

a. These chips should not exceed 1/2 inch in length, should not extend more than 1/4 inch from the glass edge, and should not exceed one fourth of the glass thickness in depth.

b. Chips longer than 0.12 inch should not be closer together than 3 inches. There is no restriction for chips less than 0.12 inch long.

56.2.2 SKYLIGHT PANEL ASSEMBLY. The skylight panel assembly consists of two layers of an acrylic (optical grade) and a vinyl interlayer as shown in figure 56-3, detail B.

a. A crack in the outer layer (0.250 thick) requires operation at reduced pressurization - 4-1/2 to 5 PSI until the panel can be replaced.

b. Cracks identified as being in the vinyl interlayer present an appearance problem but are not a problem as far as panel integrity is concerned. There is no need to reduce pressurization during flying operations.

c. A crack in the inner layer (0.500 thick) is cause for immediate panel rejection.

56.2.3 CABIN WINDOW ASSEMBLIES.

a. Inspect outer panel for cracks in the radius of the rabbet cut as indicated in figure 56-11. Windows with cracks found in this area must be replaced.

56.2.3.1 These limits apply for the JF442 cabin window panel used on JetStar production units 5001 thru 5085:

CAUTION

Do not combine allowable scratch and allowable crack damage limitations on the same outer layer. Do not combine allowable damage limitations on outer layer with crack damage on the center (middle) layer on the same window.

a. Allowable scratch damage in the outer layer. Scratches in the outer layer up to a depth of 0.020 inch that do not terminate in a fastener hole are structurally acceptable and do not require cabin pressure restrictions. However, any scratched outer layer should be monitored at the 100-hour inspection to assure that cracks do not develop.

Outer layer scratches ranging in depth from 0.020 inch to 0.030 inch require window replacement. However, full cabin pressure is allowed for up to 100 pressure cycles prior to replacement provided the scratches do not terminate in a fastener hole.

Outer layer scratches ranging in depth from 0.030 inch to 0.050 inch or scratches that terminate in a fastener hole require window replacement. However, up to 100 flights with cabin differential pressure not exceeding 6.0 psi are allowed until the window is replaced.

b. Allowable cracks in outer layer. Cracks emanating from attachment/fastener holes should be monitored for growth after pressurized flights. The window must be replaced if total crack length is 0.38 inch or if crack has extended through the window thickness to the outer surface. Cracks are not allowed for adjacent holes or for more than 25 percent of the holes along any side of the window. Up to six (6) radial cracks at the bottom of the countersunk holes not exceeding 1/8 inch in length and not in more than four (4) holes along any side are acceptable.

c. Cabin pressure restrictions for cracked outer layer. Full cabin pressure is allowed for cracks within the above limits.

d. Cracks in center (middle) layer. Any crack in the center layer requires replacement of the cabin window.

e. Cabin pressure restrictions for cracked center layer. For a flight with a center layer containing one crack up to 0.75 inch in length, restrict cabin differential pressure to 4.5 psi.

f. The inner layer is a non-structural insulation panel to prevent the inner surface of the cabin window from becoming cold at altitude. A crack in this layer does not reduce the structural strength of the cabin window and creates no structural reason for replacement.

56.2.3.2 These limits apply to the JF764 cabin window used on JetStar production units 5086 and up and earlier aircraft with the JF764 cabin window installed as a preferred spare.

CAUTION

Do not combine inner and outer layer allowable scratch limits or allowable crack damage for inner layer on same window assembly.

a. Allowable scratch damage in the inner and outer layers. Scratches in the outer or inner layers up to a depth of 0.020 inch that do not terminate at the machine step at the panel edge (outer layer) or in a fastener hole (inner layer) are structurally acceptable and do not require cabin pressure restrictions. However, any scratched panel should be monitored at the 100-hour inspection to assure that cracks do not develop.

Outer or inner layer scratches ranging in depth from 0.020 inch to 0.030 inch require window replacement. However, full cabin pressure is allowed for up to 100 pressure cycles prior to replacement provided the scratches do not terminate at the machine step at the panel edge (outer layer) or in a fastener hole (inner layer).

Outer or inner layer scratches ranging in depth from 0.030 inch to 0.050 inch or scratches that terminate at the machine step at the panel edge (outer layer) or in a fastener hole (inner layer) require window replacement. However, up to 100 flights with cabin differential pressure not exceeding 6.0 psi are allowed until the window is replaced.

b. Allowable cracks in inner layer. Cracks emanating from attachment/fastener holes should be monitored for growth after pressurized flights. The window must be replaced if total crack length is 0.25 inch or if crack has extended through the layer thickness to the outer surface. Cracks are not allowed for adjacent holes or for more than 25 percent of the holes along any side of the window.

c. Cabin pressure restrictions for cracked inner layers. Full cabin pressure is allowed for cracks within the above limits.

Note

Shallow scuffs and scratches may be removed from acrylic windows by using Micro-Mesh Kit KR-70. (Micro-Surface Finishing Products, Box 456, Wilton, IA 52778, Phone 319-732-3240.)

56.2.4 GLASS VINYL SEAL. (See figure 56-2.) To prevent possible damage from moisture entering the electrically heated windshields, the installations that have the glass-vinyl seal at the outside glass edge should be inspected for breaks, cracks, or discontinuities. Any breaks, cracks, or discontinuities in the seal covering the joint between the window vinyl bumper strip and the outside glass edge are not permissible under any circumstance, and must be repaired or the panel must be replaced. It is recommended that this inspection be performed at 25-hour intervals. For illustration of glass vinyl seal type of windshields, see figure 56-2. To repair windshields having defective bumper strips follow the instructions below:

a. Remove the defective sealant carefully, using a phenolic scraper.

b. Thoroughly clean the surface, using a clean rag moistened with aliphatic naphtha.

c. Dry the area thoroughly.

d. Mask adjacent areas on each side of the joint, leaving a space approximately 1/8 to 3/16 inch wide.

e. Apply one coat of sealant, PRO-SEAL 890A1/2; Essex Chem. Corp., or equal on exposed joint.

f. Allow the sealant to cure from 8 to 10 hours at 77°F.

Note

The curing time may be shortened by subjecting the sealant to heat, but do not permit the temperature to exceed 120°F.

56.2.5 STEEL FRAME WINDSHIELDS. (See figure 56-8.) On aircraft 5046 and up, windshield panels and preferred spare replacement windshield panels for all previous airplanes are panels with steel frames. It is recommended that inspection of the seals of these windshields be performed at each 300-hour inspection period.

The following inspection and repair, if necessary, should be performed on the external windshield panel seals:

a. Inspect the sealant around the edges of each panel for gaps, holes or voids. Use a probe such as a flat, thin spatula or a piece of wire to determine the extent of any holes in the sealant. (See figure 56-8.)

b. Apply masking tape on the glass, panel frame, and aircraft skin around the areas to be repaired. (See figure 56-8.)

c. Seal any gaps, holes, or voids with Silver Pro-Seal 895B1/2, or 895B1 sealant, which is manufactured by the Essex Chem. Corp. Mix the sealant in accordance with the manufacturer's instructions. Apply the sealant with a suitable spatula so that all gaps, holes, and voids are completely filled. The sealant between the steel frame and the glass should be applied so that a 45-degree bevel is formed between the steel frame and the glass. The sealant between the steel frame and the aircraft skin should be applied so that the sealant is flush with the steel frame and the skin. (See figure 56-8.)

CAUTION

Do not allow sealant to contact areas other than those being resealed.

56.3 REMOVAL AND REPLACEMENT.

56.3.1 REMOVAL OF WINDOWS. All windows except the clear vision windshield panels are sealed on installation with parting agent applied at suitable places. The windows may tend to stick on removal. Electrically heated windshields must have wiring disconnected and cabin windows must have the desiccant vial tubing disconnected prior to removal. If the windows do not fall free after the removal of all the attachments, use the palm of the hand or some cushioned object to apply inward pressure to one edge of the window.

CAUTION

Do not use screwdriver or similar sharp tool to pry on window assemblies. This can damage the window frame.

56.3.2 WINDOW SEALING. Three groups of sealants can be used for window installations. They are as follows:

Group I

These sealants have a work life of 2 hours and a cure time (shown in parentheses) at 77°F.

Courtaulds Aerospace Pro-Seal 895, B-2 (48 hours)

Group II

This sealant has a work life of 1 hour and a cure time of 44 hours at 77°F.

Courtaulds Aerospace Pro-Seal 895, B-1

Group III

This sealant has a work life of 30 minutes and a cure time of 24 hours at 77°F.

Courtaulds Aerospace Pro-Seal 895, B-1/2

Note

The work life and cure times given in this procedure are for a temperature of 77°F. The work life and cure times are cut approximately in half for each 15°F rise in temperature, and doubled for each 15°F drop in temperature. The sealant materials should not be worked or cured below 60°F or above 110°F. For quicker installation, sealants may be mixed and applied at the lower temperatures, and heat lamps may be used on the installation to raise the temperature and accelerate the curing. The use of sealants with work life of less than 15 minutes or cure time in excess of 48 hours is not practical for window installations.

If the above materials are not available, the sealants listed below may be used. These sealants will work satisfactorily but come in a gray or brown color:

Courtaulds Aerospace	PR1440, Class B, MIL-S-8802
Flamemaster Corp.	CS3204, Class B, MIL-S-8802
Morton International	MC-236, Class B, MIL-S-8802

Note that the Class B sealants are the only acceptable alternates. Class A sealants contain solvents which may attack some of the plastics used in the window assemblies and should not be used.

56.3.3 WINDOW REPLACEMENT. Windshield panels are replaced as detailed in figure 56-9. Skylights and cabin window panels are replaced as detailed in figure 56-10. The forward windshield side panel re-installation may require JF400-7 retainer modification. Contact Lockheed Martin Aeronautical Systems Support Company for disposition.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

56.3.4 REPLACEMENT OF CLEAR VISION WINDSHIELD MOLDED SEAL. A seal is provided around the periphery of the airframe opening for the clear vision windshield. On early serial airplanes, a rubber seal is attached to the frame for this purpose; on late serial airplanes, a molded sealant is used. In either case, when the clear vision windshield is replaced or when the seal becomes damaged, remove the old seal and install a new molded seal. Use the following procedure to replace this seal:

a. Remove the old seal in accordance with either step (b) or step (c).

b. If the old seal is rubber, remove it by applying aliphatic naphtha to the adhesive between the seal and the frame, and pull the seal up as the adhesive loosens.

c. If the old seal is composed of sealant material, scrape the old sealant off with a phenolic scraper.

d. Apply a thin coat of parting agent DOW CORNING DC-4 (MIL-S-8660B) or equal compound to the faying surface of the clear vision windshield assembly. Wipe off excess parting agent.

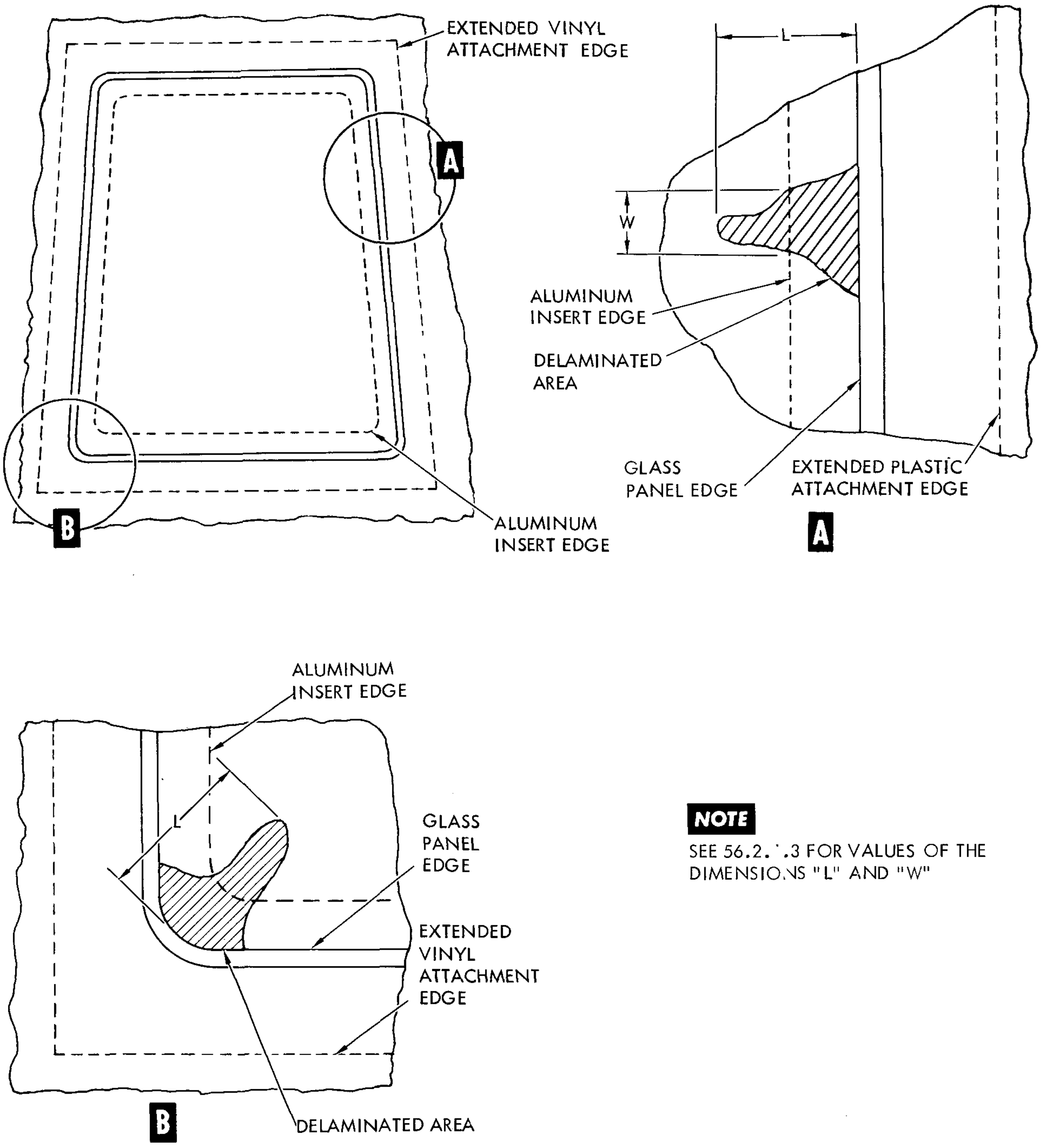
e. Coat the mating surface of the airframe opening with DOW CORNING SILASTIC RTV 1200, SILICONE PRIMER. Coat thickness should be 0.0003 to 0.001 inch. Allow primer to dry for a minimum of one-half hour.

f. Apply weather resistant synthetic rubber windshield sealing compound DOW CORNING NO. 93-006 to the mating surface of the airframe opening. Use sufficient quantity to ensure adequate seal when the clear vision windshield assembly is closed. After the assembly is closed, be sure that a continuous bead of exuded sealing compound is formed on the inner periphery of the assembly, and that the gap between the outer face of the assembly and the airframe skin surface is filled. Fair sealant on the exterior to provide a flush surface.

g. Sealant should be cured to a tack-free condition before opening the windshield assembly.

h. The mating surfaces of the windshield assembly and the seal on the airframe opening should be coated lightly with talc.

56.3.5 REPLACEMENT OF THERMOSTAT - AFT SIDE WINDSHIELD. All windshields except the aft side windshield have thermostats imbedded in the glass. The thermostat for the aft side windshield is cemented to the inner glass ply. These may become detached with service, and may be cemented back into place with Pro-Seal 890, Class B-1/2;Courtaulds Aerospace, or equal.



NOTE

SEE 56.2.1.3 FOR VALUES OF THE DIMENSIONS "L" AND "W"

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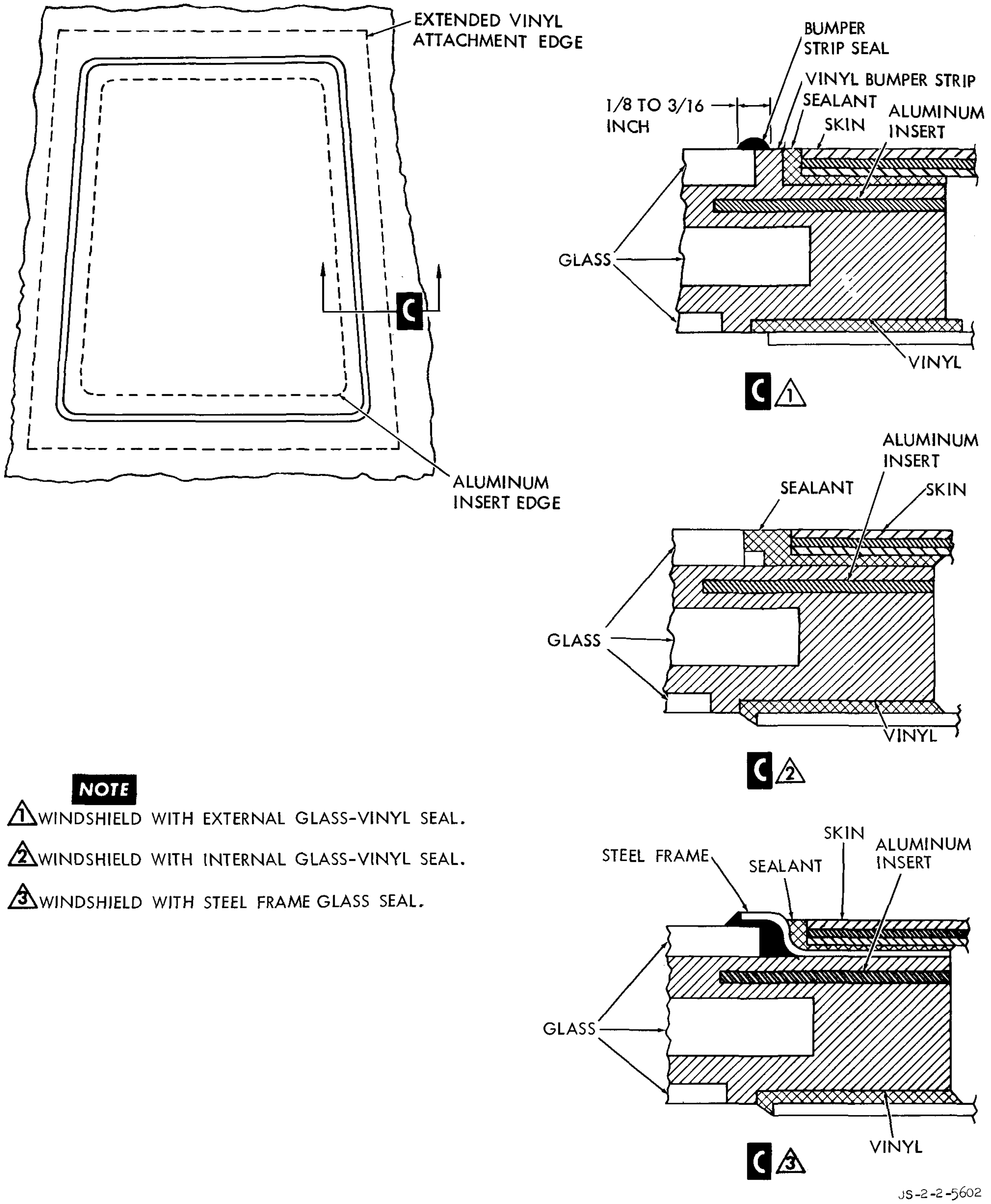
Figure 56-1. Delamination Limits for Windshields



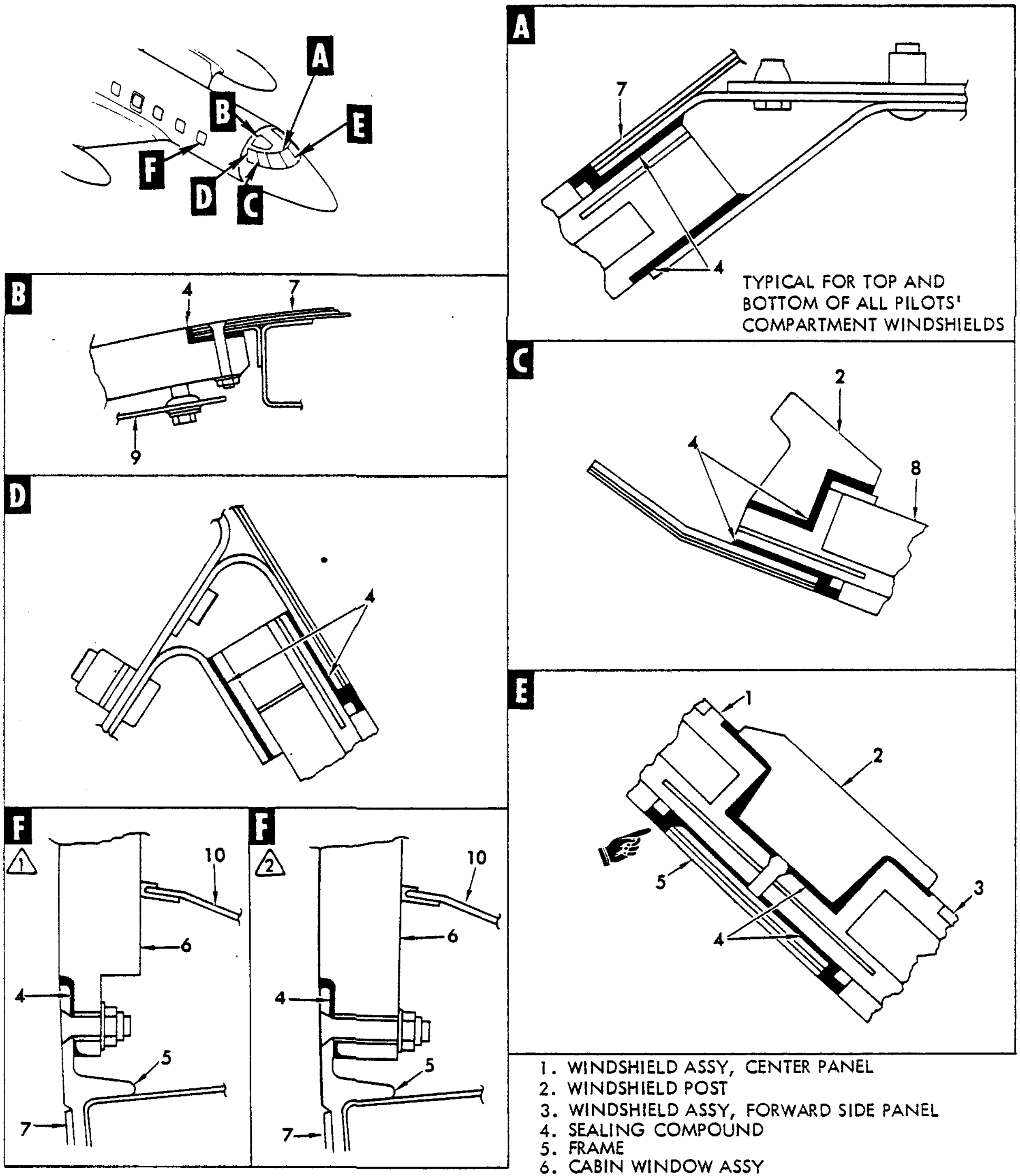
NOTE

1 WINDSHIELD WITH EXTERNAL GLASS-VINYL SEAL.

2 WINDSHIELD WITH INTERNAL GLASS-VINYL SEAL.

3 WINDSHIELD WITH STEEL FRAME GLASS SEAL.

Figure 56-2. Types of Windshields



1. WINDSHIELD ASSY, CENTER PANEL
2. WINDSHIELD POST
3. WINDSHIELD ASSY, FORWARD SIDE PANEL
4. SEALING COMPOUND
5. FRAME
6. CABIN WINDOW ASSY
7. SKIN
8. WINDSHIELD ASSY, AFT SIDE PANEL
9. SKYLIGHT ASSY
10. TRIM PANEL

NOTE

△1 AIRPLANES WITH JF 442 WINDOW ASSY

△2 AIRPLANES WITH JF 764 WINDOW ASSY

JS-1-2-X0/9-5603

Figure 56-3. Sealing of Windshields and Windows

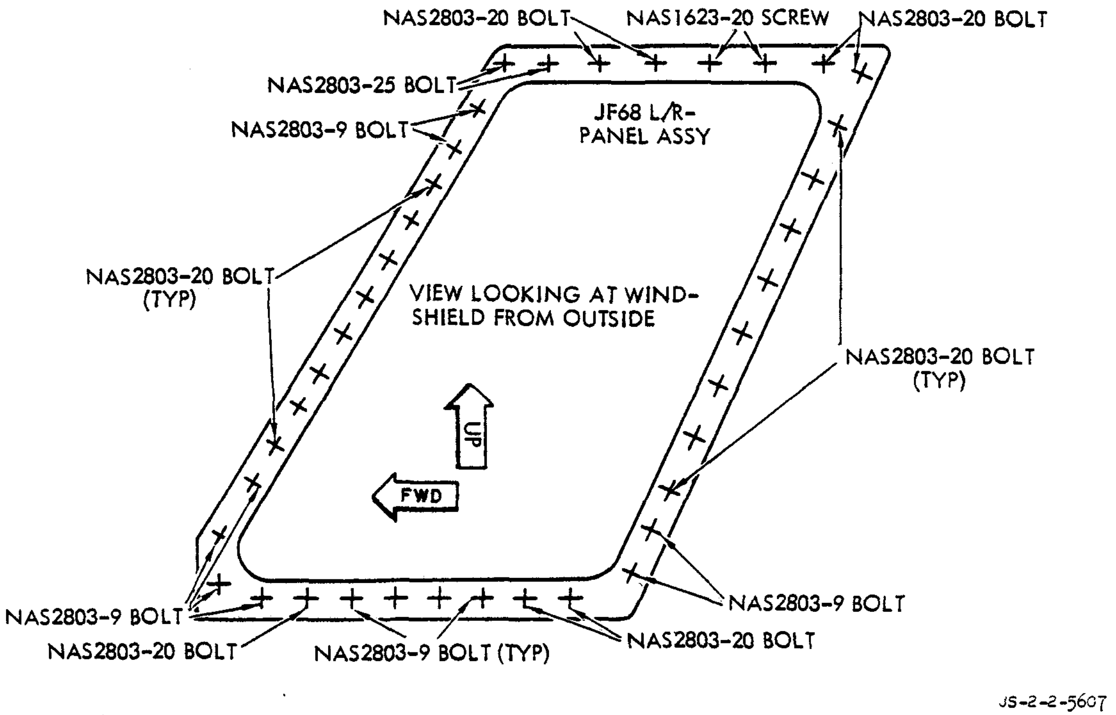


Figure 56-4. Clear Vision Windshield Screw and Bolt Length Diagram

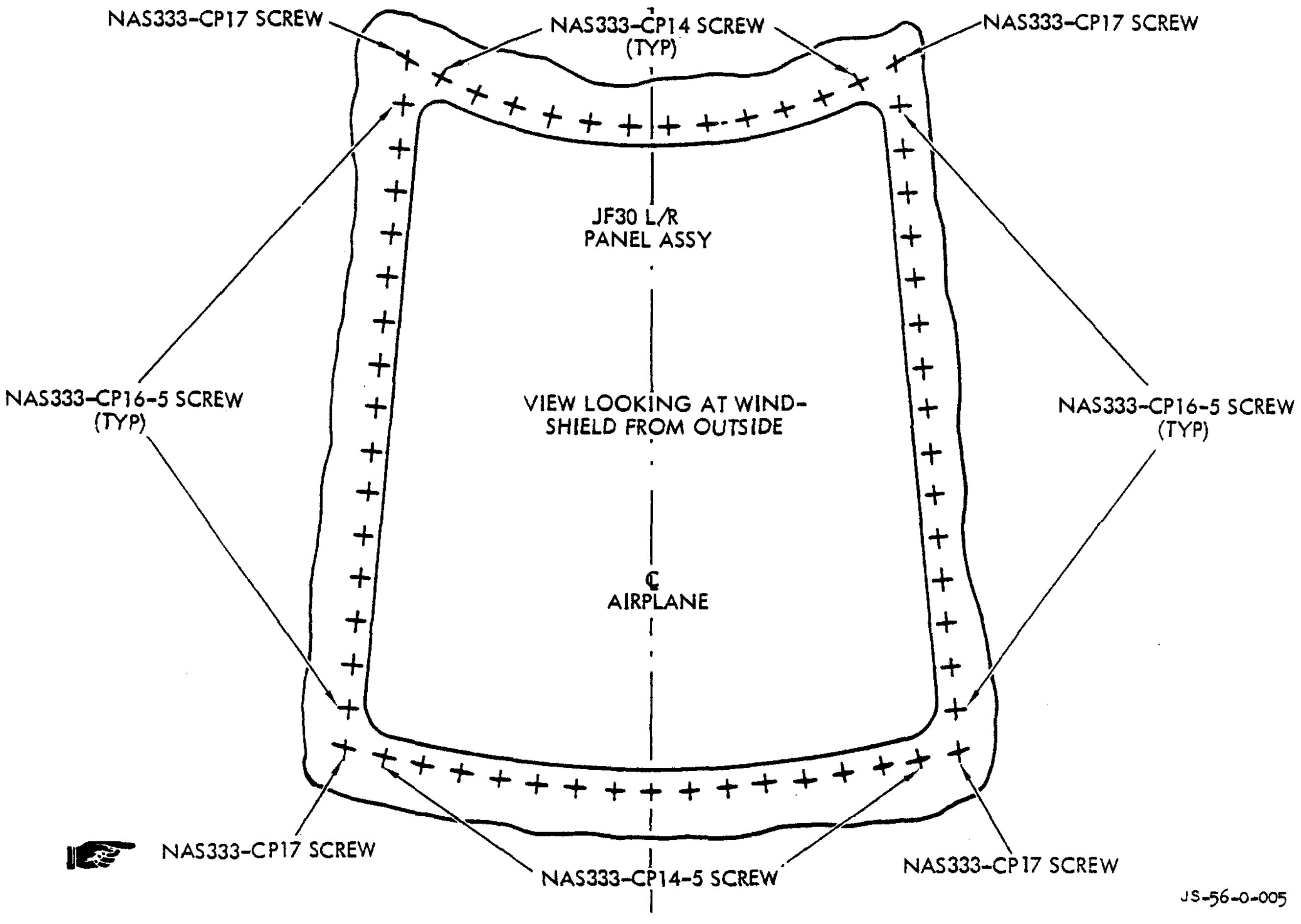


Figure 56-5. Center Windshield Screw and Bolt Length Diagram

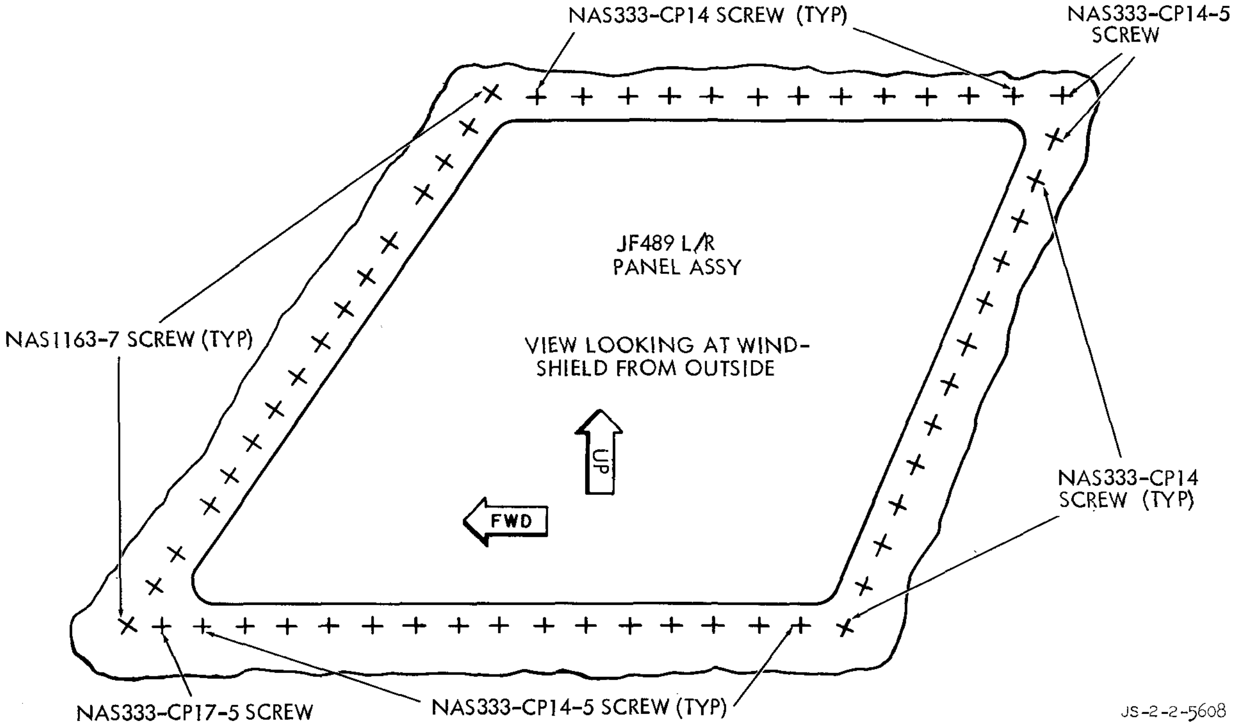


Figure 56-6. Aft Side Windshield Screw Length Diagram

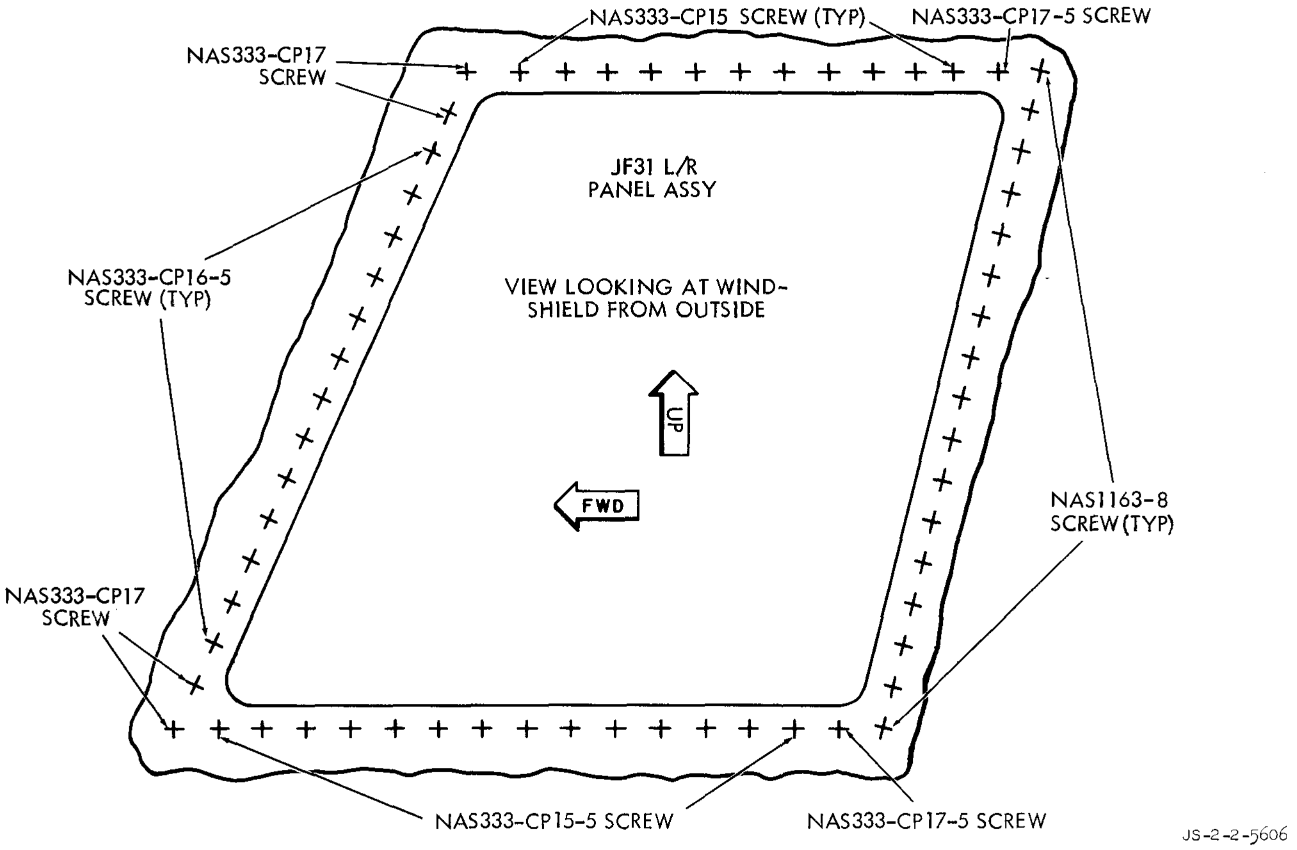
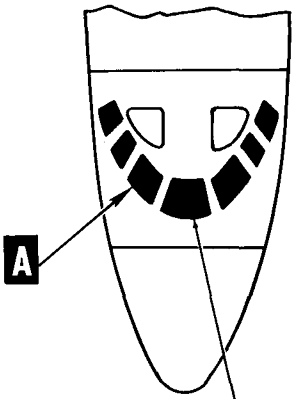
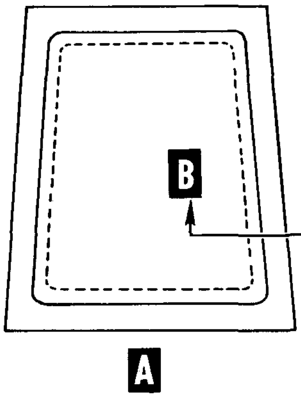


Figure 56-7. Forward Side Windshield Screw and Bolt Length Diagram



INSPECT EDGES OF SEVEN WINDSHIELDS OR WINDOWS FOR GAPS, HOLES, OR VOIDS IN SEALANT. APPLY SEALANT AS REQUIRED TO COMPLETELY FILL GAPS, HOLES, OR VOIDS.



TYPICAL

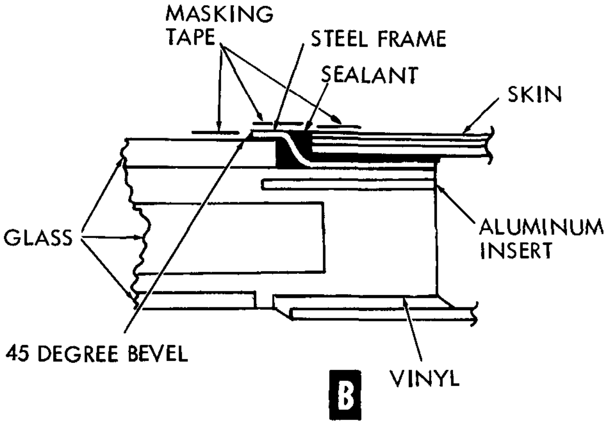


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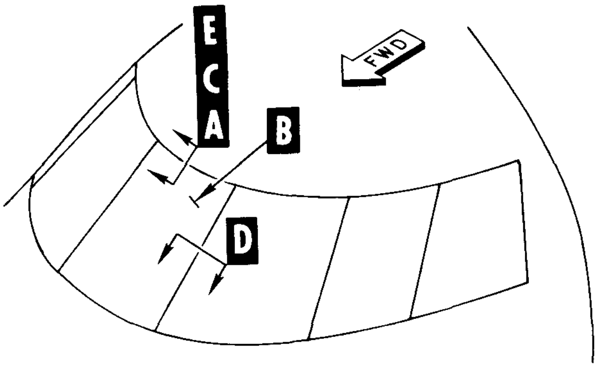
Figure 56-8. Inspection and Repair of Windshield Sealant



STEP 1

THOROUGHLY CLEAN ALL OLD SEALANT FROM THE FRAMES, FITTINGS, AND BACK-UP STRUCTURE USING A PHENOLIC SCRAPER AND ALIPHATIC NAPHA (FEDERAL SPECIFICATION TT-N-95).

STEP 2

APPLY A LIBERAL AMOUNT OF DOW CORNING DC 4 PARTING AGENT TO THE FOLLOWING:

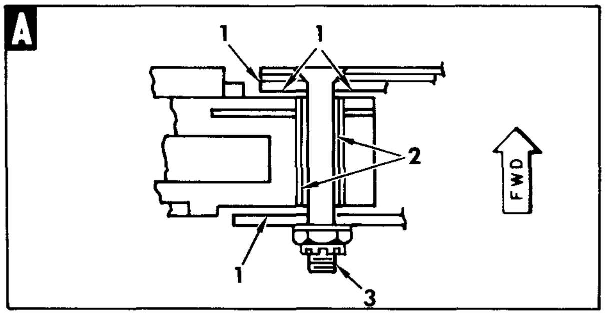
(A) - ALL SEALANT CONTACT SURFACES OF THE FUSELAGE SKIN, FRAMES, POSTS, AND BACK-UP STRUCTURE (1). WIPE OFF THE EXCESS WITH A SOFT COTTON CLOTH.

(B) - THE INSIDE ONLY OF THE BUSHINGS IN THE WINDSHIELD PANELS (2). DO NOT WIPE OFF THE EXCESS.

(C) - ALL SCREWS COMMON TO STRUCTURE AND WINDSHIELDS (3). DO NOT WIPE OFF THE EXCESS. DO NOT APPLY PARTING AGENT TO SCREWS COMMON TO STRUCTURE ONLY.

CAUTION

KEEP ALL WINDSHIELD SURFACES ABSOLUTELY FREE OF THE PARTING AGENT EXCEPT FOR THE INSIDE OF THE BUSHINGS.

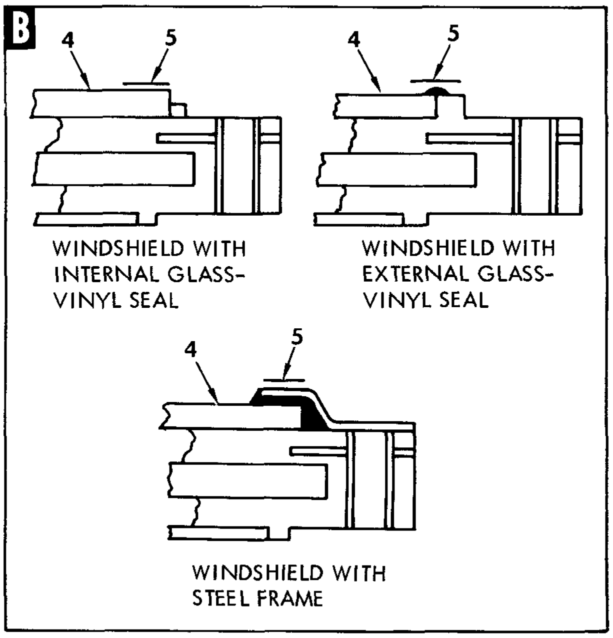


STEP 3

APPLY A STRIP OF 1/2 - INCH WIDE MASKING TAPE (5) AROUND THE WINDSHIELD (4).

NOTE

WINDSHIELDS MAY BE ONE OF THREE TYPES AS SHOWN, DEPENDING ON THE AIRPLANE SERIAL.



WINDSHIELD WITH INTERNAL GLASS-VINYL SEAL

WINDSHIELD WITH EXTERNAL GLASS-VINYL SEAL

WINDSHIELD WITH STEEL FRAME

NOTE

SEE PARAGRAPH 56.3.2 FOR SEALANT HANDLING INSTRUCTIONS.

STEP 4

INSTALL A JK 1550-1 SPACER TO THE OUTBOARD SIDE OF THE PANEL BETWEEN EVERY THIRD ATTACH HOLE TO ENSURE A CONTINUOUS LAYER OF SEALANT BETWEEN THE PANEL AND THE STRUCTURE. APPLY A LIBERAL AMOUNT OF SEALANT (6) AROUND THE OUTSIDE PERIPHERY OF THE WINDSHIELD TO THE FAYING SURFACE (8).

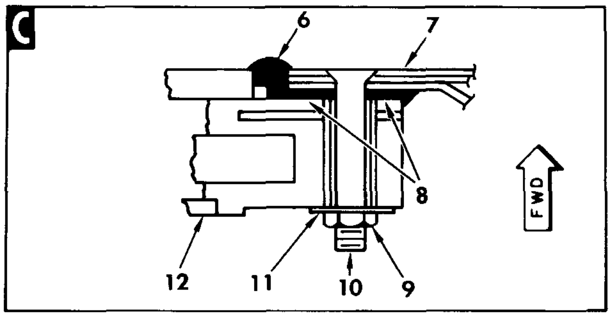


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Figure 56-9. Windshield Installation Instructions (Sheet 1 of 3)

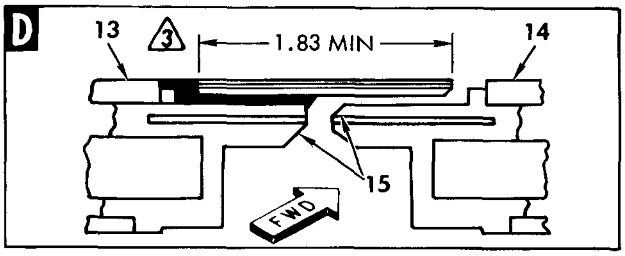
STEP 5

PUSH THE WINDSHIELD GENTLY INTO POSITION AND INSTALL APPROXIMATELY 12 TEMPORARY ATTACHMENT SCREWS (10), WITH LARGE OD WASHER (11), AND NUTS (9) EQUALLY SPACED AROUND THE WINDSHIELD.

NOTE

1. DO NOT INSTALL ANY OF THE WINDSHIELD POSTS OR OTHER BACK-UP STRUCTURE AT THIS TIME.
2. IF REPLACEMENT OF THE CENTER CURVED WINDSHIELD IS NECESSARY, IT MAY BE DESIRABLE TO CUT SOME OF THE VINYL FROM THE FORWARD EDGE OF THE SIDE WINDSHIELD (13) AS WELL AS THE EDGES OF THE NEW CENTER WINDSHIELD (14). DO NOT TRIM MORE THAN THE AMOUNT SHOWN (15).

△3 NO. 1 POST ONLY. THERE IS ALSO A 0.80 X 45° CHAMFER ON THE INSIDE OF THE INNER DOUBLER (JF391-1 L/R) ON THE INBOARD SIDE OF POST NO. 1 ONLY.



STEP 6

USE THE TEMPORARY SCREWS (10) AND NUTS (9) TO GENTLY PULL THE WINDSHIELD (12) TO A FAIRED POSITION WITH THE AIRPLANE EXTERIOR SURFACE (7). DO NOT USE MORE THAN 5 INCH-POUNDS TORQUE ON ANY SCREW OR NUT DURING THIS INITIAL BEDDING OPERATION.

CAUTION

DO NOT STRAIN OR WARP THE WINDSHIELD IN ANY WAY DURING THIS INITIAL BEDDING INSTALLATION.

STEP 7

AFTER THE SEALANT (6) HAS CURED SUFFICIENTLY TO HOLD THE WINDSHIELD IN POSITION WITHOUT THE TEMPORARY SCREWS (10), WASHER (11), AND NUTS (9), REMOVE THESE, AND PROCEED WITH THE FOLLOWING STEPS.

STEP 8

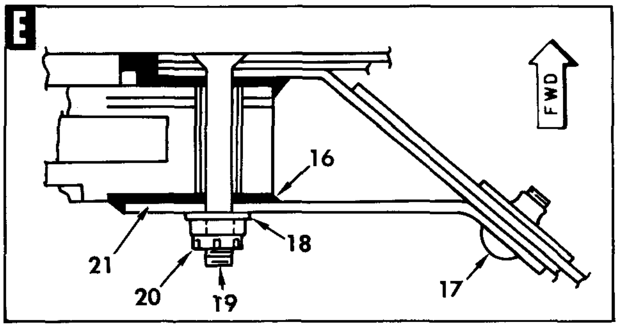
PUSH OUT THE PLUGS OF SEALANT IN THE WINDSHIELD ATTACH HOLES.

NOTE

WHEN REMOVING THE PLUGS OF SEALANT, PUSH FROM THE INSIDE OF THE AIRPLANE TO THE OUTSIDE TO PREVENT PUSHING THE WINDSHIELD PANEL OUT.

STEP 9

APPLY A LIBERAL AMOUNT OF SEALANT (16) TO THE BACK-UP STRUCTURE SURFACE (21) COMMON TO THE WINDSHIELD AND PUSH THIS STRUCTURE INTO POSITION.



STEP 10

PERMANENTLY INSTALL ALL ATTACHMENTS (17) COMMON TO THE STRUCTURE ONLY.

STEP 11

LOOSELY INSTALL ALL SCREWS (19), WASHERS (18), AND NUTS (20), THROUGH THE STRUCTURE AND WINDSHIELDS.

CAUTION

DO NOT TORQUE THE NUTS (20) OR SCREWS (19) AT THIS TIME.

STEP 12

REMOVE THE EXCESS SEALANT AND ALLOW THE SEALANT TO CURE THOROUGHLY.

CAUTION

IMPROPER TIGHTENING OF THE SCREWS AND NUTS THROUGH THE WINDSHIELDS WILL CAUSE DELAMINATION. NEVER PICKUP NUTS AND SCREWS THROUGH THE WINDSHIELDS FOR WIRE BUNDLE CLAMPS OR OTHER EQUIPMENT.

JS-1-2-X0/8-5609-2

Figure 56-9. Windshield Installation Instructions (Sheet 2 of 3)

STEP 13

APPLY THE CORRECT TORQUE, AS INDICATED IN THE TABLE BELOW, TO THE ATTACHING SCREWS (19) AFTER ALL OF THE SEALANT HAS THOROUGHLY CURED.

NOTE

SOME AIRCRAFT HAVE NAS1153 TORQUE SET SCREWS WITH NAS1291-3, NAS679C10 OR MS21042-3 NUTS. OTHER AIRPLANES HAVE NAS517-3 OR NAS333 SCREWS WITH AN310-3 NUTS. APPLY TORQUE TO THE ATTACHMENTS ACCORDING TO THE FOLLOWING TABLE. ALL AIRCRAFT USE SOME SELF-LOCKING SCREWS IN TAPPED HOLES.

TYPE OF FASTENER	TORQUE REQUIREMENTS
SCREWS WITH CASTELLATED NUTS (AN310-3)	TIGHTEN NUTS TO A TORQUE VALUE OF 10 (± 1.0) INCH - POUNDS. INSTALL A SINGLE STRAND OF 0.032 INCONEL SAFETY WIRE THROUGH THE NUTS AND SCREWS, EXCEPT AT THE TOP OF THE WINDSHIELDS IN THE VICINITY OF THE POWER OR THERMISTOR TERMINALS WHERE SAFETY WIRE WOULD INTERFERE WITH THE ELECTRICAL PLUGS. USE MS24655-130 COTTER PINS WHERE SAFETY WIRE IS NOT USED.
SCREWS WITH SELF-LOCKING NUTS (NAS1291-3, NAS679C10, MS21042-3)	TIGHTEN SELF-LOCKING NUTS AS FOLLOWS: INITIALLY TIGHTEN EACH NUT TO A TORQUE VALUE OF 20 INCH - POUNDS, THEN BACK OFF THE NUT APPROXIMATELY 6 FULL TURNS. AGAIN, TIGHTEN THE NUT ENOUGH TO OVERCOME THE SELF-LOCKING TORQUE. NOTE THIS TORQUE VALUE. TIGHTEN THE NUT 10 INCH - POUNDS MORE THAN THE SELF-LOCKING TORQUE.
SCREWS IN TAPPED HOLES AND/OR SCREWS WITH INACCESSIBLE SELF-LOCKING NUTS	TIGHTEN SELF-LOCKING SCREWS IN TAPPED WINDSHIELD POST OR IN AREAS WHERE THE NUTS ARE INACCESSIBLE FOR TIGHTENING AS FOLLOWS: TIGHTEN EACH SCREW TO 20 INCH - POUNDS OF TORQUE, THEN BACK OFF THE SCREW APPROXIMATELY SIX FULL TURNS. TIGHTEN THE SCREW ENOUGH TO OVERCOME THE SELF-LOCKING TORQUE. NOTE THIS TORQUE VALUE. TIGHTEN THE SCREW 10 INCH - POUNDS MORE THAN THE SELF-LOCKING TORQUE.

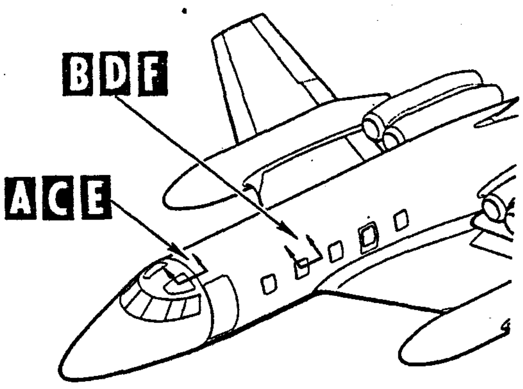
STEP 14

ATTACH HEATER ELECTRICAL CONNECTIONS AS REQUIRED.

JS-1-2-X0-3-5609-3

Figure 56-9. Windshield Installation Instruction (Sheet 3 of 3)

Use and/or disclosure is governed by the statement on the title page of this document.



STEP 1

THROUGHLY CLEAN ALL OLD SEALANT FROM THE FRAMES AND STRUCTURE USING A PHENOLIC SCRAPER AND ALIPHATIC NAPHTHA (FEDERAL SPECIFICATION TT-N-95).

WARNING

NEVER ALLOW THE SKYLIGHT OR CABIN WINDOWS TO COME IN CONTACT WITH ANY SOLVENT OR COMPOUND, OTHER THAN THOSE SPECIFIED HEREIN. THE USE OF OTHER SOLVENTS OR CLEANERS MAY CAUSE DETERIORATION OF THE PLASTIC PANELS.

STEP 2

CLEAN THE CONTACT SURFACES OF THE SKYLIGHT OR WINDOW ASSEMBLY WITH FEDERAL SPECIFICATION TT-N-95 ALIPHATIC NAPHTHA.

STEP 3

APPLY A LIBERAL AMOUNT OF DOW CORNING DC4 PARTING AGENT TO THE FOLLOWING:

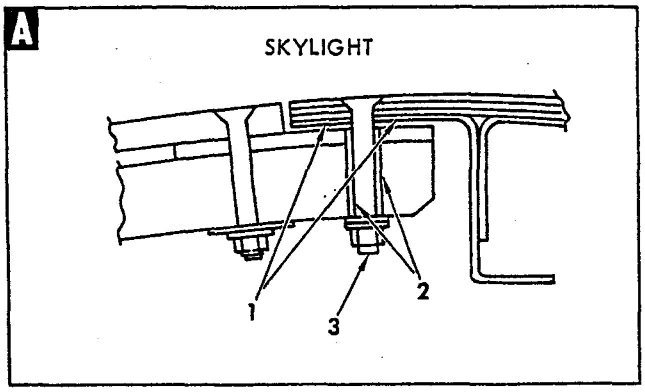
CAUTION

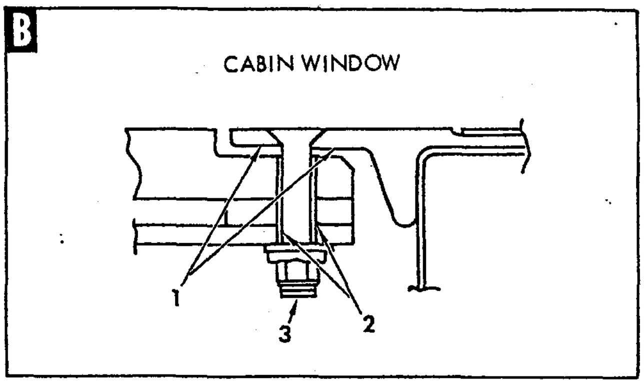
KEEP ALL PANEL SURFACES ABSOLUTELY FREE OF THE PARTING AGENT EXCEPT FOR THE INSIDE OF THE BUSHINGS.

(A) ALL SEALANT CONTACT SURFACES OF THE FUSELAGE SKIN AND FRAMES (1). WIPE OFF THE EXCESS PARTING AGENT WITH A SOFT COTTON CLOTH.

(B) ONLY THE INSIDE OF THE BUSHINGS (2) IN THE PANELS. DO NOT WIPE OFF THE EXCESS PARTING AGENT.

(C) ALL SCREWS (3) COMMON TO STRUCTURE PANELS. DO NOT WIPE OFF THE EXCESS PARTING AGENT. DO NOT APPLY PARTING AGENT TO SCREWS COMMON TO STRUCTURE ONLY.





STEP 4

APPLY A STRIP OF 1/2 INCH WIDE MASKING TAPE (5) AROUND THE PANEL (4).

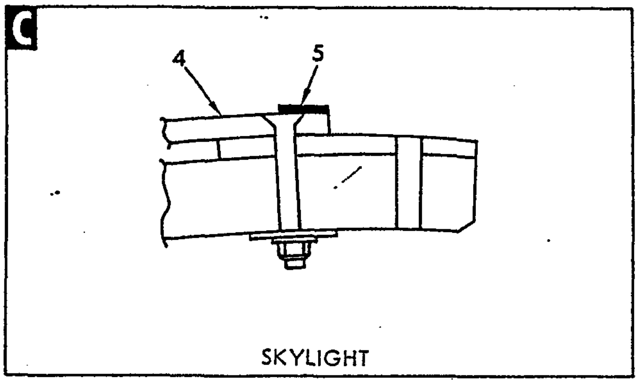


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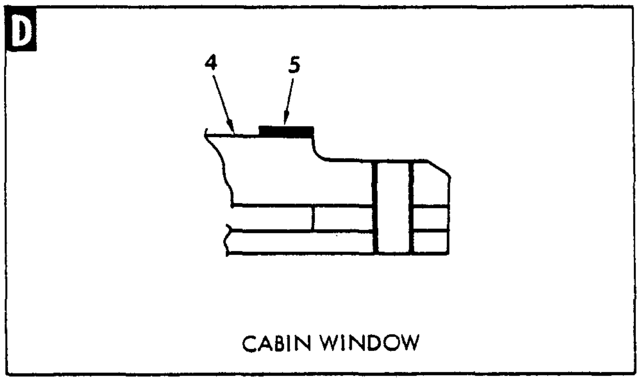
Figure 56-10. Skylight and Cabin Window Installation Instructions (Sheet 1 of 2)



STEP 5

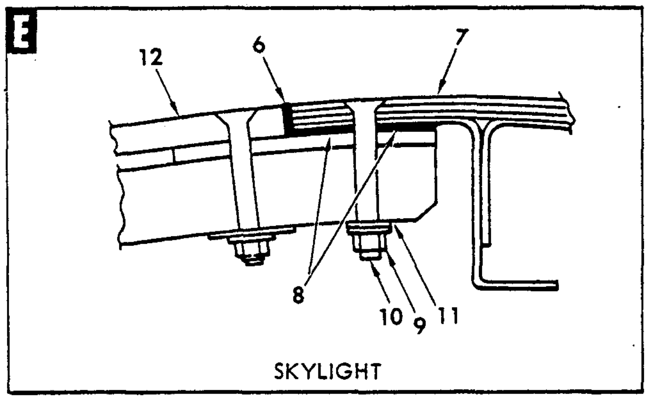
ADD A JK1550-1 SPACER (J. O. KING, INC., ATLANTA, GA) OR SIMILAR 3/16 INCH DIA. CORK DISCS, 1/32 - INCH THICK, BETWEEN EVERY THIRD ATTACH HOLE IN THE PANEL TO ENSURE A CONTINUOUS BED OF SEALANT BETWEEN THE PANEL AND THE FRAME.

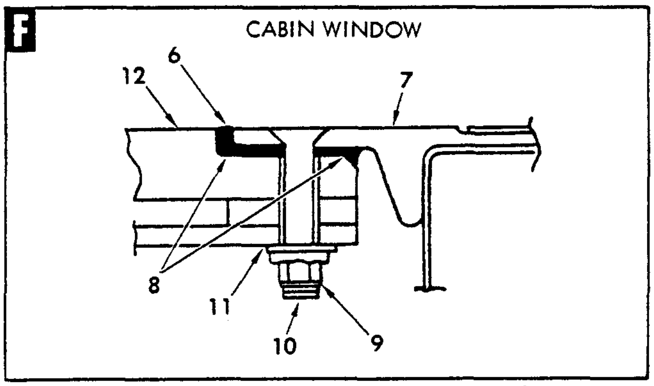
NOTE

SEE PARAGRAPH 56.3.2 FOR SEALANT HANDLING INSTRUCTIONS.

STEP 6

APPLY A LIBERAL AMOUNT OF SEALANT (6) AROUND THE OUTSIDE PERIPHERY OF THE PANEL TO THE FAYING SURFACE (8).





NOTE

CABIN WINDOW PART NO. JF 764-7 (THE LATEST SPARE) USES NINE LESS FASTENERS ON THE TOP AND THE BOTTOM THAN THE WINDOW INSTALLED ON AIRCRAFT 5086 AND UP. THE 18 EXTRA HOLES IN THE WINDOW FRAME MAY BE FILLED WITH SEALANT AND SMOOTHED UP ON THE AIRCRAFT EXTERIOR AS A NEW WINDOW IS INSTALLED OR, THE HOLES MAY BE DOUBLE-FLUSHED WITH MS20426A6 RIVETS PRIOR TO WINDOW INSTALLATION. IF RIVETS ARE USED, THE INSIDE OF THE FRAME SHOULD HAVE A 0.010 - INCH COUNTERSINK ADDED AT EACH HOLE. THE COUNTERSINK SHALL HAVE AN ALODINE 1200 TREATMENT FOLLOWED WITH A COATING OF ZINC CHROMATE PRIMER.

STEP 7

PUSH THE SKYLIGHT OR WINDOW PANEL INTO POSITION AND APPROXIMATELY EQUALIZE THE STEP ON ALL SIDES BETWEEN THE OUTSIDE STRUCTURE (7) AND THE PANEL SURFACE (12) WHERE THE OUTSIDE CONTOURS MEET. INSTALL ONE OR TWO ATTACHING SCREWS (10), WASHERS.(11), AND NUTS (9) ALONG EACH EDGE OF THE PANEL. TIGHTEN THE NUTS SUFFICIENTLY TO BRING THE OUTSIDE CONTOUR OF THE PANEL TO WITHIN 0.25 INCH OF THE FRAME CONTOUR (7). INSTALL ALL REMAINING FASTENERS AND TIGHTEN UNIFORMLY TO WITHIN ± 1/16 INCH OF THE CONTOUR. DO NOT TIGHTEN THE NUTS MORE THAN 20 INCH - POUNDS TO INTIALLY BED THE SEALANT. REMOVE EXCESS SEALANT AND MASKING TAPE (5). ALLOW SEALANT TO CURE BEFORE FINAL TIGHTENING OF THE FASTENERS.

STEP 8

ALLOW SEALANT TO CURE COMPLETELY. TIGHTEN UP OPPOSITE NUTS (CRISS-CROSS PATTERN) ALL AROUND TO A TORQUE VALUE OF 13 TO 15 INCH - POUNDS.

STEP 9

REPEAT STEP 8 BUT TIGHTEN THE NUTS (IN THE SAME MANNER) TO A TORQUE VALUE OF 19 TO 21 INCH - POUNDS.

STEP 10

USE A SOFT, COTTON CLOTH AND CLEAN THE WINDOW SURFACES WITH MIRROR GLAZE, MGH-10 (MIRROR BRIGHT POLISH CO.) OR PLEXIGLASS CLEANER.

STEP 11

RECONNECT THE DESICCANT VIALS ON THE CABIN WINDOW INSTALLATIONS (REFER TO SECTION 30.6).

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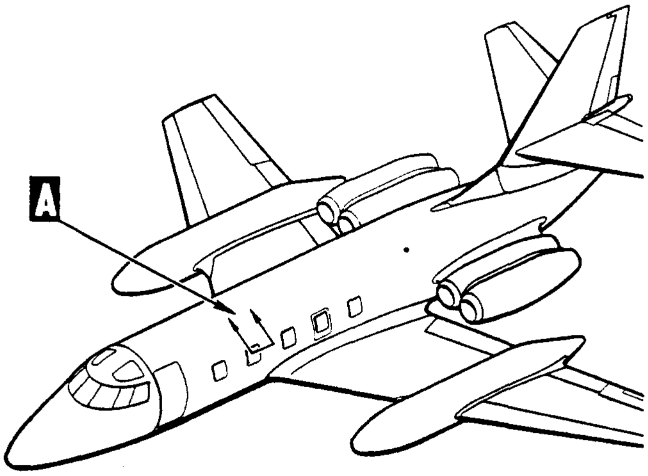
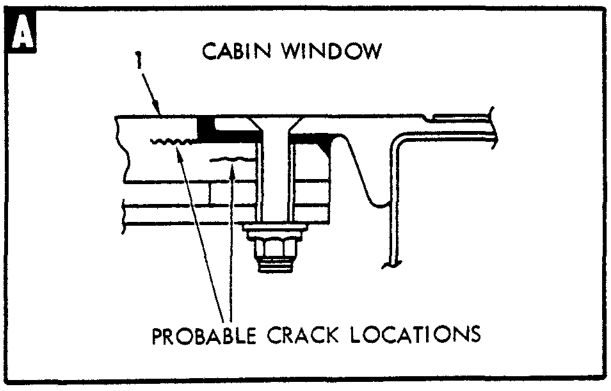
Figure 56-10. Skylight and Cabin Window Installation Instructions (Sheet 2 of 2)



NOTE

WINDOW INSPECTION KIT, P/N 04-56001-1 OR -2 (MARCAR ENTERPRISES, 847 ISOM ROAD, SAN ANTONIO, TX 78216, PHONE 512-349-3614) IS USEFUL IN MAKING THIS INSPECTION.



STEP 1

WITHOUT REMOVING THE WINDOW, INSPECT THE WINDOW FROM THE OUTSIDE SURFACE USING A STRONG LIGHT. LOOK FOR SEPARATION INDICATING A CRACK IN THE OUTSIDE PANEL (1) EMANATING FROM THE RABBET RADIUS AS SHOWN IN DETAIL A. INSPECT ALL FOUR SIDES OF EACH CABIN WINDOW IN THIS MANNER. REPLACE WINDOW IF ANY SEPARATION OR CRACKS ARE FOUND IN THIS AREA.

JS-1-2-X0/9-5611

Figure 56-11. Cabin Widow Rabbet Visual Inspection

CHAPTER 57
WINGS

Contents | Page

GENERAL 57-1
- Box Structure 57-1
- Fixed Leading Edge 57-2
- Trailing Edge Structure 57-2
- Trailing Edge Flap 57-2
- Aileron and Tab 57-2
- Wing Tip 57-2
- Fairings 57-2
- External Fuel Tank 57-2
- Leading Edge Flaps 57-2A
  - Leading Edge Flap - Sheet Metal. 57-2A
  - Leading Edge Flap - Honeycomb 57-3
  - Detail Wing Components Material and Identification 57-3

FIGURES

57-1. Wing Assembly Breakdown 57-5
57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure 57-6
57-3. Wing Leading Edge, Trailing Edge, Tip, and Box Section Skins 57-12
57-4. Wing Integrally Stiffened Skin Panels Lower Surface Geometry 57-14
57-5. Wing Integrally Stiffened Skin Panels Upper Surface Geometry 57-15
57-6. Aileron Skin and Structure 57-16
57-7. Wing Fillets Skin and Structure 57-18
57-8. Trailing Edge Flap Skin and Structure 57-20
57-9. External Wing Tank Skin and Structure 57-21
57-10. Leading Edge Flap Skin and Structure - Sheet Metal 57-24
57-11. Leading Edge Flap Skin and Structure - Honeycomb 57-25

57-1 GENERAL. (See figure 57-1.) The wing consists of the box structure, fixed leading edge, movable leading edge (leading edge flap), tips, trailing edge, aileron, trailing edge flap, fairings, and an external fuel tank. The box structure and fixed leading edge are sealed to form integral fuel tanks.

57.1.1 BOX STRUCTURE. The aluminum alloy box structure is a combination, box beam-torsion box, which forms the primary load carrying structure of the wing. This box structure consists of three shear beams which extend the full length of the wing, ribs which tie the three shear beams together, and integrally stiffened skin on both surfaces.

a. The three shear beams are located at the 12-percent, 43.5 percent, and 65-percent chord lines. These beams are assembled by riveting, and consist of upper and lower caps, webs, and web stiffeners. The rear beam is sealed over its entire length, and forms the rear boundary of the aft integral tank. The middle beam is sealed outboard to wing station 166, and separates the front and aft integral tanks in this area. The front beam is sealed from wing station 131 outboard. The front beam is the front boundary of the front integral tank between wing stations 131 and 166, and is the front boundary of the aft integral tank outboard of wing station 166.

b. The wing ribs are of riveted construction, and consist of upper and lower caps, webs, and stiffeners. Certain ribs are sealed to form portions of the integral tank boundaries. The wing root rib at wing station 44 is sealed. and forms the inboard end of both the front and aft integral tanks. At wing station 166, the rib is sealed between the front and middle shear beams and separates the front tank from the outboard portion of the aft tank. The rib at wing station 318 is sealed to form the outboard boundary of the aft tank. All of the other ribs have openings in the webs to permit fuel passage. The ribs are spaced at approximately 20 inches, and are normal to the middle shear beam.

57.1.1 (Cont).

c. The skin panels are extrusions incorporating integral "I" section stiffeners, running spanwise along the panels. These panels are both machined and chemically milled to gradually reduce the cross sectional area of the panels as they extend outboard. The lower surface panels are attached to the shear beams and ribs by flush rivets, and are sealed completely to form the lower boundary of the integral tanks. The upper surface panels are attached to the ribs and shear beams by flush screws, and are removable. For detailed instructions on wing cover removal see Chapter 28.

CAUTION

The wings must be adequately supported by cradles when the upper panels are removed, to prevent damaging the other box structure.

Small sections of rib-like structure are permanently attached to the inside surface of the upper panels. These partial ribs mate with the regular box structure ribs when the panels are installed. The upper surface panels are completely sealed to form the upper boundary of the integral tanks. This seal is broken when the panels are removed, making it necessary to reseal the tanks upon re-installation of the panels.

d. The wing is attached to the fuselage by wing joint fittings secured to the box structure at the wing root rib. These fittings are attached by tension bolts to five heavy, forged fuselage frames. This joint is at fuselage butt line 41.5.

57.1.2 FIXED LEADING EDGE. The portion of leading edge inboard of the external tank is attached to the front shear beam by flush screws. This portion of the leading edge consists of aluminum alloy skin attached to formed ribs by flush rivets, and is sealed to form a part of the integral fuel tanks.

57.1.3 TRAILING EDGE STRUCTURE. The wing trailing edge structure is fabricated of aluminum alloy formed ribs and skin using flush rivets. The trailing edge serves as a fairing between the wing box section and trailing edge flap, in the inboard portion of the wing, and between the wing box section and the aileron, in the outboard portion of the wing.

57.1.4 TRAILING EDGE FLAP. A double slotted flap divided into two sections is provided on each wing. Both flap sections are similar in design and incorporate three spanwise, formed sheet metal beams, ribs, and double skins. The double skin consists of a smooth outer skin bonded to a beaded inner skin. Spanwise channels provide additional stiffening. The ribs are of formed sheet metal. All of the flap structure is aluminum alloy clad sheet. Both spot-welding and flush riveting are used to assemble the flaps.

57.1.5 AILERON AND TAB. The ailerons are of aluminum alloy construction, consisting of front and rear beams, formed ribs, and chemically milled skins. The aileron is attached to the upper cap of the rear wing spar by a piano-type hinge. A trim tab is installed on the left aileron. The tab structure is of aluminum alloy, and consists of a front beam, skin and formed ribs. It is supported by a continuous piano-type hinge. Spot-welding and flush riveting are used to assemble the aileron and tab.

57.1.6 WING TIP. The wing tip is a molded fiberglas structure. It is attached to the wing by means of flush screws, and is removable.

WARNING

The molded fiberglass wing tips house flux valves which excite the pilot's and co-pilot's gyrosyn compass indicators (see Section 34.6). These flux valves are highly sensitive to magnetic material in their immediate area. All the screws for panel attachment at the outboard wing box structures are non-magnetic type. A careful check should be made after removal and replacement of the wing tips to make sure that no magnetic bolts have been inadvertently used for reattachment. This applies equally to the screws holding the wing tip navigation lights.

Note

The proper hardware for use in this area is identified as follows:
Attaching hardware for flux valves is NAS1631C9 screws (corrosion resistant steel A286 CRES per AMS5735) and NAS697C06 nuts (corrosion resistant steel, A286 CRES).
Flux valve access plate screws are LS5915-3-2 (corrosion resistant steel, A286 CRES)

57.1.7 FAIRINGS. Formed aluminum alloy fairings are attached by flush screws in the wing-to-fuselage junction area, and in the external tank attachment area. All of the fairings are removable.

57.1.8 EXTERNAL FUEL TANK. The external fuel tank is located approximately at wing station 154. It is fabricated from aluminum alloy skins, bulkheads, and extruded longerons. It is made in two sections, a forward section and an aft section. The forward section slips over the wing, and is attached by three bolts, into wing fittings on the lower surface, and by one bolt into a wing fitting on the upper surface. The aft section extends over the wing trailing edge, and

57.1.8 (Cont.)
is attached to the upper part of the forward section through two fittings and to the lower part of the forward section by a single housed strut. A tail cap structure of molded fiberglas and formed aluminum alloy ribs is attached to the aft portion of the aft section to provide a smooth, faired surface.

57.1.9 LEADING EDGE FLAPS. Two configurations of leading edge flaps are used. Airplanes 5001 thru 5009, 5011 thru 5016, 5018 thru 5021, 5023, 5025, 5029 and 5033 are manufactured using honeycomb construction. Airplanes 5010, 5017, 5022, 5024, 5026, 5027, 5028, 5030, 5031, 5032, 5034 and up use sheet metal construction.

57.1.9.1 LEADING EDGE FLAP - SHEET METAL. This flap is an aluminum alloy structure consisting of skin and formed ribs. It is assembled by flush

57.1.9.1 (Cont).

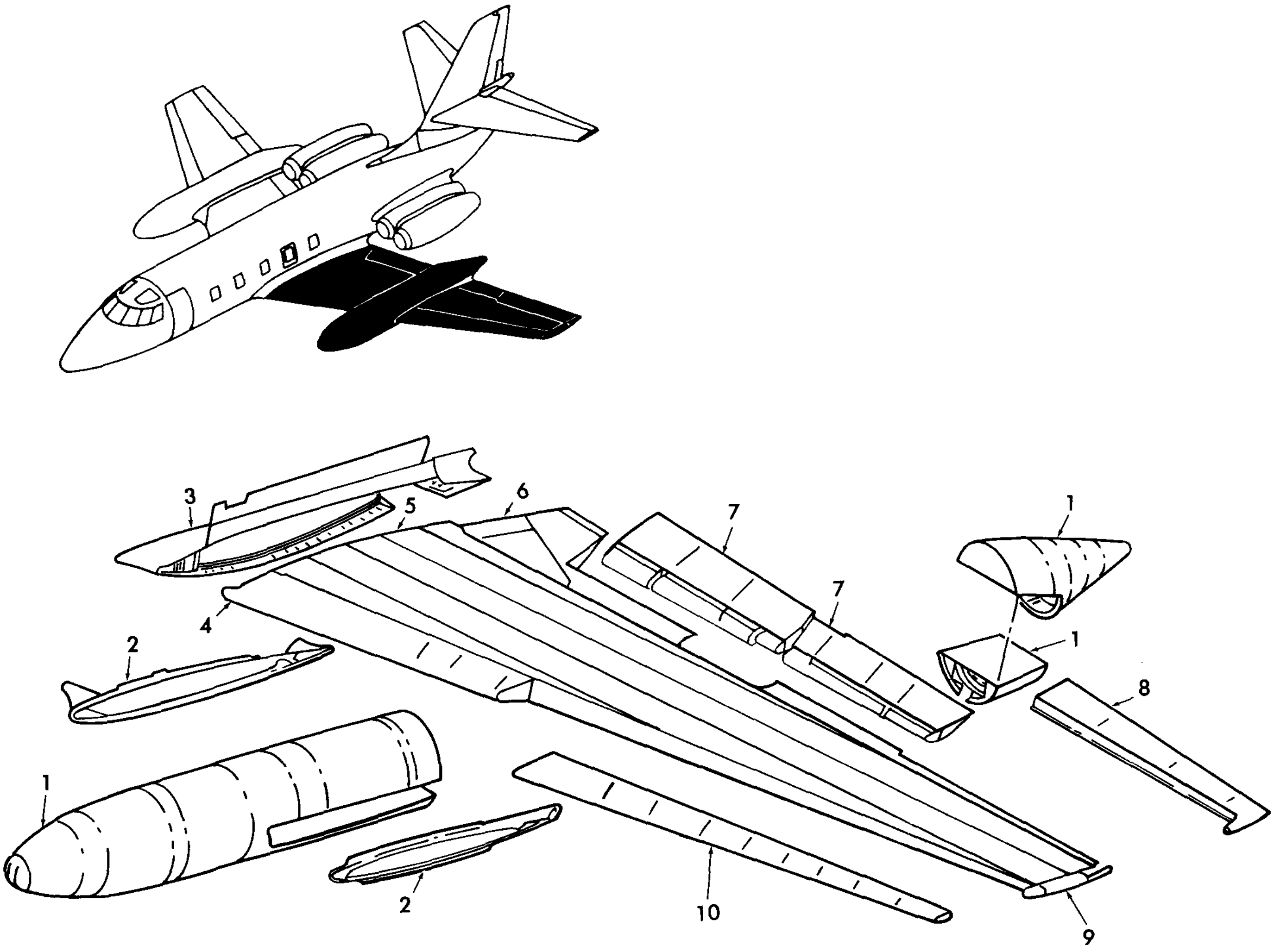
rivets and is hinged to the wing by a piano-type hinge which is attached to the lower cap of the front beam. The leading edge flap consists of two separate structural sections on each wing, one inboard of the other.

57.1.9.2 LEADING EDGE FLAP - HONEYCOMB. This flap consists of aluminum alloy formed ribs, nose skins, two span-wise beams, and bonded aluminum alloy honeycomb panel assemblies. It is assembled with flush fasteners and consists of two separate structural sections on each wing. The flaps are hinged to the wing by a piano-type hinge which is attached to the lower cap of the front beam.

57.1.10 DETAIL WING COMPONENTS MATERIAL AND IDENTIFICATION. For material and identification of detail wing components, refer to figures 57-1 through 57-11 at the end of the section. For typical repairs refer to Chapter 51.

WARNING

See the precautions in Section 51.9 before working on or near the uranium counter weights.



INDEX	COMPONENT	FIG. NO.
1	EXTERNAL WING TANK	57-9
2	WING-TO-TANK FILLETS	57-7
3	WING-TO-FUSELAGE FILLETS	57-7
4	WING LEADING EDGE	57-2, 57-3
5	WING BOX SECTION	57-2, 57-3
6	WING TRAILING EDGE	57-2, 57-3
7	TRAILING EDGE FLAP	57-8
8	AILERON	57-6
9	WING TIP	57-2, 57-3
10	LEADING EDGE FLAP	57-10, 57-11

JS-2-2-5701

Figure 57-1. Wing Assembly Breakdown

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	RIB ASSEMBLY		
	ANGLE		7075-T6 EXTRUSION
	ANGLE		2024-T4 EXTRUSION
	RIB	0.050	2024-T4 CLAD
	WEB	0.050	7075-T6 CLAD
	DOUBLER	0.050	7075-T6 CLAD
	PLATE	0.063	7075-T6 CLAD
2	RIB	0.040	7075-T6 CLAD
3	BRACE	0.063	7075-T6 CLAD
4	ROOT RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	CHEMICALLY MILLED WEB	0.190	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	FITTING		7075-T6 FORGING
	DOUBLER	0.125	7075-T6 CLAD
	DOUBLER	0.190	7075-T6 CLAD
5	DOOR ASSEMBLY		
	DOOR		X7178-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
6	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
7	DOOR ASSEMBLY		
	DOOR		X7178-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	DOOR	0.160	7075-T6 CLAD
8	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER		2024-T81 EXTRUSION
	STIFFENER		2024-T4 EXTRUSION
9	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	ANGLE		2024-T42 EXTRUSION
	ANGLE	0.040	7075-T6 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	DOUBLER	0.040	7075-T6 CLAD
10	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	CHANNEL	0.040	7075-T6 CLAD
	ANGLE		2024-T4 EXTRUSION
	ANGLE	0.040	7075-T6 CLAD
11	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	CHANNEL	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
12	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
13	RIB ASSEMBLY		
	WEB	0.100	7075-T6 CLAD
	GUSSET	0.050	7075-T6 CLAD
14	RIB ASSEMBLY		
	CHANNEL	0.050	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
15	SPAR		7075-T6 FORGING

JS-2-2-5710-1

Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 1 of 6)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
16	TEE		7075-T6 EXTRUSION
17	FORMER ASSEMBLY		
	CAP		2024-T4 EXTRUSION
	FORMER	0.040	7075-T6 CLAD
	STIFFENER	0.050	7075-T6 CLAD
18	FORMER ASSEMBLY		
	FORMER	0.040	7075-T6 CLAD
	STIFFENER	0.032	7075-T6 CLAD
	STIFFENER	0.050	7075-T6 CLAD
19	FORMER ASSEMBLY		
	FORMER	0.040	7075-T6 CLAD
	STIFFENER	0.032	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
20	CHANNEL	0.032	7075-T6 CLAD
21	FORMER ASSEMBLY		
	FORMER	0.050	7075-T6 CLAD
	FORMER	0.032	7075-T6 CLAD
	STIFFENER	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
22	FORMER	0.040	7075-T6 CLAD
23	RIB ASSEMBLY		
	FORMER	0.040	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	ANGLE	0.040	7075-T6 CLAD
24	RIB ASSEMBLY		
	FORMER	0.040	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD
25	CHANNEL	0.040	7075-T6 CLAD
26	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.050	7075-T6 CLAD
	DOUBLER	0.080	7075-T6 CLAD
27	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
28	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	DOUBLER	0.040	7075-T6 CLAD
	FORMER	0.040	7075-T6 CLAD
29	FORMER	0.032	7075-T6 CLAD
30	RIB ASSEMBLY		
	FORMER	0.040	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD
	DOUBLER	0.050	2024-T4 CLAD
	STIFFENER		2024-T4 EXTRUSION
31	RIB ASSEMBLY		
	FORMER	0.032	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD

JS-2-2-5710-2

Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 2 of 6)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
32	RIB ASSEMBLY		
	FORMER	0.032	7075-T6 CLAD
	WEB	0.032	7075-T6 CLAD
	STIFFENER	0.040	2024-T3 CLAD
33	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.040	7075-T6 CLAD
	CHANNEL	0.040	7075-T6 CLAD
34	REAR BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.190	7075-T6 CLAD
	WEB	0.071	7075-T6 CLAD
	WEB	0.063	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.063	7075-T6 CLAD
	ANGLE	0.090	7075-T6 CLAD
	FITTING		4340 STEEL, ANNEALED
	FITTING		7075-T6 FORGING
	FITTING		7075-T6 EXTRUSION
	FITTING		A356-T61 CASTING
	DOUBLER	0.080	7075-T6 CLAD
	ZEE		7075-T6 EXTRUSION
	TEE		7075-T6 EXTRUSION
35	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	CHANNEL	0.040	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	DOUBLER	0.090	7075-T6 CLAD
36	RIB ASSEMBLY		
	WEB	0.032	7075-T6 CLAD
	FORMER	0.025	7075-T6 CLAD
	ANGLE	0.040	2024-T3 CLAD
37	RIB ASSEMBLY		
	WEB	0.032	7075-T6 CLAD
	FORMER	0.025	7075-T6 CLAD
	DOUBLER	0.050	2024-T4 CLAD
	STIFFENER		2024-T4 EXTRUSION
38	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
39	RIB ASSEMBLY		
	WEB	0.032	2024-T4 CLAD
	FORMER	0.025	7075-T6 CLAD
	ANGLE	0.050	2024-T4 CLAD
40	RIB ASSEMBLY		
	WEB	0.050	2024-T4 CLAD
	ANGLE	0.032	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
	ANGLE	0.063	2024-T4 CLAD
	DOUBLER	0.063	2024-T4 CLAD
41	STIFFENER	0.032	2024-T4 CLAD
42	RIB ASSEMBLY		
	RIB	0.050	7075-T6 CLAD
	RIB	0.032	7075-T6 CLAD
	ANGLE	0.040	2024-T4 CLAD

JS-2-2-5710-3

Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 3 of 6)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
43	HINGE		7075-T6 EXTRUSION
44	RIB ASSEMBLY		
	RIB	0.032	7075-T6 CLAD
	ANGLE	0.032	7072-T6 CLAD
	ANGLE	0.040	2024-T4 CLAD
45	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.050	7075-T6 CLAD
	FORMER		7075-T6 EXTRUSION
46	SUPPORT ASSEMBLY		
	FITTING		2014-T6 FORGING
	PANEL	0.050	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
47	FITTING		2014-T6 FORGING
48	HINGE		4130 STEEL
49	RIB ASSEMBLY		
	RIB	0.032	7075-T6 CLAD
	BRACE	0.032	2024-T4 CLAD
	TEE		7075-T6 EXTRUSION
50	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER	0.032	7075-T6 CLAD
51	RIB ASSEMBLY		
	RIB	0.032	7075-T6 CLAD
	ANGLE	0.032	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
52	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.050	2024-T4 CLAD
	STIFFENER		2024-T4 EXTRUSION
	GUSSET	0.125	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
53	RIB ASSEMBLY		
	RIB	0.072	2024-T4 CLAD
	CHANNEL	0.050	2024-T3 CLAD
	ANGLE		7075-T6 EXTRUSION
54	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.050	2024-T3 CLAD
	STIFFENER		2024-T4 EXTRUSION
	DOUBLER	0.063	2024-T3 CLAD
	STIFFENER	0.040	2024-T3
55	RIB ASSEMBLY		
	RIB	0.050	7075-T6 CLAD
	DOUBLER	0.040	7075-T6 CLAD
	ANGLE		2024-T4 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	STRAP	0.032	7075-T6 CLAD
56	RIB ASSEMBLY		
	RIB	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
57	ANGLE		7075-T6 EXTRUSION
58	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	CHEMICALLY MILLED WEB		7075-T6 BARE
	STIFFENER		7075-T6 EXTRUSION

JS-2-2-5710-4

Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 4 of 6)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
59	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.071	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
60	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	CHEMICALLY MILLED WEB	0.125	7075-T6 BARE
	STIFFENER		7075-T6 EXTRUSION
61	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
62	CENTER BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.071	7075-T6 CLAD
	WEB	0.050	7075-T6 CLAD
	WEB	0.040	7075-T6 CLAD
	WEB	0.032	7075-T6 CLAD
	ANGLE		2024-T4 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	STIFFENER		7075-T6 EXTRUSION
	DOOR	0.063	7075-T6 CLAD
63	RIB ASSEMBLY		
	ANGLE		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	TEE		7075-T6 EXTRUSION
64	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER	0.040	7075-T6 CLAD
65	RIB ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER	0.032	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
66	RIB ASSEMBLY		
	TEE		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER	0.040	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
67	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.050	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
68	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.025	7075-T6 CLAD
	STIFFENER	0.032	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
69	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER	0.040	7075-T6 CLAD
70	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	TEE		7075-T6 EXTRUSION

JS-2-2-5710-5

Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 5 of 6)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
71	RIB ASSEMBLY		
	CHANNEL	0.063	2024-T4 CLAD
	ANGLE		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	TEE		7075-T6 EXTRUSION
72	FRONT BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	CHEMICALLY MILLED WEB	0.100	7075-T6 BARE
	CHEMICALLY MILLED WEB	0.050	7075-T6 BARE
	ZEE		7075-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.063	2024-T3 CLAD
	FITTING		7075-T6 EXTRUSION
	FITTING		7075-T6 ROLLED BAR
	STIFFENER		7075-T6 EXTRUSION
	ANGLE	0.040	2024-T4 CLAD
	BRACKET (LEFT SIDE ONLY)	0.090	2024-T4 CLAD
	COMPRESSION STOP		7075-T6 FORGING
	BRACKET		7075-T6 EXTRUSION
	FITTING		8630 STEEL BAR, HEAT-TREAT 160,000 - 180,000 PSI
	COMPRESSION STOP		8630 STEEL BAR, HEAT-TREAT 160,000 - 180,000 PSI
	HINGE		7075-T6 EXTRUSION
73	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.063	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
74	RIB ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.032	7075-T6 CLAD
	STIFFENER		2024-T81 EXTRUSION
75	RIB ASSEMBLY (RIGHT SIDE ONLY)		
	WEB	0.032	7075-T6 CLAD
	DOUBLER	0.050	7075-T6 CLAD
	ANGLE	0.040	2024-T3 CLAD
76	CHANNEL (RIGHT SIDE ONLY)	0.032	7075-T6 CLAD

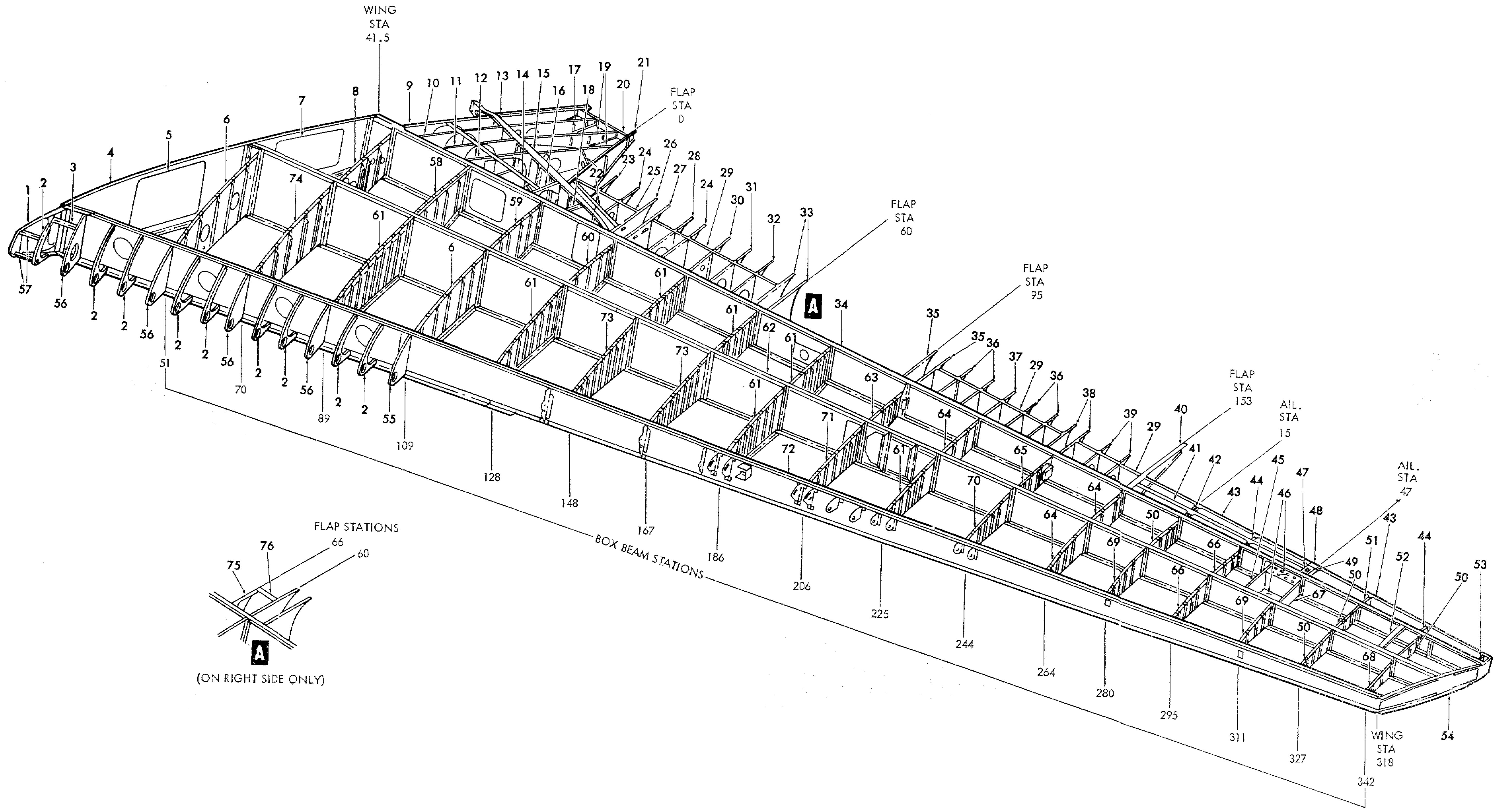
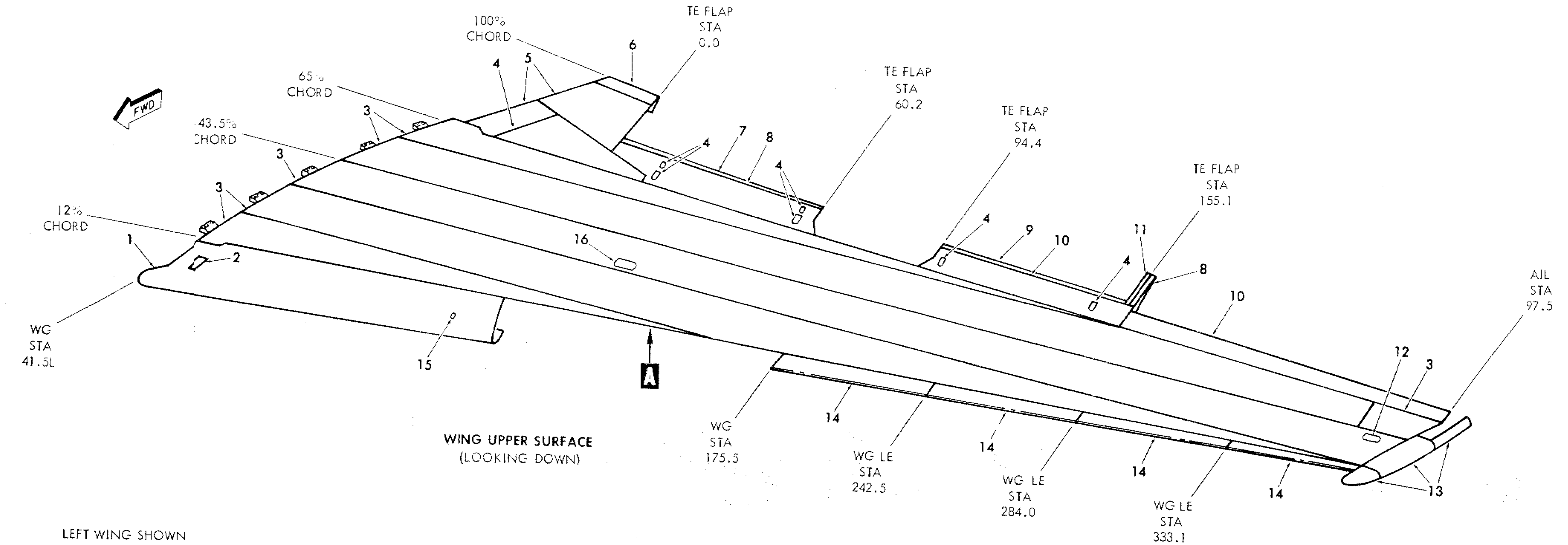
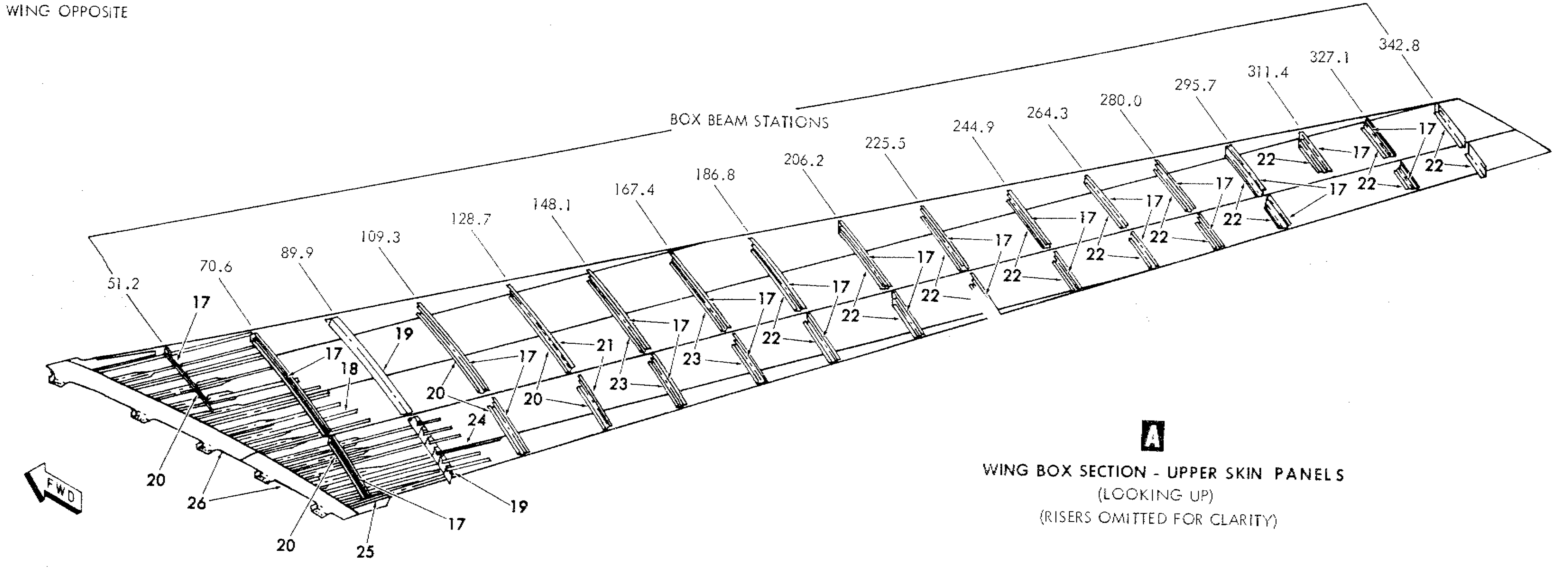


Figure 57-2. Wing Leading Edge, Trailing Edge, Tip, and Box Section Structure (Sheet 6 of 6)



WING UPPER SURFACE
(LOOKING DOWN)

LEFT WING SHOWN
RIGHT WING OPPOSITE



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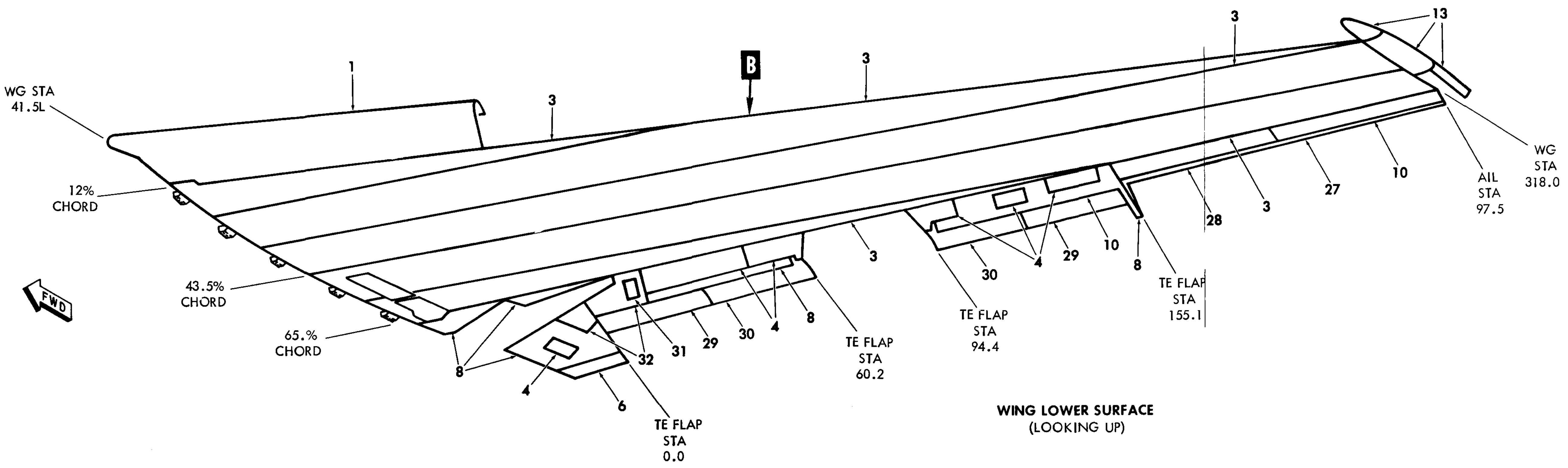
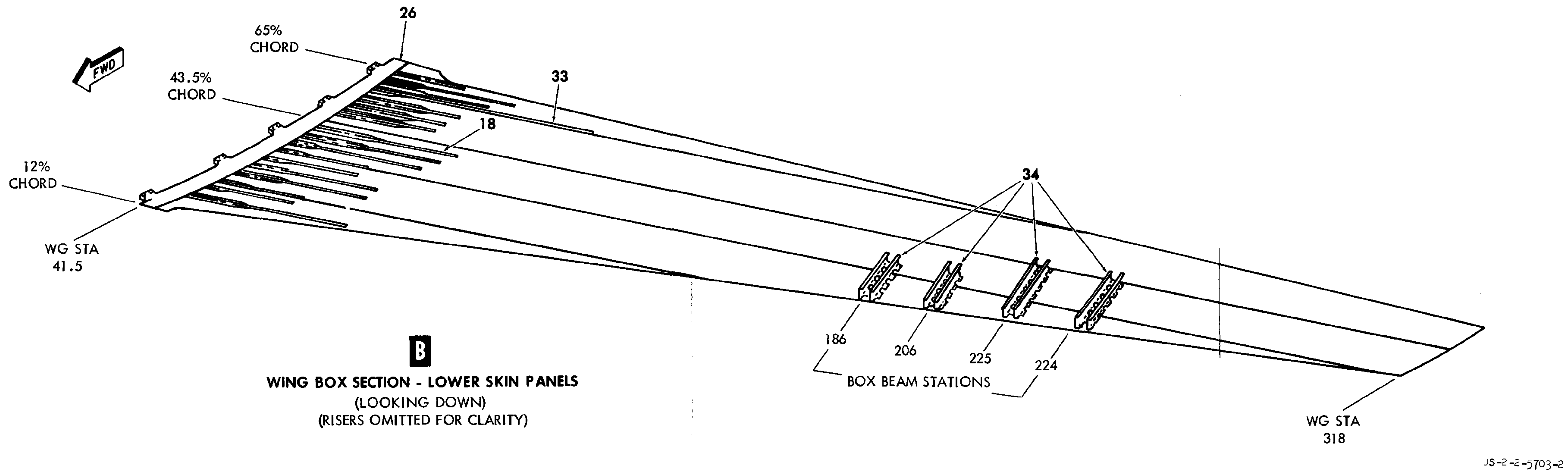
WING BOX SECTION - UPPER SKIN PANELS
(LOOKING UP)
(RISERS OMITTED FOR CLARITY)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	LEADING EDGE SKIN	0.050	7075-T6 CLAD
2	DOOR	0.100	7075-T6 CLAD
3	MACHINED AND CHEMICALLY MILLED INTEGRALLY STIFFENED SKIN PANEL (RISERS OMITTED FOR CLARITY). AIRPLANES 5001 THRU 5060 AND 5062 THRU 5066)		7075-T6 EXTRUSION
	(AIRPLANES 5061 AND 5067 THRU 5162)		7075-T6511 EXTRUSION
	(SPARE REPLACEMENT PANELS)		7075-T76511 EXTRUSION
4	DOOR	0.040	2024-T3 CLAD
5	SKIN	0.040	2024-T3 CLAD
6	SKIN ASSEMBLY		
	OUTER SKIN	0.016	2024-T3 CLAD
	INNER SKIN	0.012	2024-T4 CLAD
7	TE EXTENSION ASSEMBLY		
	EXTENSION	0.016	302 CRES ANNEALED
	STRIP	0.020	TEFLON
	STRIP		2024-T4 EXTRUSION
8	SKIN	0.032	2024-T3 CLAD
9	TE EXTENSION ASSEMBLY		
	EXTENSION	0.040	2024-T3 CLAD
	STRIP	0.020	TEFLON
	STRIP		2024-T4 EXTRUSION
10	SKIN	0.025	2024-T3 CLAD
11	CLOSURE STRIP ASSEMBLY		
	CLOSURE STRIP	0.012	17-7PH STEEL-TH1050
	RUB STRIP	0.020	TEFLON
12	DOOR	0.050	7075-T6 CLAD
13	WING TIP		
	FAIRINGS AND COVERS	0.050	FIBERGLASS
	STRIP	0.050	2024-T4 CLAD
14	SHROUD PLATE	0.050	301 CRES - 1/4 HARD
15	MACHINED MOUNTING PLATE	0.150	7075-T6
16	DOOR	0.071	7075-T6 CLAD
17	FORMER	0.025	7075-T6 CLAD
18	ANGLE		7075-T6 EXTRUSION
19	TEE		7075-T6 EXTRUSION
20	FORMER	0.071	7075-T6 CLAD
21	CHANNEL	0.032	7075-T6 CLAD
22	FORMER	0.040	7075-T6 CLAD
23	FORMER	0.050	7075-T6 CLAD
24	ANGLE	0.050	7075-T6 BARE
25	STRINGER		7075-T6 EXTRUSION
26	FITTING		7075-T6 EXTRUSION
27	TE STRIP		2014-T6 EXTRUSION
28	DOOR	0.032	2024-T4 CLAD
29	SHROUD	0.025	2024-T4 CLAD
30	SHROUD	0.025	2024-T3 CLAD
31	COVER	0.040	2024-T3 CLAD
32	DOOR	0.032	2024-T3 CLAD
33	ANGLE	0.050	7075-T6 CLAD
34	FORMER	0.063	7075-T6 CLAD

Figure 57-3. Wing Leading Edge, Trailing Edge, Tip, and Box Section Skins (Sheet 1 of 2)



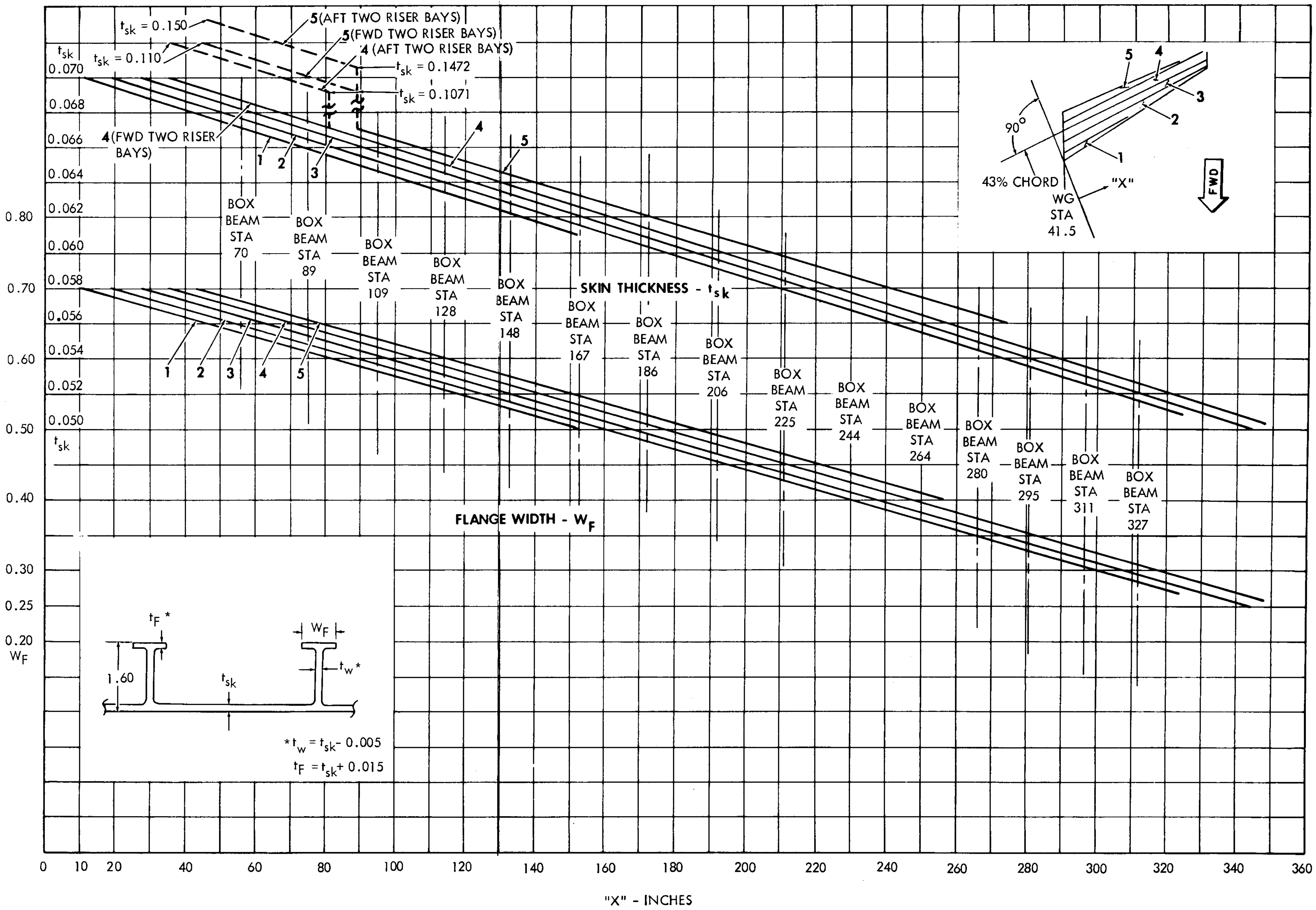
WING LOWER SURFACE
(LOOKING UP)



B

WING BOX SECTION - LOWER SKIN PANELS
(LOOKING DOWN)
(RISERS OMITTED FOR CLARITY)

Figure 57-3. Wing Leading Edge, Trailing Edge, Tip, and Box Section Skins (Sheet 2 of 2)



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NOTE

1. SPARE REPLACEMENT PANELS ARE PROVIDED WITH t_{sk} 0.005 THICKER.

Figure 57-4. Wing Integrally Stiffened Skin Panels Lower Surface Geometry

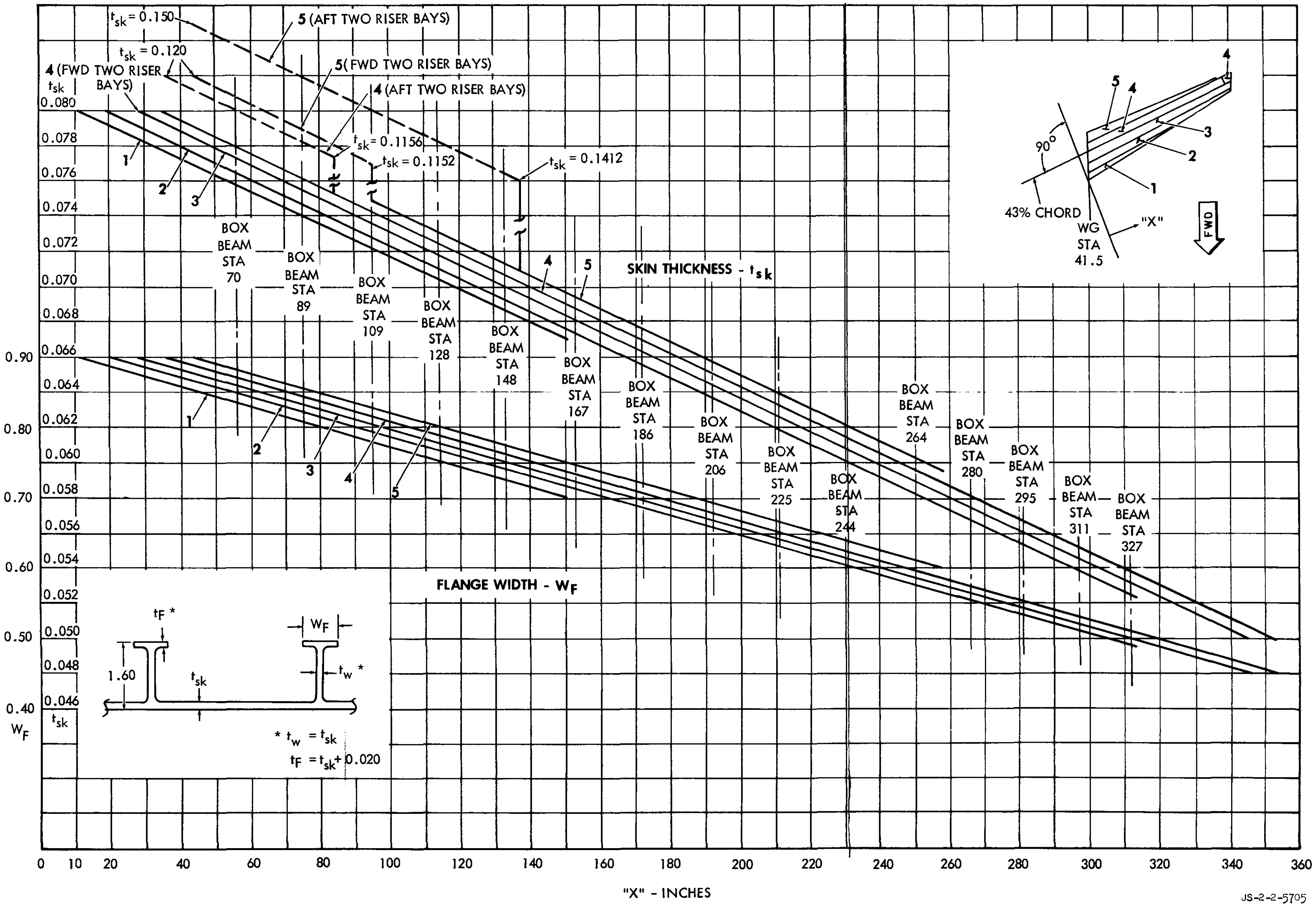
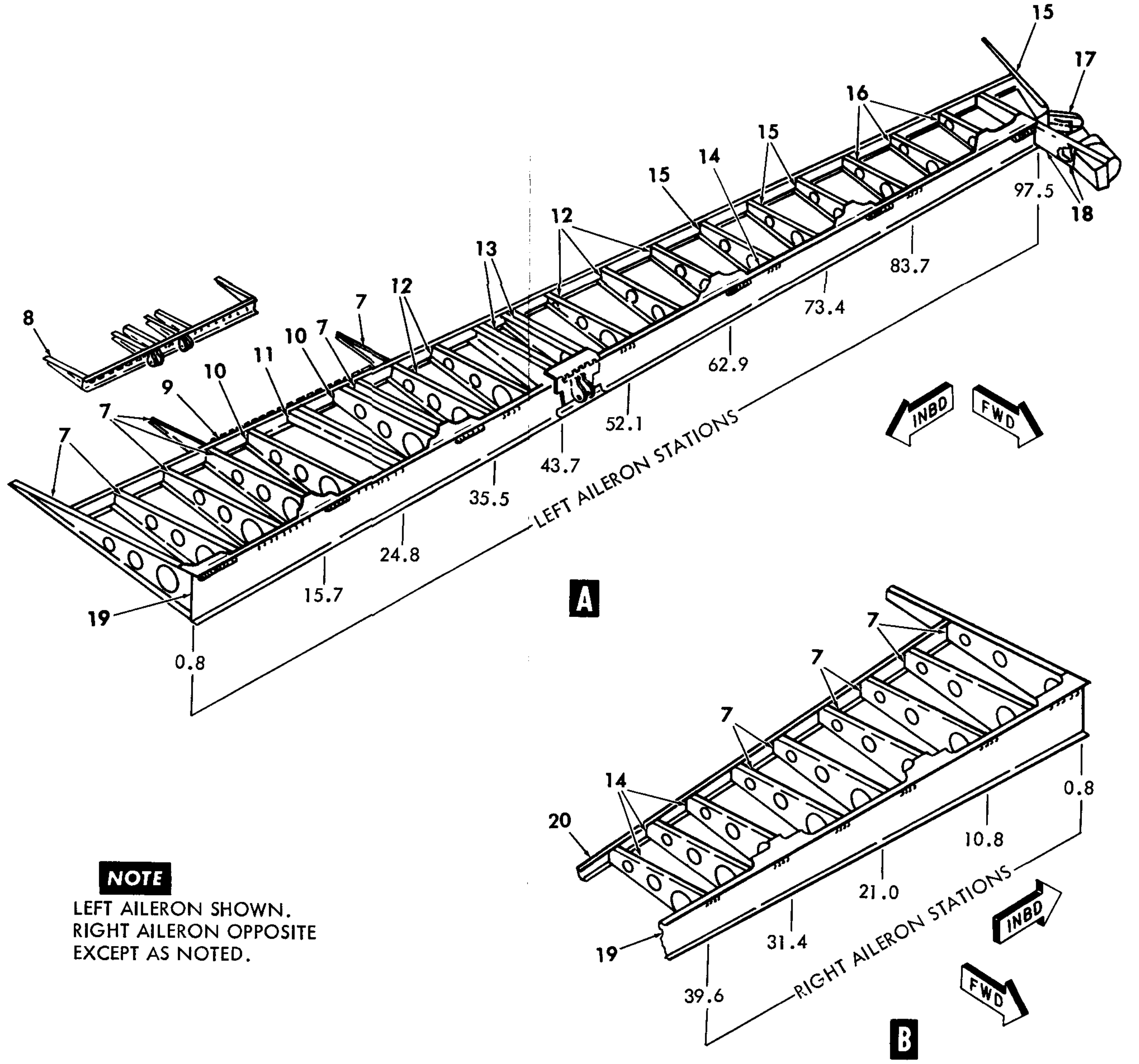
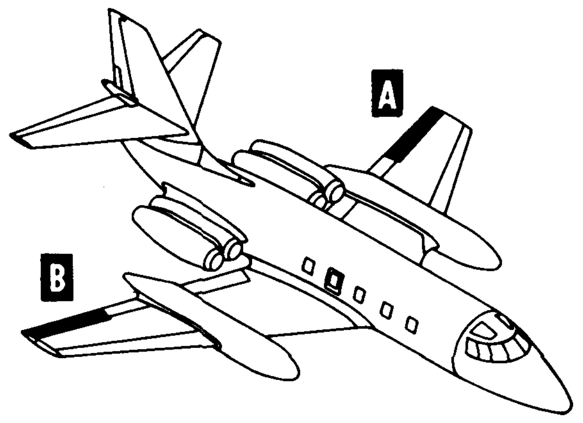


Figure 57-5. Wing Integrally Stiffened Skin Panels Upper Surface Geometry



NOTE

LEFT AILERON SHOWN.
RIGHT AILERON OPPOSITE
EXCEPT AS NOTED.



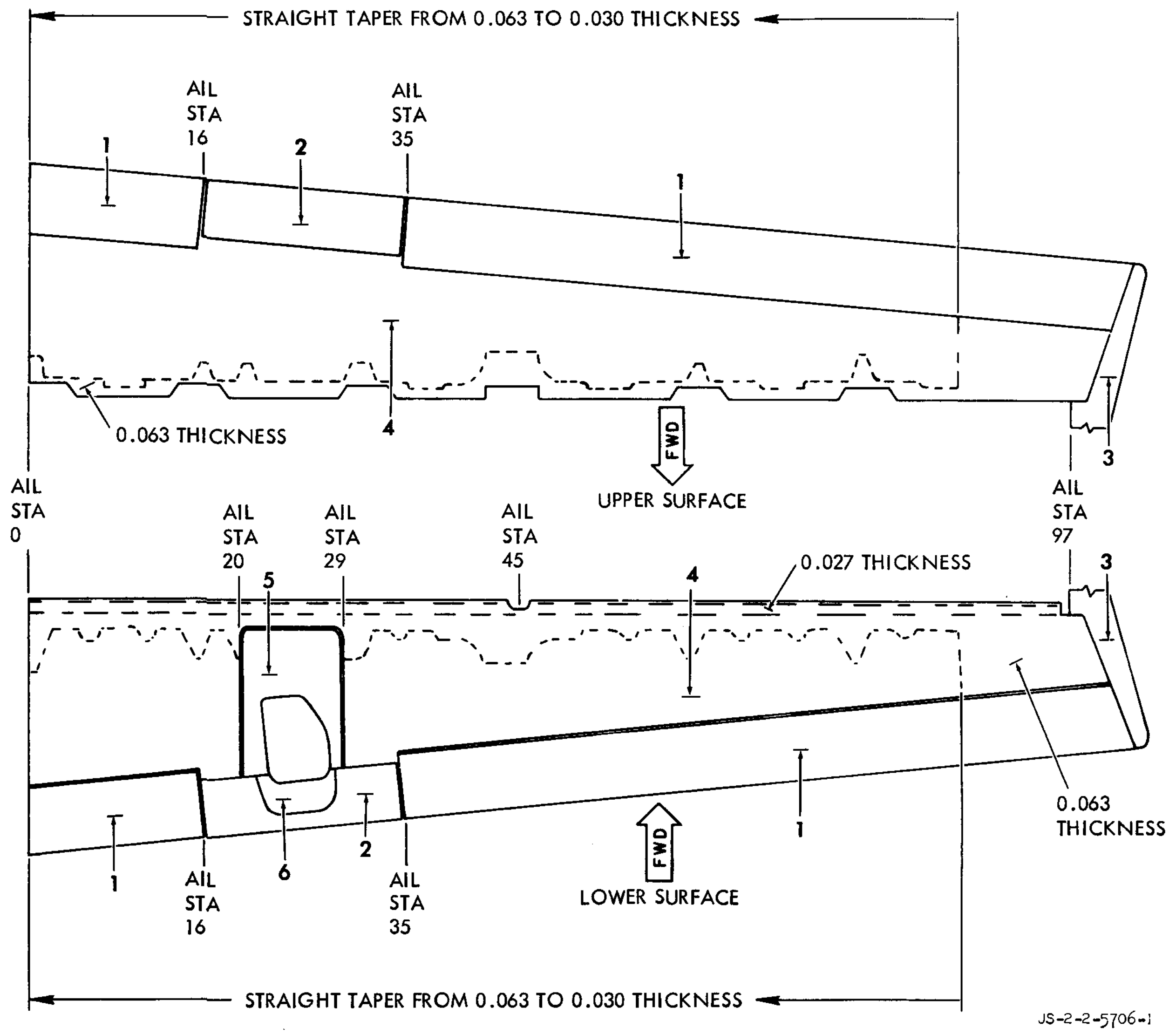


Figure 57-6. Aileron Skin and Structure (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	SKIN ASSEMBLY		
	INNER SKIN	0.012	2024-T4 CLAD
	OUTER SKIN	0.012	2024-T4 CLAD
	SPLICE PLATE	0.012	2024-T4 CLAD
2	TAB SKIN (LEFT AILERON ONLY)	0.025	2024-T4 CLAD
3	COVER ASSEMBLY		
	COVER	0.063	6061-T6 CLAD
	END PLATE	0.063	6061-T6 CLAD
4	CHEMICALLY MILLED SKIN	0.063	7075-T6 CLAD
5	ACCESS COVER ASSY (LEFT AIL. ONLY)		
	COVER	0.050	2024-T4 CLAD
	STRIP	0.012	2024-T4 CLAD
	STRIP	0.032	2024-T4 CLAD
6	HOOD (LEFT AILERON ONLY)	0.025	2024-T4 CLAD
7	RIB	0.025	2024-T4 CLAD
8	TAB (LEFT AILERON ONLY)		
	RIB	0.025	2024-T4 CLAD
	RIB	0.032	2024-T4 CLAD
	DOUBLER	0.025	2024-T4 CLAD
	SPAR	0.032	2024-T4 CLAD
	FORMER	0.032	2024-T4 CLAD
	FITTING		7075-T6 EXTRUSION
	MS20001-Y9 HINGE		2024-T4 EXTRUSION
9	SPAR ASSEMBLY (LEFT AIL. ONLY)		
	SPAR	0.040	7075-T6 CLAD
	ANGLE	0.032	2024-T4 CLAD
	BRACKET	0.032	2024-T4 CLAD
	MS20001-Y9 HINGE		2024-T4 EXTRUSION
10	RIB ASSEMBLY		
	RIB	0.032	2024-T4 CLAD
	ANGLE	0.032	2024-T4 CLAD
11	STIFFENER	0.025	2024-T4 CLAD
12	RIB	0.032	2024-T4 CLAD
13	HORN SUPPORT ASSEMBLY		
	ANGLE	0.063	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
	HORN		2014-T6 FORGING
	HINGE		4130 STEEL PLATE
14	SPAR ASSEMBLY		
	SPAR	0.050	7075-T6 CLAD
	HINGE FITTING		7075-T6 FORGING
	HINGE SUPPORT		2014-T6 FORGING
	HINGE		7075-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	SUPPORT BRACKET	0.032	2024-T4 CLAD
15	RIB	0.040	2024-T4 CLAD
16	RIB	0.050	2024-T4 CLAD
17	FORMER	0.050	2024-T4 CLAD
18	COUNTERBALANCE WEIGHT ASSY		
	RIB	0.063	2024-T4 CLAD
	FITTING		7075-T6 FORGING
	ANGLE		7075-T6 EXTRUSION
	WEIGHT		DEPLETED URANIUM
	WEIGHT		BRASS
	WEIGHT		SINTERED TUNGSTEN
19	SPAR ASSEMBLY		
	SPAR	0.040	2024-T4 CLAD
	HINGE SUPPORT		2014-T6 FORGING
	HINGE		7075-T6 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	SUPPORT BRACKET	0.032	2024-T4 CLAD
20	SPAR (RIGHT AILERON ONLY)	0.040	7075-T6 CLAD

JS-2-2-5706-2

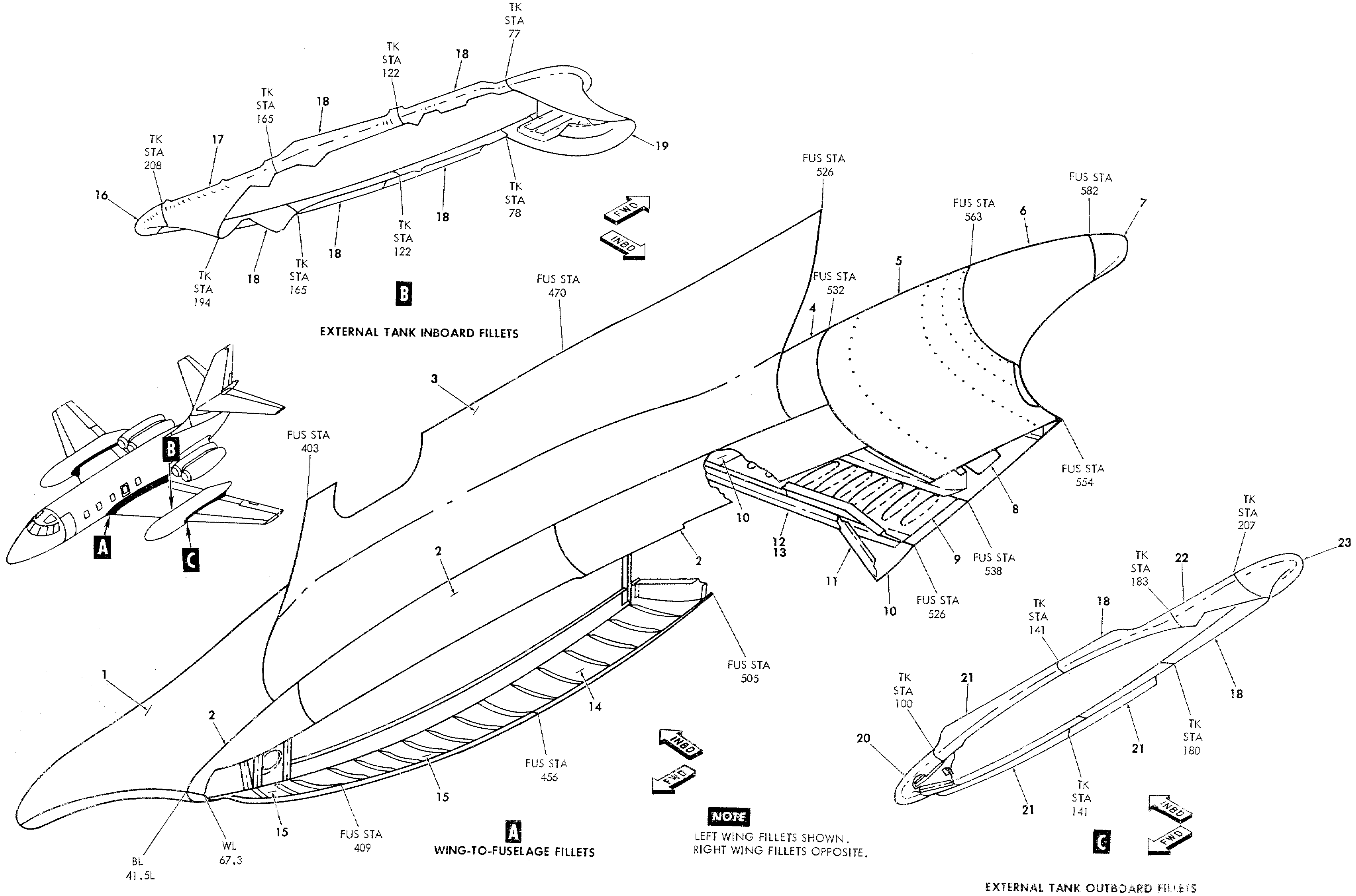
Figure 57-6. Aileron Skin and Structure (Sheet 2 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	FILLET ASSEMBLY		
	SKIN	0.040	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
	LONGERON	0.050	7075-T6 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	BULKHEAD	0.040	7075-T6 CLAD
	BULKHEAD	0.050	7075-T6 CLAD
	ANGLE	0.040	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
	SUPPORT	0.050	7075-T6 CLAD
2	FILLET ASSEMBLY		
	SKIN	0.040	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
3	FILLET ASSEMBLY		
	SKIN	0.040	2024-T4 CLAD
	STRAP	0.190	2024-T4 CLAD
	CHANNEL	0.050	7075-T6 CLAD
	FORMER	0.050	2024-T4 CLAD
	STIFFENER		2024-T42 EXTRUSION
	SPLICE	0.050	2024-T3 CLAD
4	ACCESS PANEL	0.040	2024-T3 CLAD
5	FILLET ASSEMBLY		
	SKIN	0.032	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
	RIB	0.050	7075-T6 CLAD
	PLATE	0.190	7075-T6 CLAD
	RIB	0.040	7075-T6 CLAD
	RIB	0.032	7075-T6 CLAD
	SPLICE	0.032	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
6	FILLET ASSEMBLY		
	FILLET	0.032	2024-T4 CLAD
	RIB	0.040	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
	DOOR	0.032	2024-T4 CLAD
7	FILLET	0.032	2024-T4 CLAD
8	ACCESS DOOR ASSEMBLY		
	INNER SKIN	0.032	2024-T4 CLAD
	OUTER SKIN	0.040	2024-T4 CLAD
9	DOOR ASSEMBLY		
	INNER SKIN	0.032	2024-T4 CLAD
	OUTER SKIN	0.032	2024-T4 CLAD
10	SKIN	0.040	2024-T4 CLAD
11	FORMER	0.063	7075-T6 CLAD
12	TEE		7075-T6 EXTRUSION
13	BEAM	0.080	7075-T6 CLAD
14	FILLET ASSEMBLY		
	SKIN	0.040	2024-T4 CLAD
	FORMER	0.040	2024-T4 CLAD
	FORMER	0.063	2024-T4 CLAD
15	FILLET ASSEMBLY		
	SKIN	0.040	2024-T4 CLAD
	FORMER	0.040	2024-T4 CLAD
	SPLICE	0.040	2024-T4 CLAD
16	TAIL FILLET	0.071	LAMINATED GLASS FABRIC

JS-2-2-5707-1

Figure 57-7. Wing Fillets Skin and Structure (Sheet 1 of 2)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
17	FILLET ASSEMBLY		
	SKIN	0.071	6061-T6
	FORMER	0.032	2024-T4 CLAD
	RUBBING STRIP	0.032	TEFLON
	RUBBING STRIP	0.020	TEFLON
	PANEL	0.032	2024-T4 CLAD
18	FILLET SKIN	0.071	6061-T6
19	FAIRING ASSEMBLY		
	RIB		LAMINATED GLASS CLOTH FABRIC
	FAIRING		LAMINATED GLASS CLOTH FABRIC
20	FILLET ASSEMBLY		
	SKIN	0.071	6061-T6
	RIB	0.040	2024-T4 CLAD
	ANGLE		2024-T4 EXTRUSION
	ANGLE	0.040	2024-T3 CLAD
	STRIP	0.040	2024-T4
21	FILLET ASSEMBLY		
	SKIN	0.071	6061-T6
	STIFFENER	0.025	2024-T4 CLAD
22	FILLET ASSEMBLY		
	SKIN	0.071	6061-T6
	STRAP	0.040	2024-T4 CLAD
	FORMER	0.025	2024-T4 CLAD
	FORMER	0.032	2024-T4 CLAD
	RUBBING STRIP	0.032	TEFLON
	RUBBING STRIP	0.020	TEFLON
	PANEL	0.032	6061-T6
23	FILLET ASSEMBLY		
	FILLET	0.071	LAMINATED GLASS FABRIC
	FORMER	0.032	2024-T4 CLAD
	BLOCK		CELLULAR CELLULOSE ACETATE



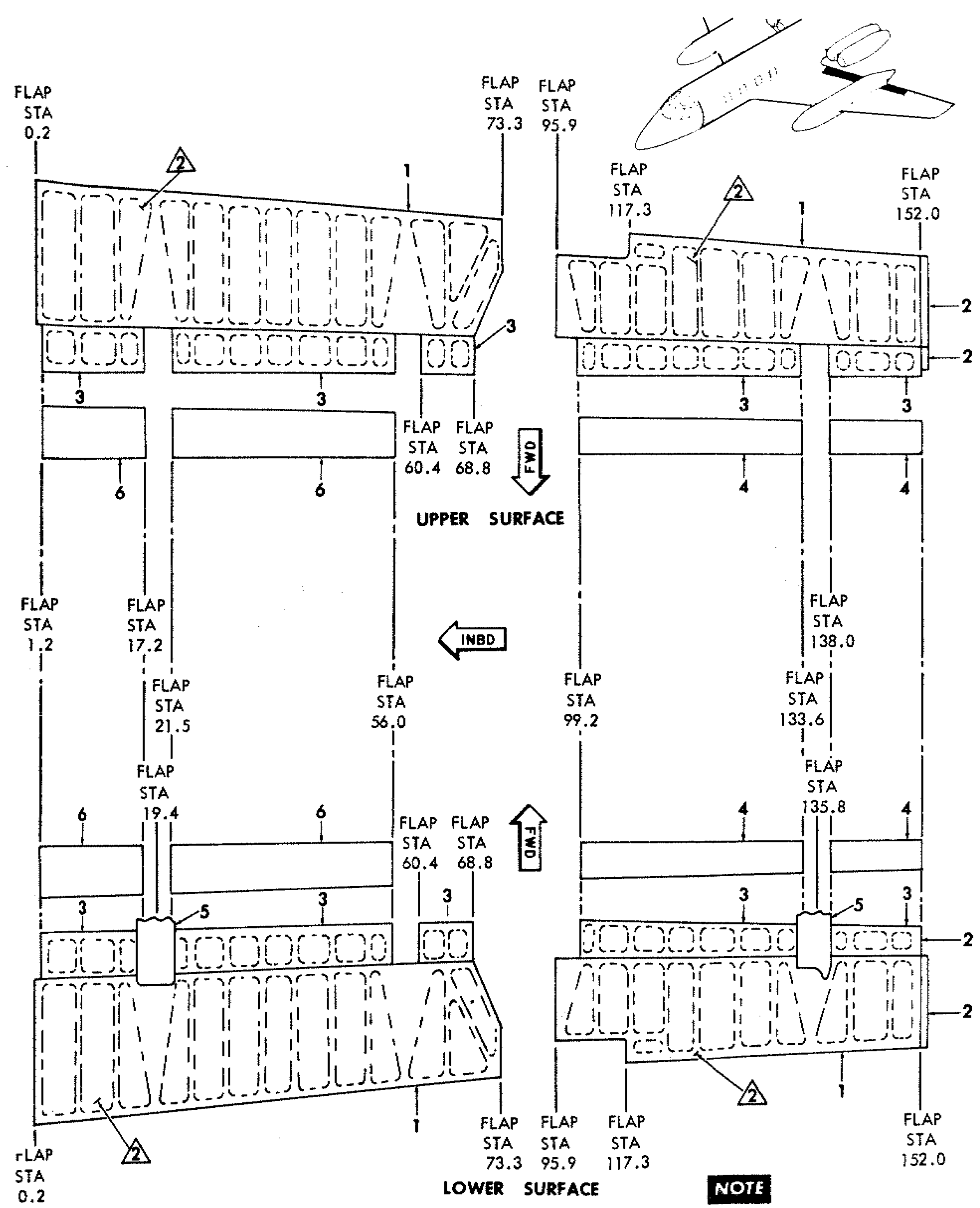
B EXTERNAL TANK INBOARD FILLETS

A WING-TO-FUSELAGE FILLETS

NOTE: LEFT WING FILLETS SHOWN. RIGHT WING FILLETS OPPOSITE.

C EXTERNAL TANK OUTBOARD FILLETS

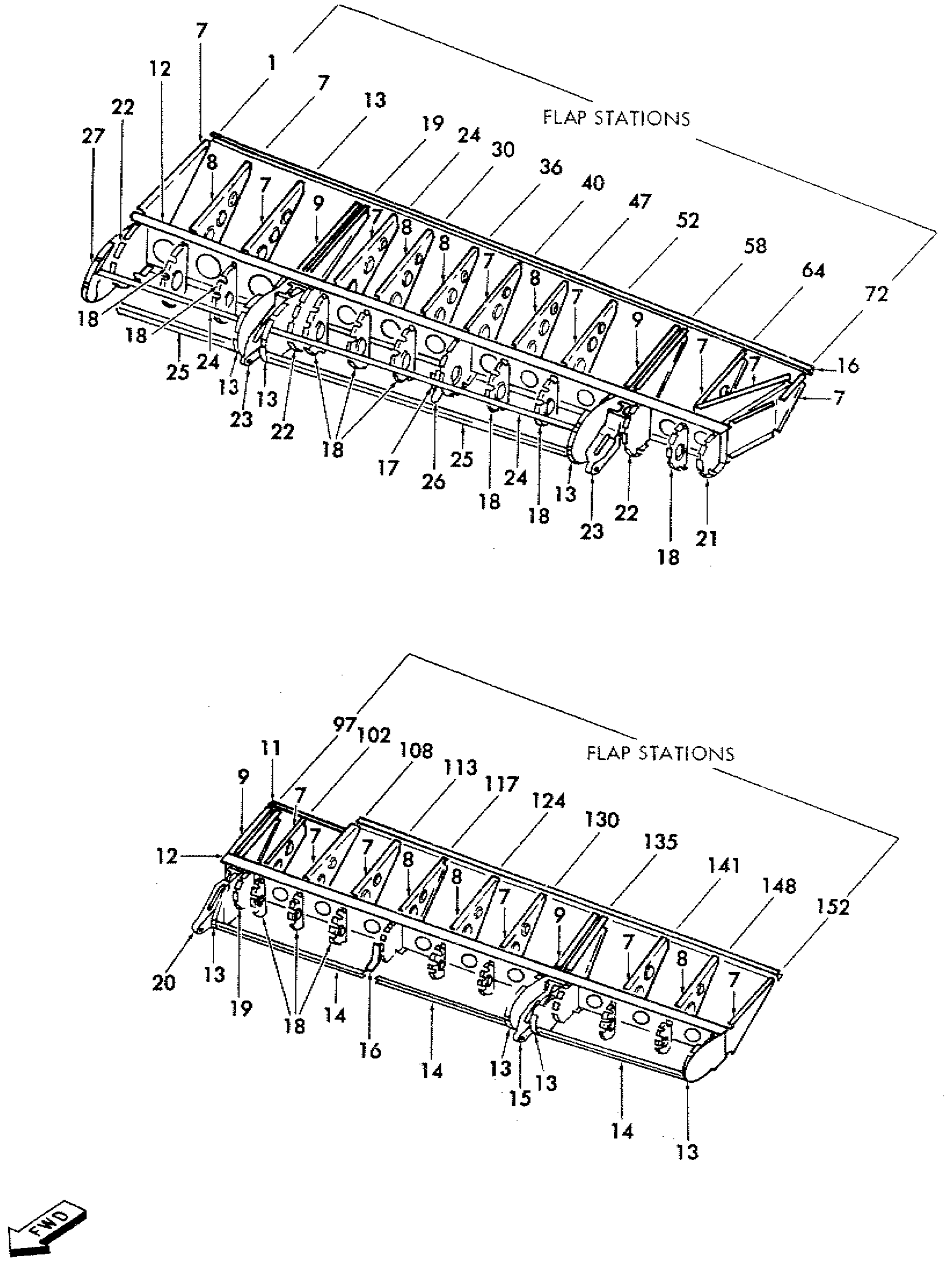
Figure 57-7. Wing Fillets Skin and Structure (Sheet 2 of 2)



NOTE

1. LEFT WING TRAILING EDGE FLAP SHOWN, RIGHT WING TRAILING EDGE FLAPS OPPOSITE.

△2 0.025 THICKNESS TYPICAL IN MILLED OUT AREAS.



INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	CHEMICALLY MILLED SKIN	0.050	2024-T3 CLAD
2	CLOSURE	0.012	17-7PH STEEL
3	CHEMICALLY MILLED LE SKIN	0.050	2024-T4 CLAD
4	VANE SKIN	0.032	2024-T3 CLAD
5	FAIRING		LAMINATED FABRIC
6	VANE SKIN	0.040	2024-T3 CLAD
7	RIB	0.032	2024-T3 CLAD
8	RIB	0.025	2024-T3 CLAD
9	FITTING SUPPORT ASSEMBLY		
	RIB	0.032	2024-T3 CLAD
	TEE		2024-T4 EXTRUSION
10	(NOT USED)		
11	CHANNEL	0.063	2024-T6 CLAD
12	BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	CHEMICALLY MILLED WEB	0.063	CLAD
13	VANE END PLATE ASSEMBLY		
	END PLATE	0.050	2024-T3 CLAD
	CLOSURE FITTING	0.625	2024-T4 PLATE
14	SPAR	0.040	2024-T4 CLAD
15	FITTING ASSEMBLY		
	SUPPORT	0.040	2024-T4 CLAD
	FITTING		7075-T6 BAR
	FITTING		7075-T6 FORGING
16	VANE SUPPORT ASSEMBLY		
	CENTER PLATE	0.040	2024-T3 CLAD
	FITTING	0.625	2024-T4 PLATE
	TEE		7075-T6 EXTRUSION
17	RIB ASSEMBLY		
	RIB	0.050	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
18	RIB	0.025	2024-T4 CLAD
19	RIB	0.050	2024-T4 CLAD
20	FITTING ASSEMBLY		
	SUPPORT	0.050	2024-T3 CLAD
	FITTING		7075-T6 BAR
	FITTING		7075-T6 FORGING
21	RIB	0.032	2024-T4 CLAD
22	RIB	0.040	2024-T4 CLAD
23	FITTING ASSEMBLY		
	SUPPORT	0.040	2024-T3 CLAD
	ANGLE	0.040	2024-T3 CLAD
	FITTING		7075-T6 BAR
	FITTING		7075-T6 FORGING
24	SPAR	0.040	2020-T3 CLAD
25	SPAR	0.032	2024-T3 CLAD
26	VANE SUPPORT ASSEMBLY		
	SUPPORT	0.050	2024-T4 CLAD
	FITTING		2024-T4 BAR
27	VANE END PLATE ASSEMBLY		
	RIB	0.040	2024-T4 CLAD
	END PLATE	0.050	2024-T3 CLAD
	CLOSURE FITTING	0.625	2024-T4 PLATE

Figure 57-8. Trailing Edge Flap Skin and Structure

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	COVER ASSEMBLY		
	PLEXIGLASS COVER	0.250	ACRYLIC PLASTIC
	FIBER CLOTH COVER		LAMINATED GLASS CLOTH
	RING	0.040	6061-T4
2	SKIN	0.040	2024-T4 CLAD
3	SKIN	0.040	2024-T3 CLAD
4	DOOR	0.040	2024-T3 CLAD
5	SKIN ASSEMBLY		
	INNER SKIN	0.032	2024-T4 CLAD
	SKIN	0.040	2024-T4 CLAD
6	DOOR	0.050	2024-T4 CLAD
7	SKIN	0.025	2024-T4 CLAD
8	SKIN	0.032	2024-T3 CLAD
9	FAIRING		LAMINATED GLASS CLOTH
10	BULKHEAD ASSEMBLY		
	BULKHEAD	0.063	6061-T6
	DOUBLER	0.050	2024-T4 CLAD
	BLANKET	0.250	POLYVINYL CHLORIDE ACETATE
11	FRAME		2024-T351 FORGING
12	RING	0.040	2024-T4 CLAD
13	FORMER	0.040	2024-T4 CLAD
14	SPLICE	0.040	2024-T4 CLAD
15	ADAPTER ASSEMBLY		
	PLATE	0.040	2024-T3 CLAD
	CHANNEL	0.040	2024-T3 CLAD
	DOUBLER	0.050	2024-T3 CLAD
16	SPLICE	0.050	2024-T4 CLAD
17	BULKHEAD ASSEMBLY		
	FRAME	0.063	2024-T4 CLAD
	DOOR	0.063	2024-T4 CLAD
	DOUBLER	0.063	2024-T4 CLAD
	DOUBLER	0.160	2024-T4 CLAD
18	STRAP	0.063	2024-T3 CLAD
19	FORMER	0.040	2024-T3 CLAD
20	BULKHEAD ASSEMBLY		
	ANGLE		2024-T4 EXTRUSION
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.040	2024-T3 CLAD
	ANGLE	0.040	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
	ZEE		7075-T6 EXTRUSION
	CHANNEL		2024-T4 EXTRUSION
	FITTING		7075-T6 EXTRUSION
	FITTING		2024-T4 EXTRUSION
	DOUBLER	0.032	2024-T3 CLAD
	DOUBLER	0.040	2024-T3 CLAD
	DOUBLER	0.040	2024-T4 CLAD
	DOUBLER	0.050	7075-T6 CLAD
	STRAP	0.063	7075-T6 CLAD
	PLATE	0.125	2024-T3 CLAD
	PLATE	0.250	7075-T6 CLAD
	WEB	0.040	2024-T4 CLAD
	COVER PLATE	0.050	2024-T4 CLAD
	COVER PLATE	0.050	2024-T3 CLAD
	COVER	0.050	2024-T3 CLAD
	COVER	0.090	LAMINATED GLASS CLOTH
21	FORMER ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	STIFFENER	0.040	2024-T4 CLAD

JS-2-2-5709-1

Figure 57-9. External Wing Tank Skin and Structure (Sheet 1 of 3)

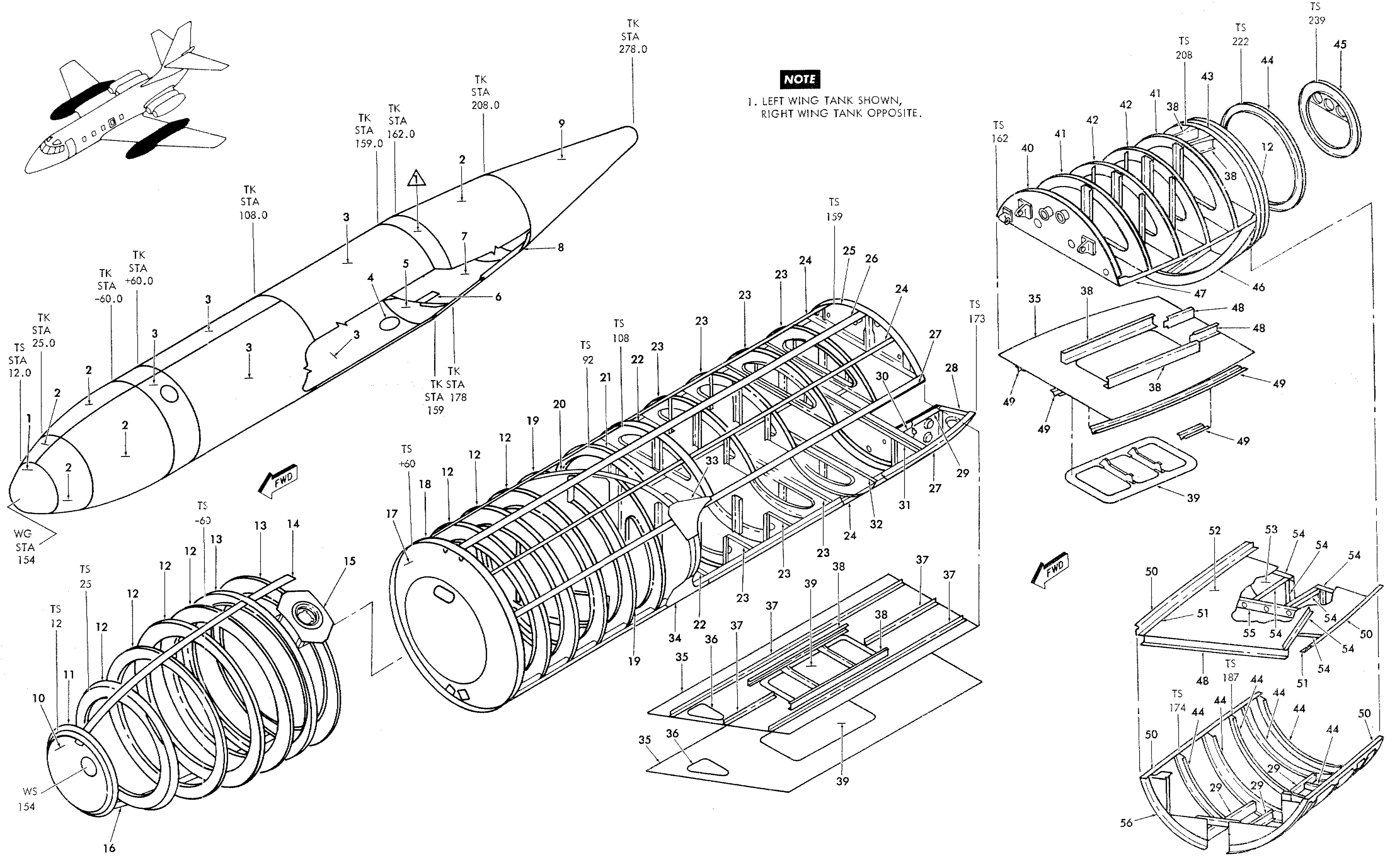
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
22	FORMER ASSEMBLY		
	TEE		2024-T42 EXTRUSION
	DOUBLER	0.032	2024-T4 CLAD
	WEB	0.040	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
	STIFFENER	0.040	2024-T4 CLAD
23	FORMER ASSEMBLY		
	FRAME	0.040	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
	STIFFENER	0.040	2024-T4 CLAD
24	STIFFENER		7075-T6 EXTRUSION
25	BULKHEAD ASSEMBLY		
	CHANNEL	0.080	2024-T3 CLAD
	ANGLE		7075-T6 EXTRUSION
	STIFFENER		7075-T6 EXTRUSION
	FITTING		2024-T4 EXTRUSION
	PLATE	0.040	2024-T3 CLAD
	PLATE	0.063	2024-T3 CLAD
	WEB	0.040	2024-T3 CLAD
	DOUBLER	0.032	2024-T3 CLAD
	DOUBLER	0.040	2024-T3 CLAD
	DOUBLER	0.125	2024-T3 CLAD
26	TEE		7075-T6 EXTRUSION
27	ANGLE		7075-T6 EXTRUSION
28	FORMER	0.032	2024-T4 CLAD
29	INTERCOSTAL	0.032	2024-T4 CLAD
30	WEB	0.032	2024-T4 CLAD
31	BULKHEAD ASSEMBLY		
	CHANNEL	0.080	2024-T4 CLAD
	ANGLE		7075-T6 EXTRUSION
	ANGLE	0.040	2024-T4 CLAD
	PLATE	0.063	2024-T3 CLAD
	DOUBLER	0.060	2024-T4 CLAD
	DOUBLER	0.125	7075-T6 CLAD
	WEB	0.040	2024-T3 CLAD
	DOOR	0.190	2024-T3 CLAD
	HANDLE		6063-T42 EXTRUSION
32	FORMER ASSEMBLY		
	FRAME	0.040	2024-T4 CLAD
	STIFFENER	0.040	2024-T4 CLAD
33	GUSSET	0.020	AMS4901 TITANIUM
34	GUSSET	0.050	2024-T3 CLAD
35	FLOOR PANEL	0.040	2024-T3 CLAD
36	DOOR	0.050	2024-T3 CLAD
37	TEE		2024-T4 EXTRUSION
38	CHANNEL	0.050	2024-T4 CLAD
39	DOOR ASSEMBLY		
	SKIN	0.040	2024-T3 CLAD
	STIFFENER	0.040	2024-T3 CLAD
	DOUBLER	0.156	2024-T3 CLAD
40	BULKHEAD ASSEMBLY		
	BULKHEAD	0.040	2024-T4 CLAD
	STIFFENER	0.080	2024-T4 CLAD
	STIFFENER	0.050	2024-T3 CLAD
	STIFFENER		7075-T6 EXTRUSION
	DOUBLER	0.040	2024-T4 CLAD
	FITTING		2024-T4 EXTRUSION
41	FORMER ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	CHANNEL	0.040	2024-T4 CLAD

JS-2-2-5709-2

Figure 57-9. External Wing Tank Skin and Structure (Sheet 2 of 3)

INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
42	FORMER ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	STIFFENER	0.040	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
43	BULKHEAD ASSEMBLY		
	CAP		2024-T42 EXTRUSION
	CHANNEL	0.063	2024-T4 CLAD
	CHANNEL	0.063	2024-T3 CLAD
	PLATE	0.040	2024-T4 CLAD
	STRAP	0.040	2024-T3 CLAD
	STRAP	0.125	7075-T6 CLAD
	ANGLE	0.050	2024-T4 CLAD
	ANGLE		7075-T6 EXTRUSION
	WEB	0.040	2024-T4 CLAD
	FITTING		356-T6 CASTING
44	FORMER	0.032	2024-T4 CLAD
45	FORMER ASSEMBLY		
	FORMER	0.032	2024-T4 CLAD
	FORMER	0.032	2024-T3 CLAD
	PLATE	0.040	2024-T3 CLAD
46	FORMER ASSEMBLY		
	FORMER	0.040	2024-T4 CLAD
	STRAP	0.063	2024-T3 CLAD
	ANGLE	0.063	2024-T3 CLAD
47	ANGLE	0.100	2024-T4 CLAD
48	CHANNEL		2024-T4 EXTRUSION
49	CHANNEL	0.040	2024-T3 CLAD
50	ANGLE	0.063	2024-T4 CLAD
51	ANGLE	0.032	2024-T4 CLAD
52	COVER PLATE	0.025	2024-T4 CLAD
53	WEB	0.025	2024-T3 CLAD
54	ANGLE		2024-T4 EXTRUSION
55	ZEE	0.071	2024-T3 CLAD
56	FORMER ASSEMBLY		
	FORMER	0.032	2024-T4 CLAD
	DOUBLER	0.032	2024-T4 CLAD
	SUPPORT	0.040	2024-T3 CLAD
	CHANNEL	0.050	2024-T4 CLAD

△1 0.080 2024-T3 CLAD COVER STRAP

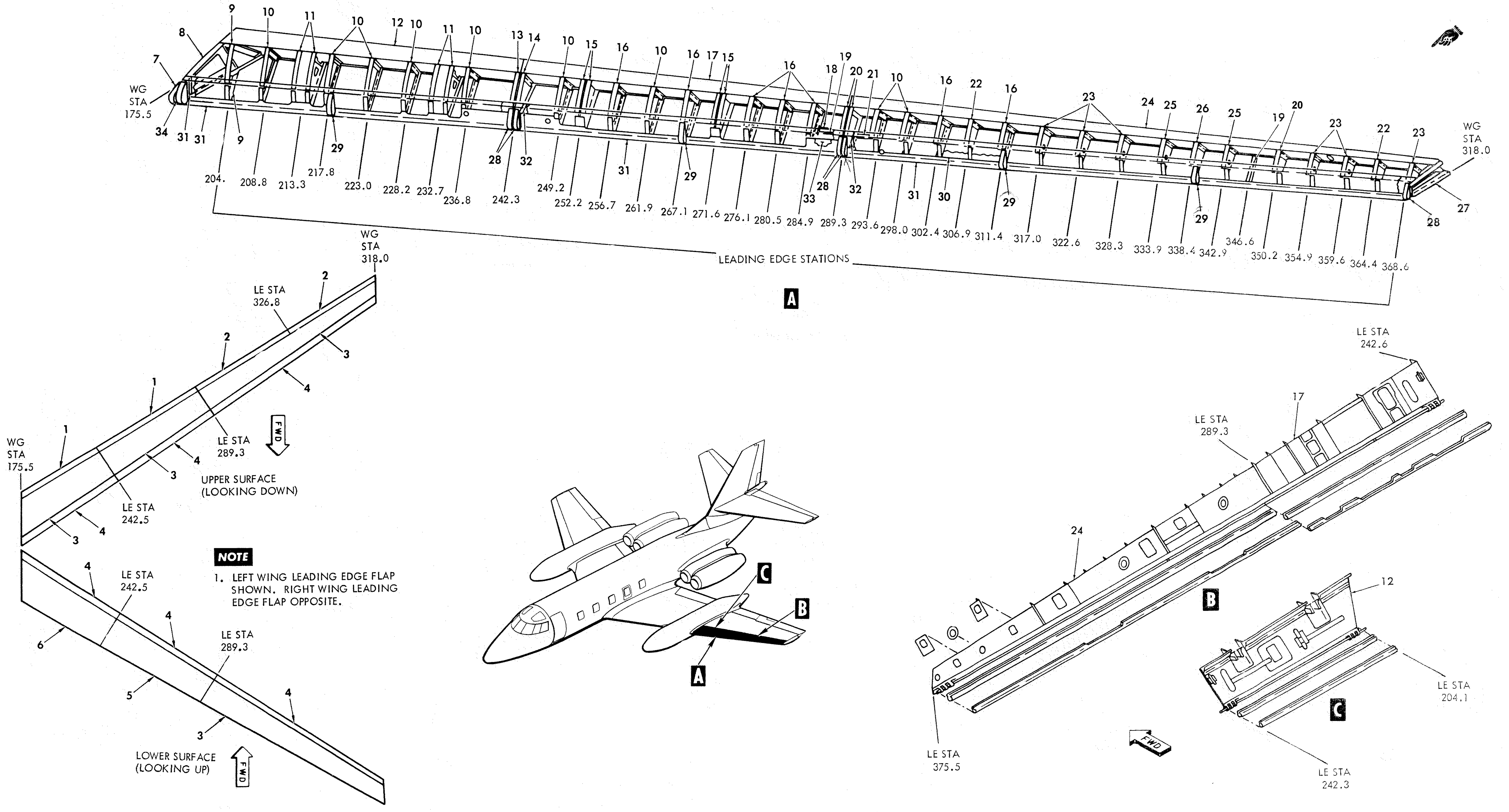


NOTE

1. LEFT WING TANK SHOWN, RIGHT WING TANK OPPOSITE.

Figure 57-9. External Wing Tank Skin and Structure (Sheet 3 of 3)

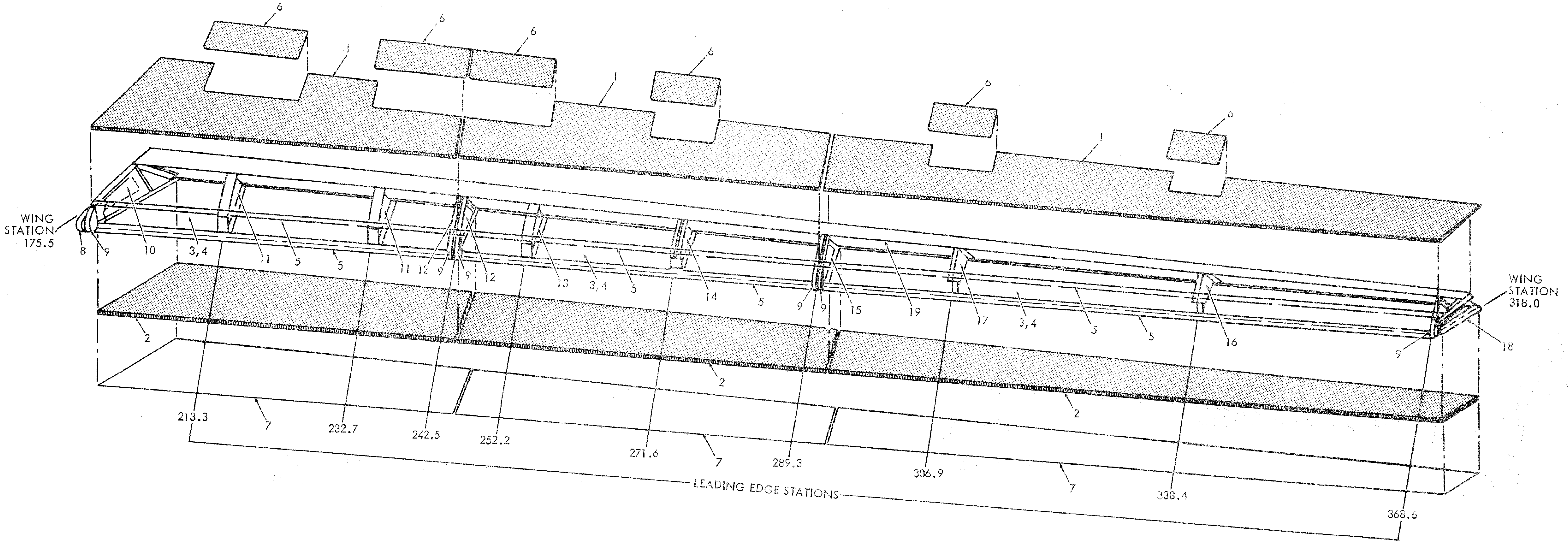
INDEX NO.	DESCRIPTION	THICKNESS	MATERIAL
1	SKIN ASSEMBLY		
	SKIN	0.160	7075-T6
	STRIP	0.032	301 CRES - 1/4 HARD
2	SKIN ASSEMBLY		
	SKIN	0.125	7075-T6
	STRIP	0.032	301 CRES - 1/4 HARD
3	SKIN	0.040	7075-T6 CLAD
4	NOSE SKIN	0.040	7075-T6 CLAD
5	CHEMICALLY MILLED SKIN	0.125	7075-T6 CLAD
6	CHEMICALLY MILLED SKIN	0.080	7075-T6 CLAD
7	NOSE RIB ASSEMBLY		
	RIB	0.060	LAMINATED FIBERGLASS
	PLATE	0.032	2024-T3 CLAD
8	RIB ASSEMBLY		
	RIB	0.040	7075-T6 CLAD
	ANGLE	0.050	7075-T6 CLAD
	DOUBLER	0.050	7075-T6 CLAD
	DOOR	0.040	7075-T6 CLAD
9	FORMER	0.040	7075-T6 CLAD
10	RIB ASSEMBLY		
	CAP	0.040	7075-T6 CLAD
	CLIP	0.040	7075-T6 CLAD
11	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	CLIP	0.050	7075-T6 CLAD
	CLIP		7075-T6 EXTRUSION
	FITTING		7075-T651 EXTRUSION
12	REAR BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.040	7075-T6 CLAD
	SUPPORT	0.071	7075-T6 CLAD
	ANGLE	0.071	7075-T6 CLAD
	STRAP	0.125	7075-T6 CLAD
	SEALING PLATE	0.050	2024-T4 CLAD
	STIFFENER		7075-T6 EXTRUSION
	BRACKET (LEFT FLAP ONLY)	0.063	2024-T4 CLAD
	DOOR	0.050	7075-T6 CLAD
	SPLICE	0.080	7075-T6 CLAD
	RETAINER		2024-T4 EXTRUSION
	SEAL		SILICONE RUBBER
	HINGE		2024-T4 EXTRUSION
13	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	CLIP	0.050	7075-T6 CLAD
	CLIP		7075-T6 EXTRUSION
14	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	STIFFENER		7075-T6 EXTRUSION
	CLIP		7075-T6 EXTRUSION
	RETAINER	0.071	7075-T6 CLAD
	SEAL		SILICONE RUBBER
15	RIB ASSEMBLY		
	ANGLE	0.050	7075-T6 CLAD
	CLIP	0.050	7075-T6 CLAD
	ANGLE		7075-T6 EXTRUSION
	FITTING	2.000	7075-T651 PLATE
16	RIB ASSEMBLY		
	CAP	0.040	7075-T6 CLAD
	CLIP	0.040	7075-T6 CLAD
	CLIP		2024-T4 EXTRUSION
17	REAR BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.040	7075-T6 CLAD
	WEB	0.063	7075-T6 CLAD
	SPLICE	0.080	7075-T6 CLAD
	FITTING	0.250	7075-T651 PLATE
	DOOR	0.050	7075-T6 CLAD
	RETAINER		2024-T4 EXTRUSION
	STRAP	0.080	7075-T6 CLAD
	SEAL		SILICONE RUBBER
	HINGE		2024-T4 EXTRUSION
18	CLIP		2024-T4 EXTRUSION
19	FORMER	0.040	7075-T6 CLAD
20	RIB ASSEMBLY		
	CHEMICALLY MILLED WEB	0.160	7075-T6
	ANGLE	0.040	301 CRES - 1/4 HARD
	CLIP	0.040	301 CRES - 1/4 HARD
	CLIP	0.050	301 CRES - 1/4 HARD
	CLIP	0.063	301 CRES - 1/4 HARD
21	INTERCOSTAL	0.040	7075-T6 CLAD
22	STOP ASSEMBLY		
	FITTING		7075-T6 BAR
	WEB	0.063	7075-T6 CLAD
	CLIP	0.063	7075-T6 CLAD
	CLIP		7075-T6 EXTRUSION
23	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	CLIP	0.040	7075-T6 CLAD
24	REAR BEAM ASSEMBLY		
	CAP		7075-T6 EXTRUSION
	WEB	0.040	7075-T6 CLAD
	DOUBLER	0.063	7075-T6 CLAD
	DOOR	0.080	7075-T6 CLAD
	ANGLE		2024-T4 EXTRUSION
	SPLICE	0.080	7075-T6 CLAD
	STRAP	0.080	7075-T6 CLAD
	DOOR	0.063	301 CREW - 1/2 HARD
	ANGLE	0.050	7075-T6 CLAD
	DOUBLER		7075-T6 CLAD
	RETAINER		2024-T4 EXTRUSION
	SEAL		SILICONE RUBBER
	HINGE		2024-T4 EXTRUSION
25	RIB ASSEMBLY		
	WEB	0.040	7075-T6 CLAD
	CLIP	0.040	7075-T6 CLAD
26	STOP ASSEMBLY		
	FITTING		7075-T6 BAR
	WEB	0.050	7075-T6 CLAD
	CLIP	0.063	7075-T6 CLAD
	CLIP		7075-T6 EXTRUSION
27	RIB ASSEMBLY		
	RIB	0.063	2024-T4 CLAD
	ANGLE	0.040	2024-T4 CLAD
	ANGLE	0.050	2024-T6 CLAD
28	FITTING	1.00	2024-T351 PLATE
29	FITTING	1.000	2024-T351 PLATE
30	DOUBLER	0.040	7075-T6 CLAD
31	ANGLE	0.040	2024-T4 2024 CLAD
32	FRONT BEAM	0.040	2024-T3 CLAD
33	DOUBLER	0.032	7075-T6 CLAD
34	FITTING	0.750	2024-T4 PLATE



NOTE

1. LEFT WING LEADING EDGE FLAP SHOWN. RIGHT WING LEADING EDGE FLAP OPPOSITE.

Figure 57-10. Leading Edge Flap Skin and Structure - Sheet Metal



INDEX NO.	DISCRIPTION	THICKNESS	MATERIAL
1	PANEL ASSEMBLY		
	UPPER FACE SHEET	0.020	7075-T6 CLAD
	LOWER FACE SHEET	0.016	7075-T6 CLAD
	UPPER T.E. FACE SHEET	0.020	302 STAINLESS STEEL
	LOWER T.E. FAIRING SHEET	0.016	302 STAINLESS STEEL
	CORE		1/4 HEX .003 5052 FOIL
2	PANEL ASSEMBLY		
	LOWER FACE SHEET	0.020	7075-T6 CLAD
	UPPER FACE SHEET	0.016	7075-T6 CLAD
	CORE		1/4 HEX .003 5052 FOIL
3	NOSE SKIN	0.050	2024-T4 CLAD
4	FRONT SPAR	0.040	2024-T4 CLAD
5	ANGLE	0.040	6061-T6
6	DOOR ASSEMBLY		
	UPPER FACE SHEET	0.040	7075-T6 CLAD
	LOWER FACE SHEET	0.040	7075-T6 CLAD
	CORE		6.0 - 1/4 HEX-30P-5052 FOIL
7	LOWER SKIN	0.050	2024-T4 CLAD
8	NOSE RIB ASSEMBLY		
	PLATE	0.032	2024-T3 CLAD
	NOSE RIB	0.060	LAMINATED FIBERGLASS
9	FITTING	0.040	2024-T4 PLATE
10	RIB ASSEMBLY		
	WEB	0.040	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
	DOOR	0.040	2024-T3 CLAD
11	RIB ASSEMBLY		
	WEB	0.050	7075-T6 CLAD
	FITTING		2 1/2 X 4 7075-T6 BAR
	TOP RIB CAP	0.100	7075-T6 EXTRUSION
	BOTTOM RIB CAP	0.100	7075-T6 EXTRUSION
	ANGLE	0.075	7075-T6 EXTRUSION
12	RIB	0.050	2024-T4 CLAD
13	RIB ASSEMBLY		
	FITTING		2 1/2-7075-T6 PLATE
	TOP CAP	0.100	7075-T6 EXTRUSION
	CLIP	0.050	7075-T6 CLAD
14	RIB ASSEMBLY		
	FITTING		3X6-7075-T6 BAR
	(2) TOP RIB CAPS	0.070	7075-T6 EXTRUSION
	CLIP	0.050	7075-T6 CLAD
15	RIB ASSEMBLY		
	WEB		7075-T6 BARE PLATE
	UPPER CAP	0.040	301-1/4 HARD STAINLESS STEEL
	AFT CLIP ANGLE	0.040	301-1/4 HARD STAINLESS STEEL
	FWD CLIP ANGLE	0.050	301-1/4 HARD STAINLESS STEEL
16	RIB ASSEMBLY		
	WEB	0.063	7075-T6 CLAD
	FITTING		2014-T6 FORGING
	UPPER CAP ANGLE	0.075	7075-T6 EXTRUSION
	LOWER CAP ANGLE	0.075	7075-T6 EXTRUSION
	FWD CLIP ANGLE	0.063	7075-T6 CLAD
17	RIB ASSEMBLY		
	WEB	0.063	7075-T6 CLAD
	FITTING		2014-T6 FORGING
	UPPER CAP ANGLE	0.075	7075-T6 EXTRUSION
	LOWER CAP ANGLE	0.078	7075-T6 EXTRUSION
	FWD CLIP ANGLE	0.063	7075-T6 CLAD
18	RIB ASSEMBLY		
	IN'BD RIB	0.040	2024-T4 CLAD
	OUT'BD RIB	0.040	2024-T4 CLAD
	NOSE RIB	0.032	2024-T4 CLAD
	AFT CLIP	0.040	2024-T4 CLAD
	ANGLE	0.050	2024-T4 CLAD
	FWD CLIP	0.040	2024-T3 CLAD
19	REAR BEAM CAP	0.080	7075-T6 EXTRUSION

Figure 57-11. Leading Edge Flap Skin and Structure—Honeycomb

(Pages 57-27 and 57-28 deleted)

CHAPTER 58

NONDESTRUCTIVE EVALUATION

TABLE OF CONTENTS

Section		Page
58.0	GENERAL	58-3
58.1	NONDESTRUCTIVE EVALUATION METHODS	58-23
58.2	WING GROUP	58-53
58.3	EMPENNAGE GROUP	58-95
58.4	FUSELAGE GROUP	58-129
58.5	LANDING GEAR GROUP	58-161
58.6	NACELLE GROUP	58-181

SECTION 58.0

GENERAL

Contents Page

INTRODUCTION 58-4

GENERAL INFORMATION 58-4

Airplane Description 58-4
How To Use The Chapter 58-4
Terminology and Standard Abbreviations 58-6
Glossary 58-6
Nondestructive Evaluation Symbols 58-7

SAFETY PRECAUTIONS FOR NDE METHODS 58-7

NDE Equipment Inside Fuel Tanks 58-7
Safety Requirements for Fluorescent Penetrant Inspections 58-11
Safety Requirements for Magnetic Particle Inspections 58-11
Safety Requirements for Eddy Current Inspections 58-11
Safety Requirements for Ultrasonic Inspections 58-11
Safety Requirements for X-Ray Inspections 58-11

CLEANING REQUIREMENTS FOR NDE METHODS 58-12

Visual Inspection Cleaning Requirements 58-12
Fluorescent Penetrant Inspection Cleaning Requirements 58-12
Magnetic Particle Inspection Cleaning Requirements 58-12
Eddy Current Inspection Cleaning Requirements 58-12
Ultrasonic Inspection Cleaning Requirements 58-13
Radiographic Inspection (X-ray) Cleaning Requirements 58-13

CLEANING METHODS 58-13

Cleaning Method A 58-14
Cleaning Method B 58-14
Cleaning Method C 58-14

TEMPORARY MARKING 58-14

Cleaning 58-14
Marking 58-14
Removal 58-14
Touch-Up 58-14

LOCALLY MANUFACTURED ITEMS 58-14

Dimension Tolerances 58-15
Angular Tolerances 58-15
Surface Roughness 58-15
Sawcut Width 58-15
Calibration Standard Materials 58-15

Contents

Page

NONDESTRUCTIVE EVALUATION MATERIALS AND EQUIPMENT 58-15

FIGURES

58.0-1 Figure Numbering Method 58-5
58.0-2 Nondestructive Evaluation Symbols 58-8
58.0-3 Cleaning Methods for Nondestructive Evaluation 58-13
58.0-4 Nondestructive Evaluation Materials and Equipment 58-16

58.0.1 INTRODUCTION.

This chapter of the manual contains all current requirements for accomplishing nondestructive evaluation of JetStar airplanes. The requirements prescribed by this chapter are intended to direct attention to known problem areas where defects would prevent the items from performing their designed functions. These requirements are developed for new equipment through maintenance engineering experience and comparison with similar installations on in-service equipment. They are refined and changed during the service life of the equipment by continually evaluating the performance of the equipment, results of scheduled inspections, and thorough study of failure data.

a. The inspections prescribed in this chapter will be accomplished at periods specified in applicable inspection requirements.

b. The inspection requirements are stated in such a manner as to establish what equipment is to be inspected and what conditions are to be sought. In scope, the requirements are designed *to direct the attention of maintenance personnel to components* and areas where defects are expected to develop.

This section of the manual provides information on safety precautions, cleaning requirements, and cleaning methods necessary for performing nondestructive evaluation (NDE) of JetStar airplane structure. In addition, requirements for locally manufactured items and a complete listing of NDE materials and equipment are included.

58.0.2 GENERAL INFORMATION.

58.0.2.1 AIRPLANE DESCRIPTION. The major assembly groups of the airplane are wings, empennage, fuselage, landing gear, and engine nacelles. For a description of each assembly group, refer to Chapters 57, 55, 53, 32, and 54 respectively. For principal dimensions and general data, refer to Chapter 6 and for location of access covers and doors, refer to Chapter 52.

58.0.2.2 HOW TO USE THE CHAPTER. The method for using this chapter is illustrated in figure 58.0-1. The sections of the chapter are numbered as follows:

58.0 - General

58.1 - Nondestructive Evaluation Methods

58.2 - Wing Group

58.3 - Empennage Group

58.4 - Fuselage Group

58.5 - Landing Gear Group

58.6 - Nacelle Group

a. The figure numbering system in sections 58.2 thru 58.6 consists of three sets of numbers, e.g., 58.2-2-5/2. The first number identifies the chapter and section of the manual; the second number designates the sequence of the figure in that section; and the third set of numbers indicates the type of primary and backup inspection procedures that the figure illustrates.

b. Types of inspection procedures are coded in the following manner:

0 - No inspection

1 - Visual inspection

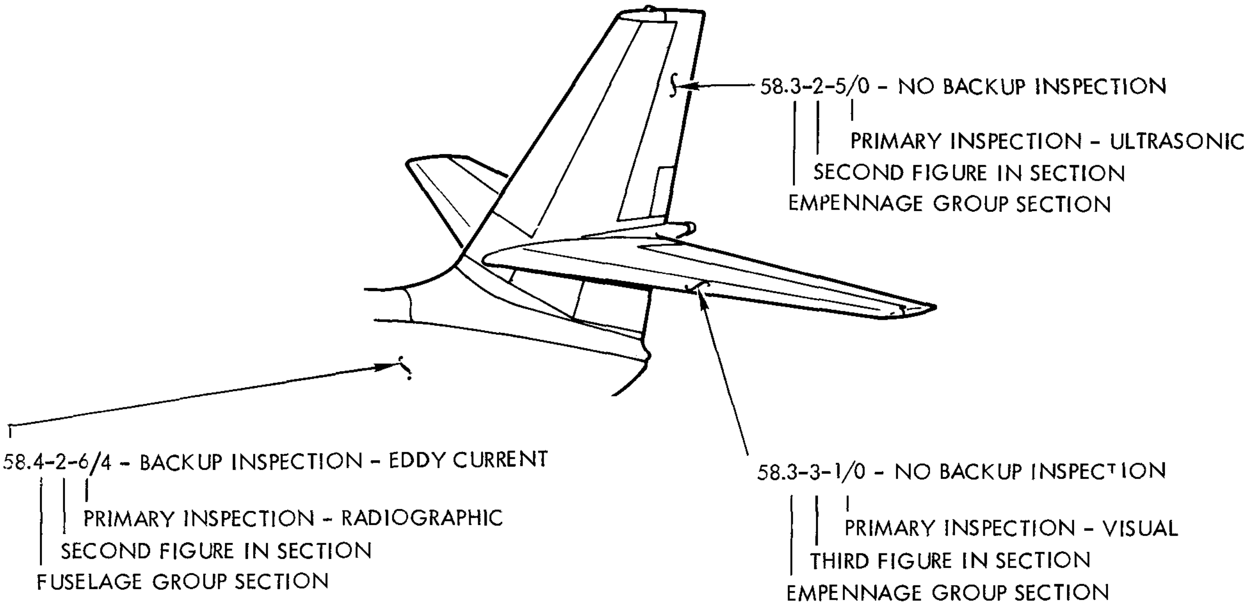
2 - Fluorescent Penetrant inspection

3 - Magnetic particle inspection

4 - Eddy current inspection

5 - Ultrasonic inspection

6 - Radiographic inspection


58.3-2-5/0 - NO BACKUP INSPECTION
PRIMARY INSPECTION - ULTRASONIC
SECOND FIGURE IN SECTION
EMPENNAGE GROUP SECTION
58.4-2-6/4 - BACKUP INSPECTION - EDDY CURRENT
PRIMARY INSPECTION - RADIOGRAPHIC
SECOND FIGURE IN SECTION
FUSELAGE GROUP SECTION
58.3-3-1/0 - NO BACKUP INSPECTION
PRIMARY INSPECTION - VISUAL
THIRD FIGURE IN SECTION
EMPENNAGE GROUP SECTION

NOTE

WHEN NDE PROCEDURES REQUIRE TWO OR MORE PRIMARY INSPECTIONS AND TWO OR MORE BACKUP INSPECTIONS TO BE PERFORMED, THE FIGURE NUMBER CODE WILL BE FORMED USING THE FIRST PRIMARY AND FIRST BACKUP INSPECTION METHOD.

THE FIGURE NUMBER CODE IS AS FOLLOWS

FIRST NUMBER .... CHAPTER AND SECTION

FIRST NUMBER INDICATION WILL BE:

58.0 GENERAL
58.1 NDE METHODS
58.2 WING
58.3 EMPENNAGE
58.4 FUSELAGE
58.5 LANDING GEAR
58.6 NACELLE

SECOND NUMBER .... SEQUENCE OF FIGURE IN SECTION

SECOND NUMBER INDICATION WILL BE:

1 - FIRST FIGURE IN SECTION
2 - SECOND FIGURE IN SECTION, ETC.

THIRD SET OF NUMBERS .... TYPE OF INSPECTION, FIRST NUMBER-PRIMARY/SECOND NUMBER-BACKUP

THIRD SET OF NUMBERS INDICATION WILL BE:

0 - NO INSPECTION
1 - VISUAL
2 - PENETRANT
3 - MAGNETIC PARTICLE
4 - EDDY CURRENT
5 - ULTRASONIC
6 - RADIOGRAPHIC

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Figure 58.0-1. Figure Numbering Method

58.0.2.2 (Cont).

c. Where two methods of inspection are given in the detail inspection procedure, the presentation is in order of primary and backup. The primary inspection is normally used to inspect the area first. The backup inspection, if required, is used to confirm a defect if detected during the primary inspection. If there is no essential preference between the primary and backup inspection, it is noted in the inspection procedure. Due to the geometry, location, and complexity of some parts, more than one primary and more than one backup inspection are required. These procedures are so noted.

d. A table of inspection procedures is located at the beginning of each structural group section. The table contains the index number, inspection title, paragraph number, and figure number for each detail inspection procedure.

The table will be followed by the detail inspection procedures listed in the table. This arrangement will be followed throughout the structural group sections.

58.0.2.3 TERMINOLOGY AND STANDARD ABBREVIATIONS. Such items as left and right, upper and lower, front and rear, forward and aft, and clockwise and counterclockwise refer to the airplane as viewed from the rear, on the centerline, looking forward. All dimensions, material gages, and distances are given in inches unless otherwise noted. Standard abbreviations used throughout this chapter are as follows:

BL	Buttock Line
BR	Back Reflection
CL	Centerline
CRES	Corrosion-Resistant Steel
CRT	Cathode Ray Tube
ES	Elevator Station
FFD	Film Focal Distance
FREQ	Frequency
FS	Fuselage Station
FUS STA	Fuselage Station
IN	Inches
INBD	Inboard
KV	Kilovolts
LE	Leading Edge
LEL	Lower Explosive Limit
LES	Leading Edge Station
LH	Left Hand (Left side of airplane)
MA	Milliamperes
MAM	Milliampere Minutes
MAS	Milliampere Seconds
MAX	Maximum
MHz	Megahertz
MIN	Minimum
NDE	Nondestructive Evaluation
NS	Nacelle Station
OUTBD	Outboard
QTY	Quantity
RH	Right Hand (Right side of airplane)
RHR	Roughness Height Ratio
RMS	Root Mean Square
SEC	Seconds
STGR	Stringer
TS	Tank Station
TYP	Typical
VSS	Vertical Stabilizer Station
WG	Wing
WL	Waterline
WS	Wing Station

58.0.2.4 GLOSSARY. The definitions and appreviations given in this glossary are not necessarily general definitions. They may apply only to the word phrase as it is used in this chapter and are described as follows:

AMPERE TURNS — When a coil is used to induce a longitudinal magnetic field in a part, the amperage of the current used multiplied by the number of complete turns in the coil gives the number of ampere turns.

BLACK LIGHT — Black light is a term applied to the invisible radiant energy in that portion of the ultraviolet spectrum just beyond the blue of the visible spectrum. The black light filter transmission normally peaks between 3,200 and 4,000 Angstrom units in wavelength. It is used to make visible, in the dark, materials containing fluorescence.

CORROSION — The deterioration of a metal by chemical or electro-chemical reaction with its environment.

COUPLANT — A substance used between the face of a transducer and test surface to permit or improve transmission of ultrasonic energy across this boundary or interface.

CRACKS, FATIGUE — Progressive cracks which develop in the surfaces caused by the repeated loading and unloading of the part by what is called reverse bending. These are also known as service cracks or service incurred failures.

DEFECT — Defect is a discontinuity which interferes with the usefulness of the part; a fault in any material, or part, detrimental to its serviceability.

DEVELOPERS — Developers are materials, wet or dry, which will draw or absorb penetrant from a surface crack or defect to the extent the defect will be visible under black light. Developers also control the background of the high-contrast penetrant color system.

DISCONTINUITY — An interruption in the normal physical structure or configuration of a part such as cracks, forging laps, seams, inclusion, porosity, etc. A discontinuity may or may not affect the usefulness of a part.

EMULSIFIER — An emulsifier is an agent which will, when added or applied to an oil-like penetrant material, make the penetrant removable from surfaces by water rinsing.

EMULSIFIER TIME — In penetrant inspection, the length of time elapsing between the addition of the emulsifier to the penetrant and the removal of the emulsified penetrant by the rinse operation.

FLUORESCENT — Fluorescent describes the effects produced by certain chemical products which exhibit the property of emitting visible light during the activation by black light.

FLUX DENSITY — The strength of the magnetic field, expressed in gauss units.

58.0.2.4 (Cont).

INCLUSION — Particles of impurities, usually oxides, sulphides, silicates, and such, which are retained in the metal during solidification of the metal, or which are formed by subsequent reaction of the solid metal.

INDICATION — Term used to describe a flaw-like condition, obtained during a primary or backup nondestructive technique, that will not require further inspection to determine its significance.

MAGNETIC FLUX — Also called lines of force. Imaginary lines used as a means of explaining the behavior of magnetic fields. Their conception is based on the pattern of lines produced when iron filings are sprinkled on a piece of paper laid over a permanent magnet.

PENETRANT — A liquid used in fluorescent penetrant inspection to penetrate into surface openings of parts to produce indictions at flaws.

PENETRANT REMOVER — A penetrant remover is a solvent-type liquid used to clean penetrants from the surface of a material.

PENETRANT SENSITIVITY — Penetrant sensitivity is the ability of the penetrant, processing technique, and developer to detect surface-connected discontinuities and provide an indication visible to the unaided eye.

PENETRATION TIME — In penetrant inspection, the length of time elapsing between the application of the penetrant to the part and the removal of the penetrant by the rinse operation.

POST EMULSIFIABLE — A post-emulsifiable penetrant is an oil-like material which, after application to a surface, can be made water-washable by the application of an emulsifier.

SATURATION — The point in the magnetization of a magnetizable object at which an increase in the magnetizing force produces no increase in the magnetic field within the part.

SUBSURFACE DEFECTS — Any defect which does not break the surface of the part in which it exists.

SUSPECT — Term used to describe a flaw-like condition, obtained during primary nondestructive technique, that will require further inspection to determine its significance.

SUSPENSION — The term applied to the liquid bath in which is suspended the ferromagnetic particles used in magnetic particle, wet continuous method.

ULTRASONIC SATURATION - Ultrasonic saturation is the maximum height of response that a given piece of ultrasonic equipment is capable of producing on the oscilloscope screen.

WATER-WASHABLE — A water-washable penetrant is an oil-like material containing an emulsifying agent which makes it washable by water rinsing.

WET CONTINUOUS METHOD — The magnetic particle inspection in which particles in wet form are applied to the surface of the part at the same time the part is being magnetized.

WET METHOD — The magnetic particle inspection method employing ferromagnetic particles suspended in a light oil bath as a vehicle.

58.0.2.5 NONDESTRUCTIVE EVALUATION SYMBOLS. Symbols used in detail inspection procedure figures are shown in figure 58.0-2. Also illustrated in the figure are examples of inspection areas. The examples illustrate various methods used to identify inspection areas and inspection methods.

WARNING

NDE equipment will not be used inside fuel tanks and/or in an explosive environment unless all safety precautions and protective requirements are strictly maintained. Failure to comply with these instructions may result in injury to personnel and/or damage to the aircraft.

WARNING

When making an inspection, survey the general area in advance to eliminate possible hazards due to loose structure, protruding work stands, auxiliary equipment, or electric cords, and presence of flammable or toxic fluids and/or fumes.

WARNING

If power is supplied to equipment, care should be taken to ensure that the equipment is well grounded to prevent electrical hazards.

58.0.3 SAFETY PRECAUTIONS FOR NDE METHODS.

Safety precautions start with good housekeeping practices. A clean work area and clean, well-kept equipment are prerequisites for conducting any nondestructive test.

58.0.3.1 NDE EQUIPMENT INSIDE FUEL TANKS. NDE equipment can be used inside fuel tanks and/or in an explosive environment under the following conditions:

a. Work area around the airplane shall be roped off 50 feet in all directions from the airplane.

b. Enclosed area shall be clearly marked from all angles with signs: Danger — Open Fuel Tanks — Unauthorized Personnel Keep Out.

c. The area shall be equipped with at least two fully charged, 50-pound CO_2 or equal fire extinguishers. Additional extinguishers shall be provided as available or needed.

d. Airplane must be electrically grounded to the earth to prevent static discharge. Any powered ground equipment on the

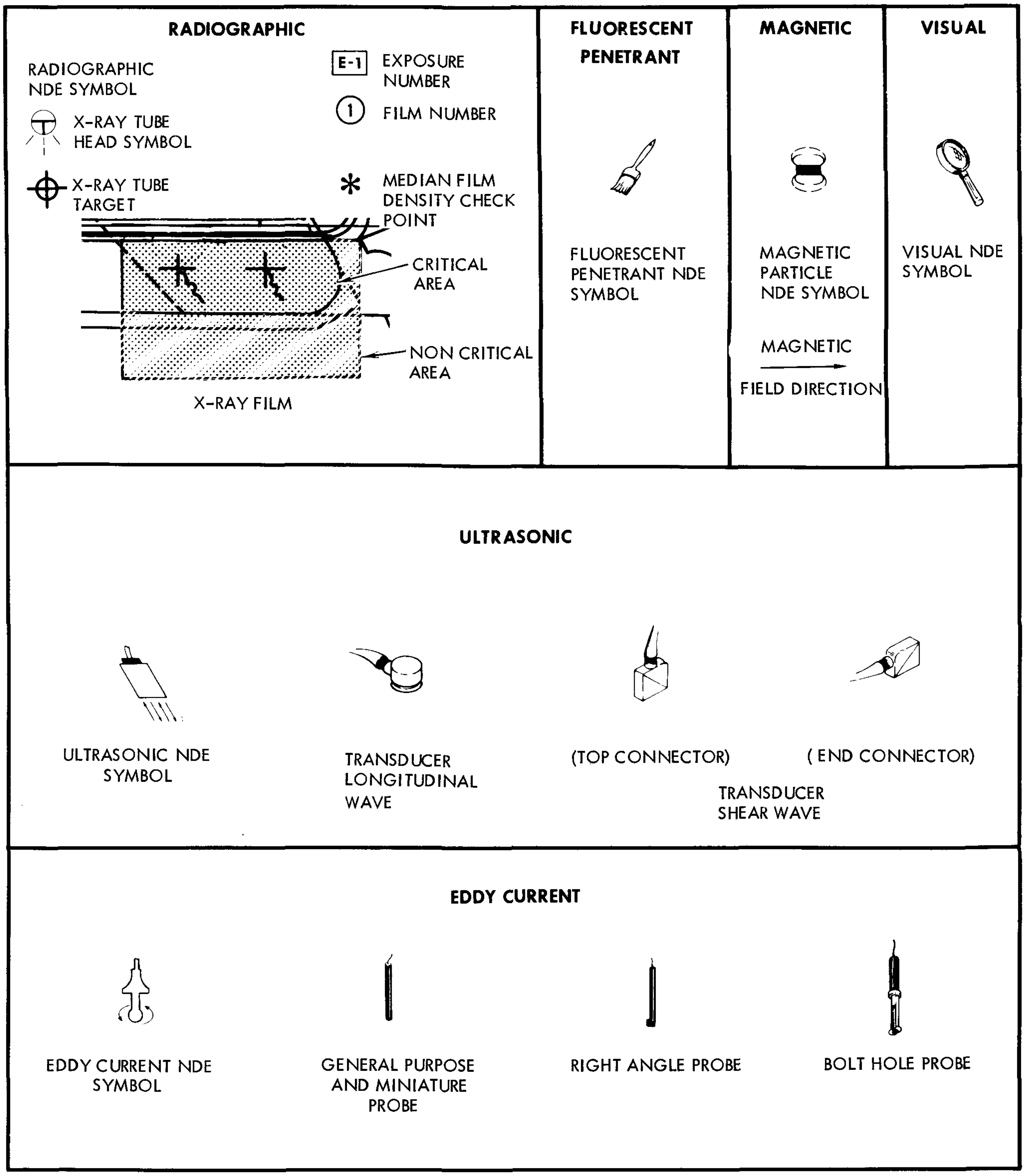

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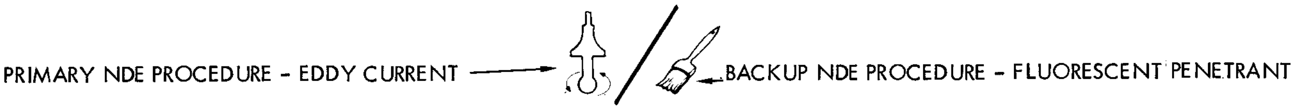
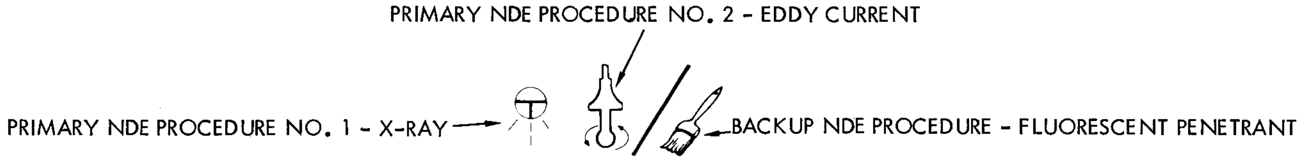
Figure 58.0-2. Nondestructive Evaluation Symbols (Sheet 1 of 3)

NOTE

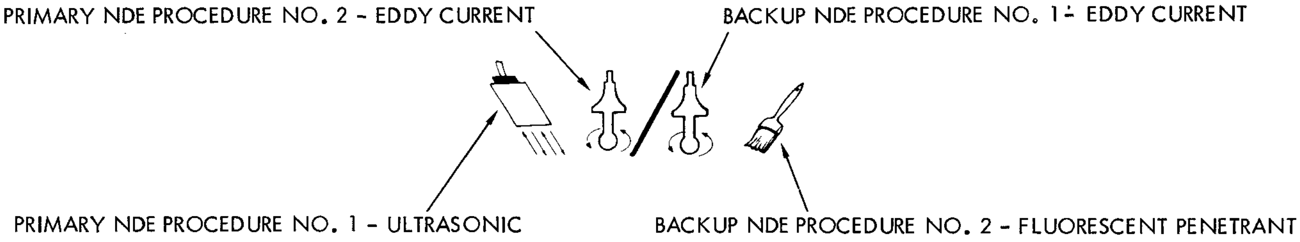
SYMBOLS SHOWN ARE EXAMPLES OF INSPECTION METHOD CODES USED IN ILLUSTRATIONS TO IDENTIFY THE TYPE OF INSPECTION METHODS BEING ILLUSTRATED.


PRIMARY NDE PROCEDURE - EDDY CURRENT
BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT

EXAMPLE NO. 1


PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT
PRIMARY NDE PROCEDURE NO. 1 - X-RAY
BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT

EXAMPLE NO. 2


PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT
BACKUP NDE PROCEDURE NO. 1 - EDDY CURRENT
PRIMARY NDE PROCEDURE NO. 1 - ULTRASONIC
BACKUP NDE PROCEDURE NO. 2 - FLUORESCENT PENETRANT

EXAMPLE NO. 3

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Figure 58.0-2. Nondestructive Evaluation Symbols (Sheet 2 of 3)

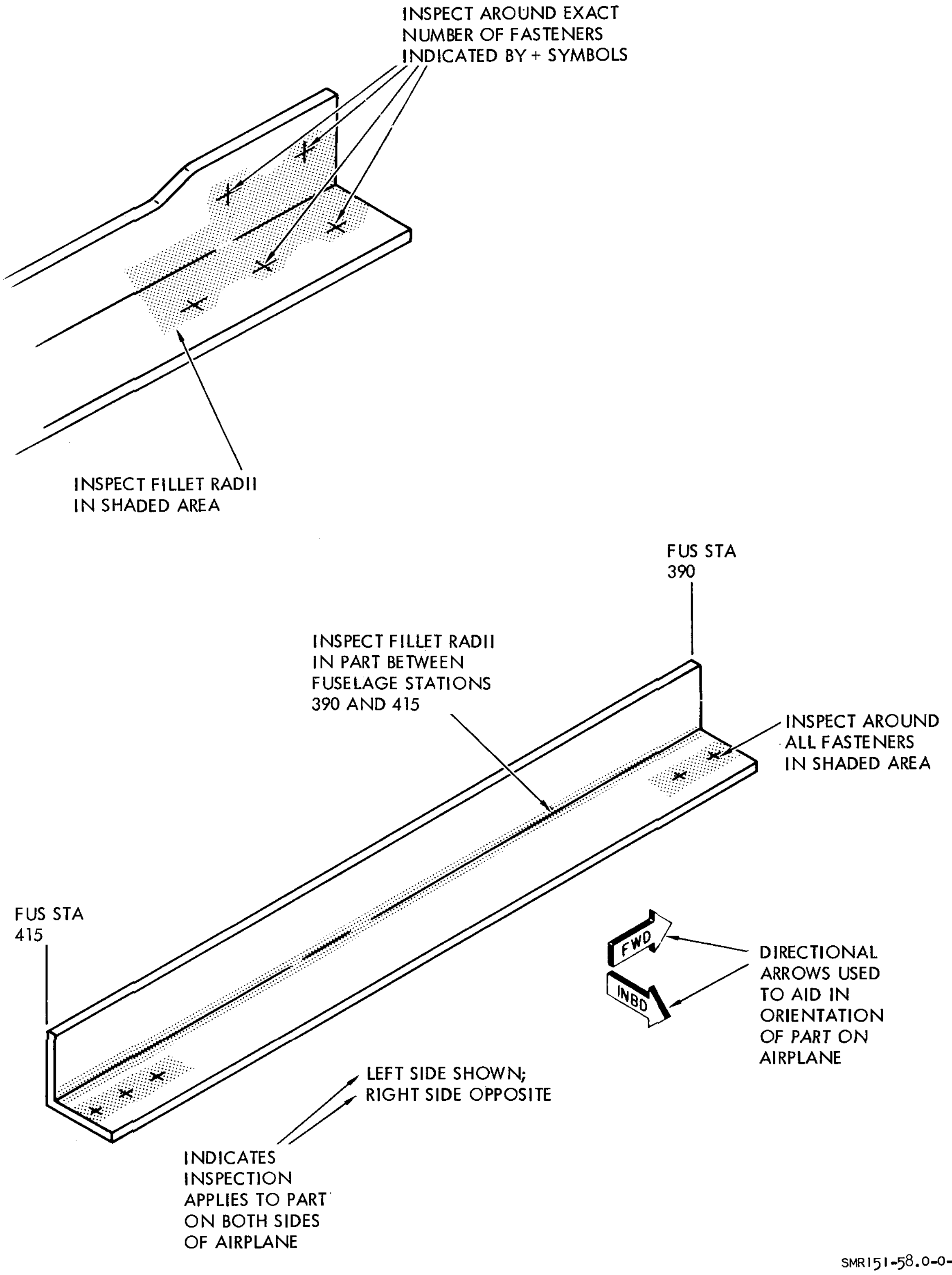

INSPECT AROUND EXACT NUMBER OF FASTENERS INDICATED BY + SYMBOLS
INSPECT FILLET RADII IN SHADED AREA
FUS STA 390
INSPECT FILLET RADII IN PART BETWEEN FUSELAGE STATIONS 390 AND 415
INSPECT AROUND ALL FASTENERS IN SHADED AREA
FUS STA 415
FWD
INBD
DIRECTIONAL ARROWS USED TO AID IN ORIENTATION OF PART ON AIRPLANE
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE
INDICATES INSPECTION APPLIES TO PART ON BOTH SIDES OF AIRPLANE
SMR151-58.0-0-002-3

Figure 58.0-2. Nondestructive Evaluation Symbols (Sheet 3 of 3)

58.0.3 (Cont).

airplane shall be grounded to the airplane and to the earth ground.

e. No switches (nonexplosion proof) will be activated or deactivated in the airplane. Power controls will be external (on/off controls) except for battery operated equipment.

f. Adjacent tanks and the tank to be entered shall be drained and purged. The fuel vapors inside the tanks shall be lowered to 1.5 percent of the lower explosive limit (LEL) or lower. This environment shall be reached and maintained whenever personnel are working inside the tanks without the use of respiratory equipment.

g. Continuous air purge will be implemented prior to and during entry into the fuel tanks.

h. Fuel vapor checks shall be made at least once each working shift and/or prior to starting entry into the fuel tanks.

i. An observer shall be present when an inspector is inside a fuel tank or small enclosed area.

j. Introduction of nonexplosion proof NDE equipment into the tanks shall be accomplished in the following manner:

(1) Perform an operational bench check of the equipment to verify proper functioning with no arcing or sparks.

(2) Remove all power from the unit and cord by disconnecting at the power supply point.

(3) Move equipment into operating position and verify that all power connections have been properly made and are secure, prior to connection with the main power supply.

k. Once fuel and/or any hazardous fluid is reintroduced in any system or tank structure that could expose trapped fuel vapors, the aircraft will be considered to be in a fire hazardous configuration.

58.0.3.2 SAFETY REQUIREMENTS FOR FLUORESCENT PENETRANT INSPECTIONS.

a. Adequate ventilation shall be used while handling cleaner, emulsifiers, penetrants, or developers. Prolonged or repeated inhalation of vapors or powders may cause irritation of mucous membrane areas of the human body.

b. Avoid contact with penetrant inspection materials. Continual exposure to penetrant materials may cause skin irritation.

(1) Wear neoprene gloves.

(2) Keep insides of neoprene gloves clean.

(3) Wash exposed areas of body with soap and water.

(4) Check for fluorescent penetrants on skin, clothes, and gloves with black light.

c. Follow manufacturer's instructions when using black light sources, and filter all light sources requiring filtering. Unfiltered light sources may damage the eyes.

d. Store all pressurized spray cans in a cool dry area protected from direct sunlight. Avoid exposure of pressurized spray cans to open flames. Temperatures in excess of 120 degrees F may cause the can to burst and injure personnel.

e. Avoid heating of penetrants to the point where certain lighter constituents are released. Volatile fumes may occur, creating both a fire and health hazard.

58.0.3.3 SAFETY REQUIREMENTS FOR MAGNETIC PARTICLE INSPECTIONS.

a. If the part being inspected is high strength steel, cleaning solvents used must be only those specified for high strength steel.

b. Keep cleaning fluids and magnetic particles out of areas where they could become entrapped.

c. Dissimilar metals such as bearings, bushings, and inserts must be removed and/or masked prior to cleaning and inspection.

d. Do not magnetize bearings.

e. Arcing caused by poor contact or excessive current between instrument heads could damage the eyes and skin.

f. Prolonged handling of pastes and cleaners could cause irritation of the skin.

58.0.3.4 SAFETY REQUIREMENTS FOR EDDY CURRENT INSPECTIONS. Under normal conditions, the use of eddy current equipment does not require special safety precautions other than the general warnings given.

58.0.3.5 SAFETY REQUIREMENTS FOR ULTRASONIC INSPECTIONS.

a. Ground instrument to airplane structure and use a grounded power cord.

b. Turn power OFF before connecting or disconnecting cable to transducer.

WARNING

The use of radiation in nondestructive testing presents a potential hazard to operating and adjacent personnel unless all safety precautions and requirements are observed.

58.0.3.6 SAFETY REQUIREMENTS FOR X-RAY INSPECTIONS. Personnel safety is one of the most important considerations in the use of X-ray equipment. Radiation from X-ray units and radioisotope sources is destructive to living tissue; therefore adequate protective methods and detection devices must be used. Since the detrimental effects of excessive exposure are not immediately apparent, personnel frequently exposed to X-rays should have periodic blood counts and physical examinations. The United States Bureau of Standards has issued a number of

58.0.3.6 (Cont).

handbooks on the subject of protection against radiation. These books can be obtained from the United States Department of Commerce. It is required that organizations using radiation equipment contact state agencies for guidance in preparing protection and safety requirements for industrial radiographic work.

a. Measuring and monitoring devices. Three types of radiation monitoring devices are available.

(1) One type consists of a small, pencil-like ionization chamber which is given an electrostatic charge at the beginning of each day's work. As it is subjected to penetrating radiation, it discharges proportionately to the amount of radiation received. By inserting this chamber into an electrometer, the amount of radiation received between the time of charge and the time of reading can be determined.

(2) The second and most commonly used radiation monitoring device is the film badge, consisting of a holder, a filter, and special X-ray film. These film badges are distributed to the radiographic operator, assistants, and all persons who may be in the vicinity of an exposure area. After 1 or 2 weeks of exposure, the film is processed and the resultant density of the negative is read by means of a densitometer. By comparing the density of the film with a master guide, the radiation received by the badge wearer can be determined.

(3) A third type of radiation monitor uses either a large *ionization chamber or geiger, proportional or scintillation* counters in conjunction with an electronic rate meter. This type of instrument reads the radiation intensity being received at a given location at the time the instrument is in operation and is independent of time. These devices are useful for posting the areas of radiation hazard and for determining the safe distance from the exposure area at which the operators and personnel must remain.

b. Protection of personnel. The objective sought by all protective measures, either by the use of distance or barriers, is to reduce the amount of radiation consistent with currently accepted practice. Distance from the radiation source is the most reliable radiation barrier.

(1) Protection of personnel may be provided in a number *of different ways, using various materials and thicknesses,* depending upon the energy of the radiation source and its application. Protection is usually built into the radiographic laboratory by lining the exposure room walls with sheet lead, or by constructing the room of concrete thick enough to give protection.

(2) In field work with X-ray sources, it may be necessary to depend on distance to provide proper protection. However, lead protective screens or operator booths can be provided to protect workers if they must remain in the immediate vicinity. The danger area surrounding the source during exposure should be enclosed with ropes and warning signs.

(3) The mechanical and electrical components of any radiographic equipment should not be adjusted, serviced, or repaired by anyone unfamiliar with the exact function of each part.

58.0.4 CLEANING REQUIREMENTS FOR NDE METHODS.

All nondestructive evaluation methods referred to in this chapter require some degree of cleaning, sealant removal, or finish stripping of the part or area under inspection. The degree of cleaning, sealant removal or finish stripping depends upon the NDE method to be used, the specific technique, and the condition of the finish system itself. For example, normal thicknesses of urethane coatings (2 to 8 mils) will produce some signal attenuation with ultrasonic methods for which compensation can be made. However, poorly bonded or excessively thick coatings can cause excessive signal loss with ultrasonic techniques sufficient to render the inspection useless. The NDE inspector must evaluate the specific finish condition, determine its effect on the NDE procedure being used, and then recommend the type of cleaning method which will provide reliable NDE results.

58.0.4.1 VISUAL INSPECTION CLEANING REQUIREMENTS. The visual examination of a part or surface requires the removal of any material which prohibits or limits the visual observation of the surface to be inspected. Protective coatings are not normally removed; however, should it become necessary to remove protective coatings from a suspect area, care should be exercised to preclude disturbing the metal or filling a potential crack or void. Should corrosion oxides or surface pitting prevent satisfactory inspection, remove the corrosion chemically prior to inspection. Preparation of parts for visual inspection shall be in accordance with figure 58.0-3. Removal of material with use of abrasive or metal cutting methods prior to visual inspection is prohibited.

58.0.4.2 FLUORESCENT PENETRANT INSPECTION CLEANING REQUIREMENTS. Penetrant techniques are highly dependent upon fatigue cracks or voids being clean and free from oxides, sealants, paints or liquids. Any contaminant in the crack or void may greatly reduce the effectiveness of the inspection. Prepare surfaces for penetrant inspection in accordance with figure 58.0-3. Should corrosion oxides or surface pitting prevent satisfactory inspection, remove chemically prior to inspection. Removal of material with use of abrasive or metal cutting methods prior to penetrant inspection is prohibited.

58.0.4.3 MAGNETIC PARTICLE INSPECTION CLEANING REQUIREMENTS. Magnetic particle inspection procedures require the induction of a magnetic field into parts to be inspected and the application of powders or liquids containing metal particles which can be influenced by the magnetic field. Plated or heavily painted areas can greatly affect the sensitivity of magnetic particle inspections. Heavy paint or other organic coatings shall be removed in accordance with figure 58.0-3.

58.0.4.4 EDDY CURRENT INSPECTION CLEANING REQUIREMENTS. Eddy current techniques require the induction of the electromagnetic flux fields into the material to be inspected. In order to accomplish this, the probe must be in close proximity to the material to be inspected and the surface must be reasonably smooth. Normally, smooth paint coatings have little effect on eddy current techniques. However, pitting and/or corrosion or peeling paint may prevent satisfactory inspection. Should pitting and/or corrosion or peeling paint prevent inspection, remove prior to inspection. Preparation of parts for eddy current inspection shall be in accordance with figure 58.0-3.

58.0.4.5 ULTRASONIC INSPECTION CLEANING REQUIREMENTS. Normally, smooth paint coatings of very thin levels (2 to 6 mils) are acceptable for ultrasonic inspection without removal. However, since different coating materials have different absorption characteristics, each inspection condition will have to be evaluated by the inspector and a decision must be made as to whether the coatings interfere sufficiently with the inspection to require removal. Preparation of parts for ultrasonic inspection shall be accomplished in accordance with figure 58.0-3. Should corrosion oxides or surface pitting prevent satisfactory inspection, remove prior to inspection.

58.0.4.6 RADIOGRAPHIC INSPECTION (X-Ray) CLEANING REQUIREMENTS. Radiographic techniques require the exposure of film to X-rays which have penetrated the part(s) to be inspected. Any foreign material such as heavy layers of sealant, pitting or corrosion may affect the inspection. Remove sealants and other coatings that interfere with film interpretation. Preparation of parts for radiographic inspection shall be accomplished in accordance with figure 58.0-3.

58.0.5 CLEANING METHODS.

The following is to be used as a guide for cleaning airplane surfaces by qualified cleaning personnel prior to conducting the inspection.

Note

The selection and use of cleaning materials and solutions should be done with extreme caution due to harmful effects some materials and solutions can have on certain materials and finishes.

CAUTION

Do not apply acid solutions to high strength steel for cleaning, sealant removal, paint stripping, or corrosion removal operations. Failure to comply could result in damage to the airplane.

CAUTION

After a part or area has been stripped of its protective finish, inspected and repaired, the finish system must be restored to completely satisfy the original finish requirements. For finish restoration instructions, refer to Section 51.8. In areas where corrosion has occurred, the corrosion oxides, pits and/or delaminations may have to be removed to adequately clean the area without use of abrasive products such as metal scrapers, drills or other removal tools which can obliterate small fatigue cracks and render the NDI method ineffective. See Section 51.2 for acceptable sealant and coating removal tools. The cleaning method for each type inspection and material is shown in figure 58.0-3.

CAUTION

Solvents and strippers are corrosive materials. Therefore, extreme care must be exercised to ensure that they are applied only to the area requiring cleaning or stripping.

TYPE INSPECTION	PREPARATION REQUIRED	CLEANING METHOD
Visual	Remove oil, grease, dirt and sealant	A
Fluorescent Penetrant	Remove oil, grease, dirt, sealant, and paint	B
Magnetic Particle	Remove oil, grease, dirt, sealant, and paint	B
Eddy Current	Remove oil, grease, dirt, and sealant	A
Ultrasonic	Remove oil, grease, dirt and sealant	A
Radiographic	Free of sealant and other foreign matter which interferes with film interpretation	C

Figure 58.0-3. Cleaning Methods for Nondestructive Evaluation.

58.0.5.1 CLEANING METHOD A. Clean the area to be inspected as follows:

a. Brush or wipe all loose dirt or debris from the areas using a non-metallic brush or clean dry rags.

b. Remove all heavy deposits of dirt by washing with water then wipe dry with clean dry rags.

c. Mechanically remove all sealants from the area using a phenolic scraper. Do not use a metal scraper.

d. Remove oil, grease, and residual sealant by wiping area clean with clean rags wet with Specification O-T-620, type 1 trichloroethane. Residual thin films of sealant may take several applications of solvent, scraping, and wiping to be completely removed.

e. The wall of fastener holes will be cleaned of sealant and foreign debris by squirting each hole with trichloroethane and inserting an aluminum oxide impregnated nylon brush into the hole. Move the brush back and forth through the hole for several strokes to dislodge residues of sealant and foreign matter. Remove the brush and again flush the hole with solvent. Repeat the brushing cycle if any residues remain. No abrasives will be used in cleaning holes other than the bristle brush.

f. Wipe all applied solvents dry with clean dry rags.

58.0.5.2 CLEANING METHOD B. Clean the area to be inspected as follows:

a. Brush or wipe all loose dirt or debris from the areas using a non-metallic brush or clean dry rags.

b. Remove all heavy deposits of dirt by washing with water then wipe dry with clean dry rags.

c. Mechanically remove all sealants from the area using a phenolic scraper. Do not use a metal scraper.

d. Remove oil, grease, and residual sealant by wiping area clean with clean rags wet with Specification O-T-620, type 1 trichloroethane. Residual thin films of sealant may take several applications of solvent, scraping, and wiping to be completely removed.

e. Remove paint finishes from the area to be inspected in accordance with Section 51.7.

58.0.5.3 CLEANING METHOD C. Clean the area to be inspected as follows:

a. Mechanically remove excess sealant using a phenolic scraper.

b. Remove residual films of sealant by wiping area with clean rags wet with Specification O-T-620, type 1 trichloroethane as required then wipe dry with clean dry rags.

58.0.6 TEMPORARY MARKING.

The following markers and procedures are recommended for all temporary marking on the airplane within the constraints as noted.

MATERIAL	VENDOR
Pen, Felt Tip, Blue	Scripto, Inc.
Alcohol Base Ink,	423 Houston St., N. E.
P/N Scripto Audio	Atlanta, Georgia 30303
Visual Grease Pencil	or equivalent
Red or Blue	

Note

If the area to be marked is extremely dirty, oily, or greasy, wipe surfaces dry with a clean cloth and then degrease area by wiping with a cloth moistened (not dripping) with aliphatic naphtha. Wipe surfaces dry with additional clean cloths.

58.0.6.1 CLEANING. Carefully wipe the area to be marked with a clean cloth.

CAUTION

Black felt tip pens or black grease pencils may contain materials that are detrimental to metals and must not be used.

58.0.6.2 MARKING. Perform necessary marking operations with marker specified above using a light touch.

58.0.6.3 REMOVAL. Remove markings as soon as there is no further need for them, preferably within 24 hours after application. The longer markings remain on a surface the more difficult they become to remove. Marking removal is accomplished by wiping the surface with a cloth moistened (not dripping) with isopropyl alcohol. Dry surfaces by wiping with additional cleaning cloths.

58.0.6.4 TOUCH-UP. In those cases where the alcohol wipe does not sufficiently remove the applied marking and where the remaining residue is objectionable, locally re-apply the required finish system over the markings.

58.0.7 LOCALLY MANUFACTURED ITEMS.

Calibration standards, angle wedges, and other items required by specific procedures but not listed in figure 58.0-4 are intended to be manufactured locally. Except as noted in the figure of the specific procedure, the following requirements apply:

58.0.7.1 DIMENSION TOLERANCES. Tolerance on all dimensions is as follows except where shop practice dictates fit:

a. Dimensions expressed in thousandths (X.XXX) shall have a standard tolerance of ± 0.010.

b. Dimensions expressed in hundredths (X.XX) shall have a standard tolerance of ± 0.03.

c. Dimensions expressed in tenths (X.X) shall have a standard tolerance of ± 0.1.

d. Dimensions expressed in fractions shall have a standard tolerance of ± 1/16.

58.0.7.2 ANGULAR TOLERANCES. Machined angular dimensions shall have a tolerance of ± 0 degrees 15' unless otherwise specified; all other angular tolerances shall be ± 2 degrees.

58.0.7.3 SURFACE ROUGHNESS. Maximum surface roughness is 125 RHR except for sound conducting surfaces of ultrasonic transducer holders and angle wedges, where the maximum surface roughness is 63 RHR.

58.0.7.4 SAWCUT WIDTH. Width of sawcuts is 0.001 to 0.020 inch.

58.0.7.5 CALIBRATION STANDARD MATERIALS. Materials for ultrasonic calibration standards may be substituted as follows when the specified materials are not available:

AIRPLANE PART MATERIAL	ACCEPTABLE CALIBRATION STANDARD MATERIAL
All aluminum alloys	6061-T6, 7075-T6, or 7079-T6 aluminum alloy

Note

Specify power requirements when ordering equipment.

58.0.8 NONDESTRUCTIVE EVALUATION MATERIALS AND EQUIPMENT.

Nondestructive evaluation materials and equipment are listed in figure 58.0-4.

TYPE INSPECTION	PART NUMBER	NOMENCLATURE	* MANUFACTURER
VISUAL	10X	Magnifying Glass, Pocket	W.M. Ziegler Tool Co.
	730N002-002A	Mirror, Inspection	Battey Machinery Co.
	255F005-002A	Flashlight, Inspection	W.M. Ziegler Tool Co.
	L309MC	Borescope, 0.394 OD X 34.53 Working Length, Right Angle	Lenox Instrument Co. Inc.
	CL1G18	Cable, 72-inch	Lenox Instrument Co. Inc.
	6127-2	Light	Lenox Instrument Co. Inc.
FLUORESCENT PENETRANT	ZA-43	Zyglo Kit	Magnaflux Corp.
	ZB-26	Black Light, Portable	Magnaflux Corp.
	3901	Fluted Filter, ZB-23A/ZB-26	Magnaflux Corp.
	1914	Bulb, Black Light ZB-23A/ZB-26	Magnaflux Corp.
	Type ZL-22A	Penetrant, Fluorescent	Magnaflux Corp.
	Type SKC-NF/ ZC-7B	Remover, Penetrant	Magnaflux Corp.
	Type ZP-9C	Developer, Penetrant	Magnaflux Corp.
MAGNETIC PARTICLE	H720	16-inch Coil Magnetic Particle Unit, 6000 Amps	Magnaflux Corp.
	GD54	Hood-H700	Magnaflux Corp.
	167220	12,000 Ampere SB 1619 Demagnetizer	Magnaflux Corp.
	14A Magnaglo	Magnetic Particle Material, Magnetic Dry Suspension Powder Concentrate	Magnaflux Corp.
	1-2120-40	Magnaglo Carrier	Magnaflux Corp.
	8493	Centrifuge Tube	Magnaflux Corp.
	1837A	Centrifuge Tube Stand	Magnaflux Corp.
	3901	Fluted Filter	Magnaflux Corp.
	1914	Black Light Bulb	Magnaflux Corp.
	DA-200	Portable Magnetic Inspection Unit, Hand Probe	Parker Research

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 1 of 6)

TYPE INSPECTION	PART NUMBER	NOMENCLATURE	* MANUFACTURER
MAGNETIC PARTICLE (Cont'd)	14AM Magnaflo	Prepared Fluorescent Bath	Magnaflux Corp.
	ZB-26	Black Light, Portable	Magnaflux Corp.
	2480	Field Indicator	Magnaflux Corp.
EDDY CURRENT	ED-520	Eddy Current Crack Detector	Magnaflux Corp.
	PDL-457A	Standard, General Eddy Current, Aluminum	Lockheed-Georgia Co.
	PDL-457S	Standard, General Eddy Current, Steel	Lockheed-Georgia Co.
	6100-1/8-S-BNC	Probe, Surface, General Purpose	Ideal Specialty Co.
	Optional: VM100P	Probe, Surface, General Purpose, with 6-foot cable and BNC connector	VM Products
	6100-1/4-S x 1.00	Probe, Surface, General Purpose 1/4-inch Dia. x 1-inch long	Ideal Specialty Co.
	VM100P-8	Probe, Surface, General Purpose, Flexible, 8-inches long	VM Products
	VM100SS	Probe, Shielded, Pencil	VM Products
	6310-90-1/4-1/8-BNC	Probe, Surface, Right Angle	Ideal Specialty Co.
	VM102AS	Probe, Shielded, 90-Degree	VM Products
	VM202RA	Probe, Shielded, 90-Degree	VM Products
	6200-3/16-BH-BNC	Probe, Bolt Hole, 3/16-inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-3/16	Probe, Bolt Hole, 3/16-inch Diameter	VM Products
	6200-1/4-BH-BNC	Probe, Bolt Hole, 1/4-inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-1/4	Probe, Bolt Hole, 1/4-inch Diameter	VM Products

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 2 of 6)

TYPE INSPECTION	PART NUMBER	NOMENCLATURE	* MANUFACTURER
EDDY CURRENT (Cont'd)	6200-5/16-BH-BNC	Probe, Bolt Hole, 5/16 inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-5/16	Probe, Bolt Hole, 5/16-inch Diameter	VM Products
	6200-3/8-BH-BNC	Probe, Bolt Hole, 3/8-inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-3/8	Probe, Bolt Hole, 3/8-inch Diameter	VM Products
	6200-1/2-BH-BNC	Probe, Bolt Hole, 1/2-inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-1/2	Probe, Bolt Hole, 1/2-inch Diameter	VM Products
	VM101BS-1/2	Probe, Bolt Hole, Shielded, 1/2-inch Diameter	VM Products
	6200-5/8-BH-BNC	Probe, Bolt Hole, 5/8-inch Diameter	Ideal Specialty Co.
	Optional: VM101BC-5/8	Probe, Bolt Hole, 5/8-inch Diameter	VM Products
	VM101BS-5/8	Probe, Bolt Hole, Shielded, 5/8-inch Diameter	VM Products
	6200-3/4-BH-BNC	Probe, Bolt Hole, 3/4-inch Diameter	Ideal Specialty Co.
	Optional: VM101B-3/4	Probe, Bolt Hole, 3/4-inch Diameter	VM Products
	VM101BS-3/4	Probe, Bolt Hole, Shielded, 3/4-inch Diameter	VM Products
	6200-1.252-BH-BNC	Probe, Bolt Hole, 1.252-inch Diameter	Ideal Specialty Co.
	VM101BS-1½	Probe, Bolt Hole, Shielded, 1½-inch Diameter	VM Products
	JSK-7	Bore Inspection Assembly Equipment	Lockheed-Georgia Company
	98418	Battery, ED-520 Replacement	Magnaflux Corp.
	SM-3/16	Brush, Hole Cleaning, Nylon, Aluminum Oxide Impregnated, 3/16-inch Diameter	Weiler Brush Co.

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 3 of 6)

TYPE INSPECTION	PART NUMBER	NOMENCLATURE	* MANUFACTURER
EDDY CURRENT (Cont'd)	24180	Brush, Hole Cleaning, Nylon, Aluminum Oxide Impregnated, 1/4-inch Diameter	Anderson Co.
	Optional: 21126	Brush, Hole Cleaning, Nylon, Aluminum Oxide Impregnated, 1/4-inch Diameter	Weiler Brush Co.
	4110	Brush, Hand Use, Nylon Bristle	Weiler Brush Co.
	Optional: 840	Brush, Hand Use, Nylon Bristle	Osborne Mfg. Corp.
	Optional: 7229	Brush, Hand Use, Nylon Bristle	Milwaukee Brush Co.
	7230	Brush, Hole Cleaning, Nylon, Aluminum Oxide Impregnated, 1/2-inch Diameter	Milwaukee Brush Co.
	Optional: 841	Brush, Hole Cleaning, Nylon, Aluminum Oxide Impregnated, 1/2-inch Diameter	Osborne Mfg. Corp.
	Optional: 44111	Brush, Hole Cleaning, Nylon Aluminum Oxide Impregnated, 1/2-inch Diameter	Weiler Brush Co.

NOTE

△1 VM Products Bolt Hole Probe, P/N VM101BC-() or VM101BCS-() is fabricated with a permanently attached cable. VM products Bolt Hole Probe P/N VM101B-() or VM101BS-() is fabricated with a microdot connector and may be used with the HMT 7-41-BM or 57A2271 cable.

ULTRASONIC	Mark I	Portable Ultrasonic Flaw Detector	Sonic Instruments Inc.
	Optional: Mark IV	Portable Ultrasonic Flaw Detector	Sonic Instruments Inc.
	HMT 7-41-BM	Cable, 6-foot Long, Microdot/BNC	Mastercraft Enterprises Inc.
	Optional: 57A2271	Cable, 6-foot Long, Microdot/BNC	Automation Industries Inc.
	SAE 10	Oil, Couplant	Commercial

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 4 of 6)

TYPE INSPECTION	PART NUMBER	NOMENCLATURE	* MANUFACTURER
ULTRASONIC (Cont'd)	Z-103-ABFR	Transducer, Longitudinal Wave, 5MHz, 0.250 Diameter	Krautkramer-Branson
	Optional: A110	Transducer, Longitudinal Wave, 5 MHz, 0.250 Diameter	Panametrics, Inc.
	Aerotech Model FDU	Transducer, Longitudinal Wave, 5 MHz, 0.250 Diameter, Pitch-Catch, with Cable	Krautkramer-Branson
	SFZ57A2279	Transducer, Longitudinal Wave, 10 MHz, 0.250 Diameter	Automation Industries, Inc.
	HMT7-121TC	Transducer, Shear Wave, 45-degrees, 5 MHz	Mastercraft Enterprises, Inc.
	Optional: 57A9466	Transducer, Shear Wave, Sub-Miniature, 45-degrees, Special Order	Automation Industries, Inc.
	HMT7-123TC	Transducer, Shear Wave, 70-degrees, 5 MHz	Mastercraft Enterprises, Inc.
	HMT 7-221	Transducer, Shear Wave, 45-degrees, 5 MHz	Mastercraft Enterprises Inc.
	HMT 7-223	Transducer, 0.250 x 0.750, Shear Wave, 70-degrees, 5 MHz	Mastercraft Enterprises Inc.
RADIOGRAPHIC	MXR150A	X-Ray Unit, Portable, 40-Degree Tube Head, End Port, 150 KVP	Magnaflux Corp.
	300132	Tripod	Magnaflux Corp.
	300133	Tube Head Bracket	Magnaflux Corp.
	300365	Maintenance Kit MX150R	Magnaflux Corp.
	Optional: SPX-160E-8-C	X-Ray Unit, Portable, 40-Degree Tube Head, End Port, 160 KVP	Automation Industries, Inc.
	NDT 35	Day Pack, Radiographic Film, (Specify Size)	E.I. Dupont De Nemours & Co. Inc.
	NDT 45	Day Pack, Radiographic Film, (Specify Size)	E.I. Dupont De Nemours & Co. Inc.
	NDT 55	Day Pack, Radiographic Film, (Specify Size)	E.I. Dupont De Nemours & Co. Inc.

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 5 of 6)

*Manufacturer address and phone number:

Anderson Co.	1083 Southbridge St. Worcester, MA 01610	(617) 791-3681
Automation Industries, Inc.	Shelter Rock Road Danbury, CT 06810	(203) 748-3581
Battey Machinery Co.	Kings Bridge Road Carrolton, GA	(404) 577-6983
E.I. Dupont De Nemours & Co., Inc.	Wilmington, DE 19898	(302) 994-4528
Ideal Specialty Co.	2531 E. Independence Tulsa, OK	(918) 834-1657
Krautkramer-Branson	250 Long Beach Blvd. P. O. Box 408 Stratford, CT 06497	(203) 377-3900
Lenox Instrument Co. Inc.	111 E Luray St. Philadelphia, PA 19120	(215) 324-4543
Lockheed-Georgia Co.	86 South Cobb Drive Marietta, GA 30063	(404) 424-2009
Magnaflux Corp.	7338 W. Lawrence Ave. Chicago, IL 60656	(312) 867-8000
Mastercraft Enterprises, Inc.	Box 6065 Marietta, GA 30065	(404) 428-4664
Milwaukee Brush Co.	2234 N. 30th St. Milwaukee, WI 53245	(414) 374-4010
Osborne Mfg. Corp.	5401 Hamilton Ave. Dept. T Cleveland, OH 44114	(216) 361-1900
Panametrics, Inc.	221 Crescent St. Waltham, MA 02154	(617) 899-2719
Parker Research	Box 596 Dunedin, FL 33528	(813) 733-6081
Sonic Instruments, Inc.	1018 Whitehead Rd. Ext. Trenton, NJ 08638	(609) 883-5030
VM Products	7420 Park Ave. Tacoma, WA 98408	(206) 475-6892
Weiler Brush Co.	Box T Cresco, PA 18326	(717) 595-7496
W.M. Ziegler Tool Co.	13570 Auburn Ave. Detroit, MI	(313) 838-2360

Figure 58.0-4. Nondestructive Evaluation Materials and Equipment. (Sheet 6 of 6)

SECTION 58.1

NONDESTRUCTIVE EVALUATION METHODS

Contents Page

INTRODUCTION 58-24

VISUAL METHOD 58-24

Purpose 58-24
Visual Aids 58-24
Inspection Procedure 58-24

FLUORESCENT PENETRANT METHOD 58-24

Purpose 58-24
Capabilities 58-25
Detection Limitations 58-25
Temperature Limitations 58-25
Materials 58-25
Inspection Procedure 58-25
Cleaning of Parts After Inspection 58-25

MAGNETIC PARTICLE METHOD 58-25

Purpose 58-25
Defects 58-25
Circular Magnetization 58-27
Longitudinal Magnetization 58-29
Wet Continuous Method 58-29
Parker Probe 58-29
Demagnetizing 58-29
Materials 58-32
Concentration Control 58-32
Inspection Lights 58-32
Temperature Limitations 58-32

EDDY CURRENT METHOD 58-32

Purpose 58-32
Flaw Detection Methods 58-32

Calibration of Magnaflux P/N ED-520 58-35
Scanning Techniques 58-39
Equipment 58-41
Inspection Procedure 58-41
Temperature Limitations 58-41

ULTRASONIC METHOD 58-44

Purpose 58-44
Flaw Detector Equivalency 58-44
Ultrasonic Sensitivity (Gain) Compensation 58-44
Ultrasonic Instrument Calibration 58-48

RADIOGRAPHIC (X-RAY) METHOD 58-48

Purpose 58-48
Limitations 58-50

Contents Page

Radiographic Techniques 58-50
Set Up 58-50
Adjustments 58-50
Film Substitution 58-50
Film Density 58-50
Film Loading 58-50
Film Identification 58-50
Radiographic Inspection Data Form 58-50
Airplane Records 58-50
Polaroid Film 58-50

ACCEPTANCE/REJECTION CRITERIA 58-50

FIGURES

58.1-1 Fluorescent Penetrant Inspection 58-26
58.1-2 General Procedure for Fluorescent Penetrant Inspection 58-27
58.1-3 Circular Magnetization Applications 58-28
58.1-4 Longitudinal Magnetization Applications 58-30
58.1-5 Longitudinal Magnetization Application on Typical Landing Gear Part 58-31
58.1-6 Typical Eddy Current Surface Probe 58-33
58.1-7 Typical Eddy Current Hole Probe 58-34
58.1-8 Eddy Current Inspection Instrument 58-36
58.1-9 Eddy Current Calibration Standard 58-37
58.1-10 Eddy Current Calibration Setup 58-38
58.1-11 Probe Guide for Scanning Around Flush Mounted Fasteners 58-39
58.1-12 Probe Guide for Scanning Around Protruding Fasteners 58-40
58.1-13 Scanning Between Fasteners 58-42
58.1-14 Scanning Fillet Radii 58-42
58.1-15 Scanning Bolt Holes 58-43
58.1-16 Ultrasonic Equipment Comparison 58-45
58.1-17 Computation for Ultrasonic Sensitivity (Gain) Compensation 58-46
58.1-18 Ultrasonic Instrument Calibration 58-49
58.1-19 Equivalent Film Class and Source 58-51
58.1-20 Sample Radiographic Inspection Data Form 58-52

58.1.1 INTRODUCTION.

This section of the manual provides general information on the purpose, capabilities, and limitations of the nondestructive evaluation methods employed in the NDE procedures of Sections 58.2 through 58.6. Specific instructions for NDE euipment calibration and initial settings, as well as detailed inspection procedures, are included.

58.1.2 VISUAL METHOD.

58.1.2.1 PURPOSE. Nondestructive evaluation by visual means is by far the oldest and most economical method. Those defects which would escape detection by the naked eye may be seen when magnified.

58.1.2.2 VISUAL AIDS. Borescopes and magnifying glasses may be used to perform visual inspection.

Note

> Whenever geometry of a part conceals the inspection area, visual aids such as a mirror or borescope should be used during the inspection.

58.1.2.3 INSPECTION PROCEDURE. In order to visually inspect aircraft components, the inspector must have a knowledge of defects which are likely to develop and recognize areas where such failures might occur. Recommended equipment and lighting aids should be used. A preliminary inspection of the overall general area should be made for cleanliness, presence of any foreign objects, security of parts, corrosion, and damage. Upon completion of the general area inspection, perform a specific visual inspection of the desired area as follows:

a. Provide proper lighting.

b. Clean surface of parts prior to inspection.

c. Carefully inspect area, using visual aids as necessary.

d. Record all discrepancies for disposition.

58.1.3 FLUORESCENT PENETRANT METHOD.

58.1.3.1 PURPOSE. The fundamental purpose of fluorescent penetrant inspection is to increase the visible contrast between the discontinuity and its background. This is done by treating the whole area with an appropriate searching liquid of high

58.1.3.1 (Cont).

mobility and penetrating power (which enters the surface opening of the discontinuity), and then encouraging the liquid to emerge from the discontinuity to reveal the flaw pattern to the inspecting personnel, when exposed to ultraviolet light for the purposes of inspection and interpretation. The success and reliability of penetrant inspection depends upon the thoroughness with which the part is prepared from the precleaning process all the way through to the final interpretation of the indications. Penetrant inspection is not a method by which a part is processed through a machine which separates the good parts from the bad. The inspector must carefully process the part, search out indications, and then determine the extent of defects found.

58.1.3.2 CAPABILITIES. Fluorescent penetrant inspection is a nondestructive test for defects open to the surface in parts made of any nonporous material. (See figure 58.1-1). Penetrant inspection is used with equal success on such metals as aluminum, magnesium, brass, copper, cast iron, stainless steel, titanium, nonmagnetic alloys, and on ceramics, molded rubber, powdered metal products, cermet, or glass.

58.1.3.3 DETECTION LIMITATIONS. The chief limitation of fluorescent penetrant inspection is that the defect must be open to the surface in order to let the penetrant get into the defect. It is also essential that there be no contaminating material within the defect which might either prevent the penetrant from entering or reduce its visibility.

CAUTION

Activities will use only one family of materials (or group) manufactured and supplied by one manufacturer as a group of materials when performing a penetrant inspection authorized for use.

WARNING

If a penetrant is heated to the point where some of its lighter constituents are driven off, the sensitivity of inspection may be greatly reduced, and volatile fumes may create a fire or health hazard. Heating the penetrant is therefore not recommended.

58.1.3.4 TEMPERATURE LIMITATIONS.

a. Fluorescent penetrant temperature: Most penetrants operate satisfactorily at temperatures as low as 4 degrees to 5 degrees C (40 degrees F), but operation below this temeprature can result in unstatisfactory conditions. Most penetrants designed for use in open tanks are reasonably stable at temperatures up to about 38 degrees C (100 degrees F).

b. Wet developer temperature: Wet developer, being a water mixed solution, cannot operate below the freezing temperature of water. Heating wet developer above 90 degrees F does not improve its function.

c. Drier temperature: The desired temperature of the drier oven is 150 degrees F to 180 degrees F. Parts should be removed from the drier as soon as they are dry. If overheated prior to the application of the penetrant, excessive evaporation of the penetrant may occur. If the part is allowed to remain in the drier oven too long after the penetrant has been applied, the sensitivity may be reduced. Temperatures above 180 degrees F or excessive drying times may decrease the sensitivity of certain fluorescent penetrants.

d. Temperature of part: Warming parts to approximately 100 degrees F will minimize the required penetrant dwell time. The warm part reduces the viscosity of the penetrant, permitting the penetrant to more readily enter the discontinuities.

58.1.3.5 MATERIALS. The penetrant materials specified in figure 58.1-2 are Magnaflux Corporation penetrant materials. These materials and any equivalent materials must conform to Specification MIL-I-25135. Equivalent materials must exhibit equal sensitivity for crack detection.

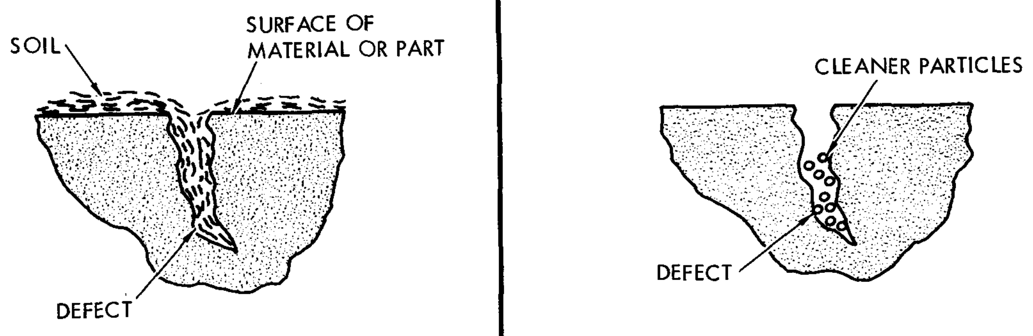
58.1.3.6 INSPECTION PROCEDURE. The procedure given in figure 58.1-2 is to be used for parts that are to be inspected on the airplane or that are removed from the airplane. These inspection procedures are for support of the fluorescent penetrant inspections required in this manual.

58.1.3.7 CLEANING OF PARTS AFTER INSPECTION. After inspection, the developer can be removed from the part by washing by hand with clean cloth and water or by solvent.

58.1.4 MAGNETIC PARTICLE METHOD.

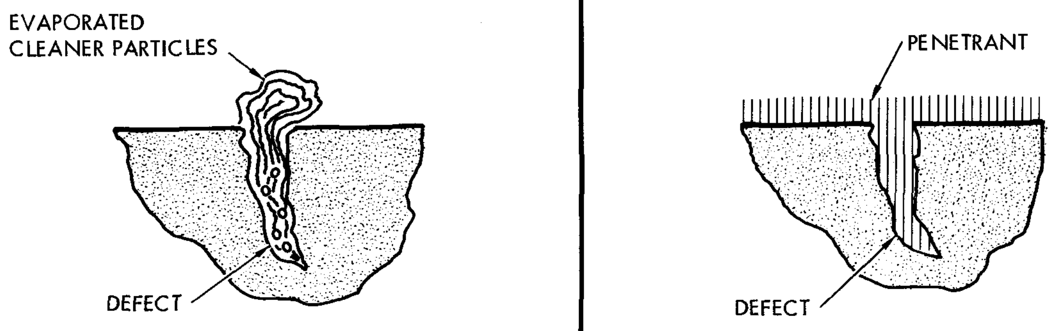
58.1.4.1 PURPOSE. The magnetic particle method of testing is effective only on materials having ferromagnetic properties, principally iron and steel. Flaws or cracks located on the surface, or interior flaws that extend outward to within approximately 1/4 inch of the part surface can be detected by this method. This method will not reveal defects located well beneath the surface. Testing by the magnetic particle method is based on the fact that magnetic lines of force, or flux, of an applied field tend to pass through the metal rather than through the air. If there is a surface crack or a defect near the surface, the distribution of the magnetic flux is disturbed and some of the flux is forced to pass out through the surface of the metal. Since the greater number of lines of force are cut by a flaw at right angles to the lines of force, the maximum amount of flux flow is obtained in the defect area. Thus in this area, the field strength is increased by the presence of the flaw and opposite magnetic poles form on either side of the defect. Then, when fine magnetic particles are applied to the part, the particles are attracted to these regions and form a pattern. In addition to cracks, the types of flaws that can be detected include seams, laps, nonmetallic inclusions near the surface, and folds. Defects that are large and open to the surface can be detected by the use of either AC or DC equipment. Small cracks, or near surface defects, are generally better detected by DC equipment since direct current will provide deeper penetration.

58.1.4.2 DEFECTS. To locate a defect, it is necessary for the magnetic flux to pass at approximately right angles to the defect. This is accomplished by control of the direction of magnetization. By controlling the direction of magnetization, the lines of force can be induced at right angles to the discontinuities. Since slight discontinuities can be oriented at any



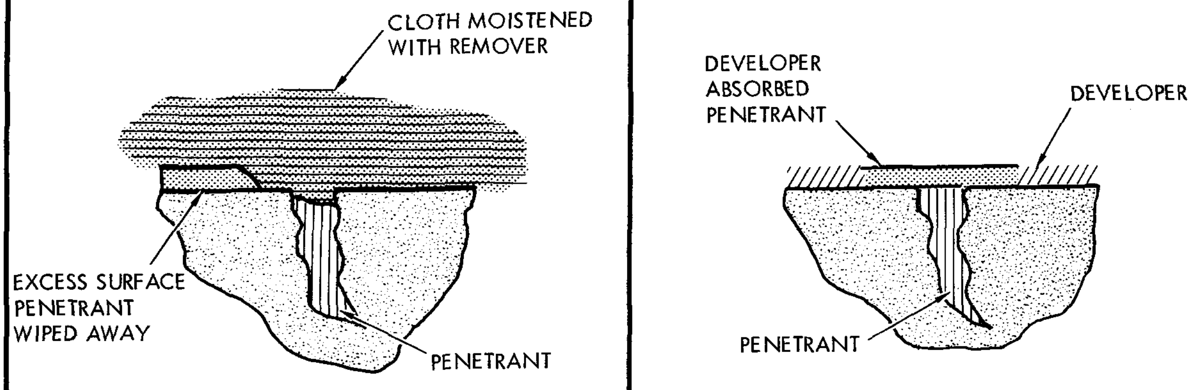
1. PART AND DEFECT SOILED

2. CLEANER PARTICLES LEFT AFTER PRE-CLEANING WHICH MAY CONTAMINATE PENETRANT



3. DRYING AFTER PRE-CLEANING EVAPORATES CLEANER PARTICLES

4. PENETRANT APPLIED


CLOTH MOISTENED WITH REMOVER
EXCESS SURFACE PENETRANT WIPED AWAY
PENETRANT
DEVELOPER ABSORBED PENETRANT
DEVELOPER
PENETRANT

5. SURFACE PENETRANT IS WIPED OFF

6. DEVELOPER APPLIED DRAWING OUT AND ABSORBING PENETRANT

SMR151-58.1-0-001

Figure 58.1-1. Fluorescent Penetrant Inspection

		INSTRUCTIONS
1.	PREPARATION OF PART:	Refer to specific procedure where applicable for preparation of part. Refer to Section 58.0 of this manual when required for cleaning and finish removal.
2.	PRE-CLEANING:	After the finish has been removed, pre-clean the area to be inspected with the solvent cleaner and wipe dry with a clean dry cloth. High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts.
3.	PENETRANT APPLICATION:	Apply penetrant either by brushing or spraying. In a confined area apply with a brush to prevent an overspray in the area to be inspected. If the temperature of the part and inspection material to be used in inspection are below 60°F, preheat and maintain the temperature at or above 60°F during applications of penetrant. The surface area may be heated with the black light or with hot air heaters. On all parts, minimum penetrant dwell time is 30 minutes.
4.	PENETRANT REMOVAL:	Use the solvent remover and a clean lint free cloth to remove the excess penetrant. Check the area to be inspected with the black light to assure that the excess penetrant has been removed prior to applying the developer.
5.	DEVELOPER:	Prior to application, shake developer can to mix developer thoroughly. Spray a light film of developer over the area to be inspected. Observe part under black light and allow a minimum developer dwell time of 15 minutes.
6.	MATERIALS:	Use the following materials or equivalent (in properties/characteristics, sulphur, and chlorine content). Penetrant: ZL-22A Remover: SKC-NF/ZC-7B Developer: ZP-9C

Figure 58.1-2. General Procedure for Fluorescent Penetrant Inspection

58.1.4.2 (Cont).

angle in the part, it is sometimes necessary to perform the operation in one direction and then magnetize in a direction 90 degrees from the first. Thus, two types of magnetization - circular and longitudinal, are used.

58.1.4.3 CIRCULAR MAGNETIZATION. (See figure 58.1-3.) Circular magnetization occurs when a circular magnetic field is induced in a part in such a manner that the magnetic lines of force (in any one plane normal to the axis of the current) take the form of concentric rings about the axis of the current. This is accomplished by passing the current directly through the part or through a conductor which passes into or through a hole in the piece.

CAUTION

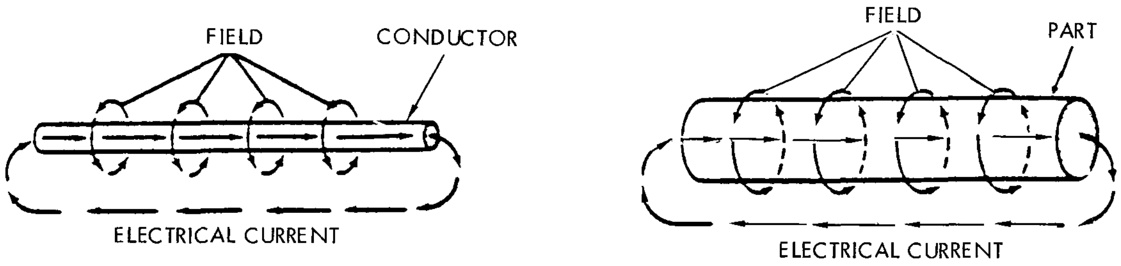
> Where it is necessary to pass current through the part itself, exercise care to prevent burning at the electrode contact areas. The points of contact made with the part must be electrically sound to prevent arcing, and subsequent damage to the part.

a. All contact areas shall be clean, parts shall be mounted between contact plates, and suitable pressure exerted to insure electrical contact. Use of copper braided pads between the electrodes and the part is recommended. When practical, large and heavy parts shall be mounted in suitable fixtures to insure proper orientation.

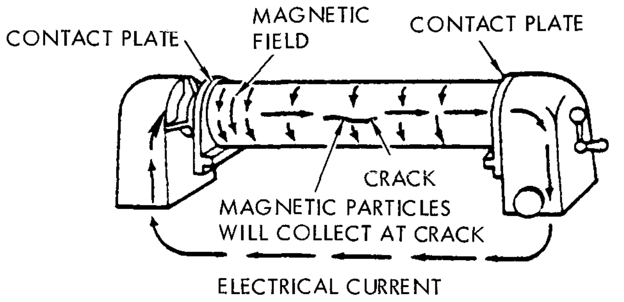
b. Magnetizing currents applied as "head shots" shall be a minimum of 1000 amperes per inch of outside diameter or per diagonal inch of the cross section perpendicular to the direction of current flow. Magnetizing currents through a central conductor shall be based on outside diameter measurement. If current requirements exceed the capacity of the equipment, reduce current, rotate and inspect through successive segments.

c. A central conductor is used in cases where inspection of the inside surfaces of cylindrically shaped parts is required. Whenever possible, inspect hollow parts on a copper or 1100 aluminum bar between electrodes, keeping the central conductor against one side of the part.

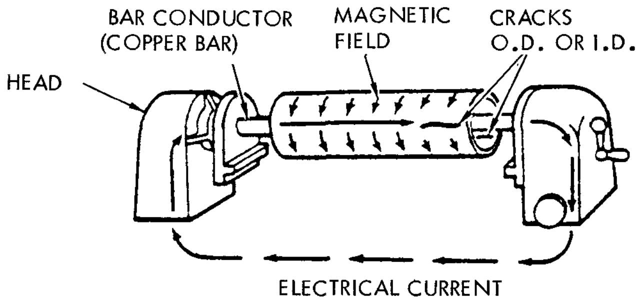
d. The conductors used should be as near the size of the inside diameter of the part as is possible. When it is not possible to fill the inside diameter of the part with a central conductor, inspect the entire circumference by repositioning the cylinder on the conductor, allowing for approximately 10 percent magnetic field overlap. The distance along the circumference effectively magnetized is approximately four times the diameter of the central conductor.



EXAMPLES OF CIRCULAR MAGNETIC FIELD SURROUNDING
A CONDUCTOR CARRYING AN ELECTRICAL CURRENT



CIRCULAR MAGNETISM USED TO DETECT
LONGITUDINAL CRACK IN SOLID TEST PART



CIRCULAR MAGNETISM USED TO DETECT
LONGITUDINAL CRACK IN HOLLOW TEST PART

SMR151-58.1-0-002

Figure 58.1-3. Circular Magnetization Applications

58.1.4.4 LONGITUDINAL MAGNETIZATION. (See figures 58.1-4 and 58.1-5.) When current is passed through a coil wrapped around a part, or when a part is placed between the poles of a magnet, lines of flux are created that are parallel to the axis of the part. This type of magnetization is called longitudinal magnetization and is useful for locating defects perpendicular to the axis of the coil. In establishing the correct indicated current in amperes to be used for any specific part, the following factors should be considered:

a. If the part is more than 18 inches long, calculate the L/D ratio (length-to-diameter ratio) using 18 inches as the length.

b. To find the L/D ratio divide the part length by the part diameter or width, as applicable.

c. When calculated L/D is less than 2, use a ratio of 2 and when the calculation exceeds 14, use 14 as the ratio.

d. To find the correct ampere turns, divide 45,000 by the L/D ratio.

e. To find the correct indicated current in amperes, divide the ampere turns by the number of turns in the coil to be used.

EXAMPLE:
L = 8, D = 2
L/D = 8/2 = 4
Ampere turns = 45,000/4 = 11,250
(Coil has 5 turns)
Indicated current = 11,250/5 = 2,250 amperes.

f. For long parts, each 15 inches of length must be magnetized and inspected progressively; approximately 6 inches on either side of the coil vertical centerline are effectively magnetized.

g. Leakage fields develop across sharp changes of section, such as radii under bolt heads, threads, and other sharp angles in parts. Keep longitudinal shots just below the intensity at which these leakage fields develop, except when inspecting ground chrome-plated parts for grind cracks.

58.1.4.5 WET CONTINUOUS METHOD. The magnetizing circuit shall be closed before the suspension is diverted from the part. A minimum of two shots of magnetizing current, each 1/2 to 1 second in duration shall be applied.

58.1.4.6 PARKER PROBE. The Parker Probe is a portable hand probe which is operated on 105 to 125 volts alternating current, 60 cycle, single phase, power supply. The maximum duty cycle ratio is 2 minutes "on," 2 minutes "off." This cycle applies to both the alternating current and direct current operations. Operate the Parker Probe as follows:

a. Place the selector switch in the AC or DC position in accordance with detail inspection procedure.

b. Place the pole pieces on the part with the suspected defect at right angles to the poles.

c. Press the "on" switch and magnetize the part.

d. Use the wet continuous method to inspect for defects.

e. Turn the poles 90 degrees and repeat steps c and d to locate defects oriented 90 degrees from the first inspection.

Note

The pole pieces are adjustable. If a large area is being inspected, a pole spacing of 6 inches to 8 inches is recommended.

f. Place the selector switch in the AC position to demagnetize the part.

g. Place the poles in the inspection position.

h. Press the "on" switch and withdraw the Parker Probe from the part a distance of 2 feet before turning it off.

58.1.4.7 DEMAGNETIZING. All parts shall be demagnetized after magnetic particle inspection. After demagnetization, test parts with a magnetic field indicator at several locations. Test parts of complex configuration at all significant changes in geometry. Demagnetized parts shall not produce deflections exceeding 2 units or about 3 oersteds on the field indicator after all demagnetization.

a. Using portable hand probe: The local areas inspected on the aircraft with the portable hand probe (Parker, P/N DA-200) can be demagnetized to a serviceable condition by placing the selector switch to the AC position and withdrawing the probe from the part for a distance of approximately 2 feet before turning it off. This operation may have to be repeated several times to reduce the residual magnetic field in the part to the lowest possible level.

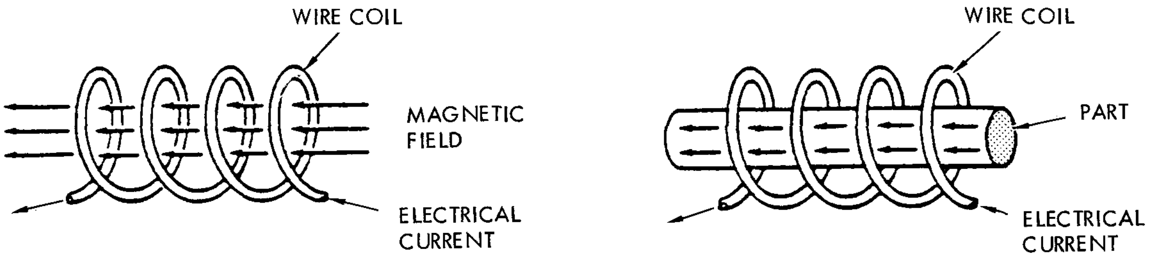
CAUTION

Care must be used in demagnetizing small parts using machines equipped with stepdown demagnetizers, which do not have adjustable tap switches. A small part such as a bolt being circularly demagnetized with this equipment may be overheated. Failure to comply may cause damage to the part being inspected.

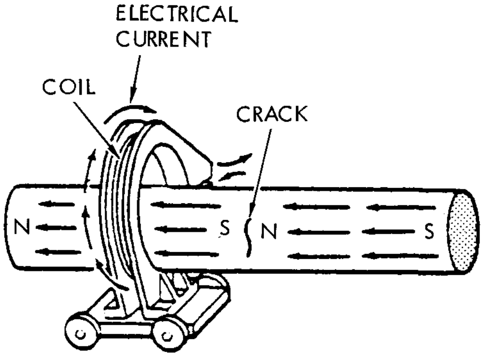
CAUTION

When using the direct induction method (head shot) to induce a circular magnetic field, care must be used to prevent arcing at the contact areas. Contact areas must be clean and suitable head pressure must be applied. Failure to comply may cause damage to equipment and part being inspected.

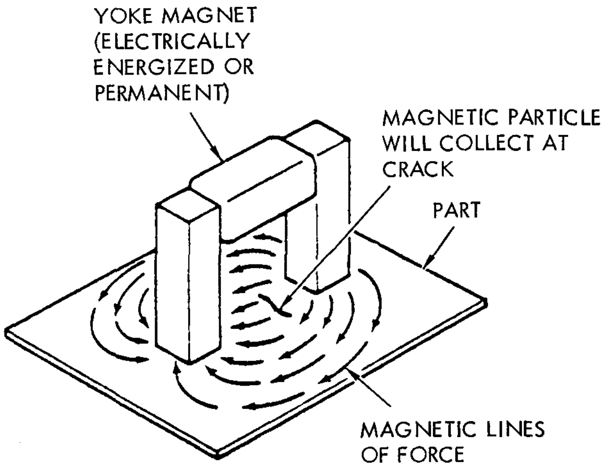
b. Using mobile or stationary equipment: The coil is considered to be the most effective method of demagnetizing the part. Parts that are inspected using DC may be difficult to demagnetize. The demagnetizing operation may have to be repeated with the magnetic poles in the part reversed with each pass through the coil.



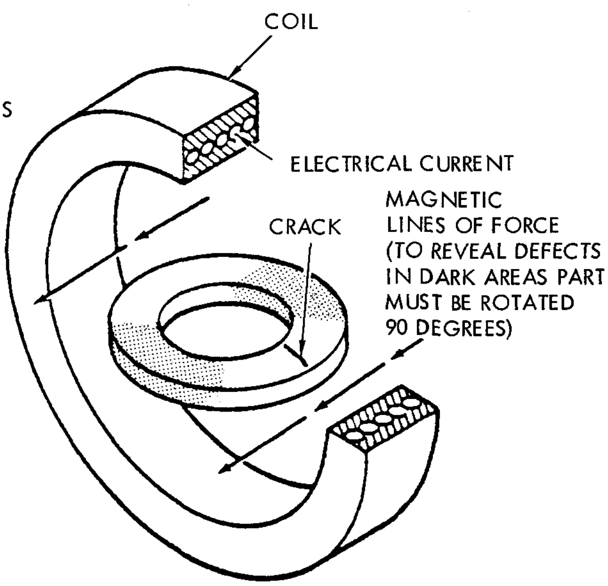
EXAMPLES OF LONGITUDINAL MAGNETIC FIELD AS PRODUCED BY A COIL



LONGITUDINAL MAGNETISM USED TO DETECT A RADIAL CRACK IN SOLID TEST PART


YOKE MAGNET (ELECTRICALLY ENERGIZED OR PERMANENT)
MAGNETIC PARTICLES WILL COLLECT AT CRACK
PART
MAGNETIC LINES OF FORCE

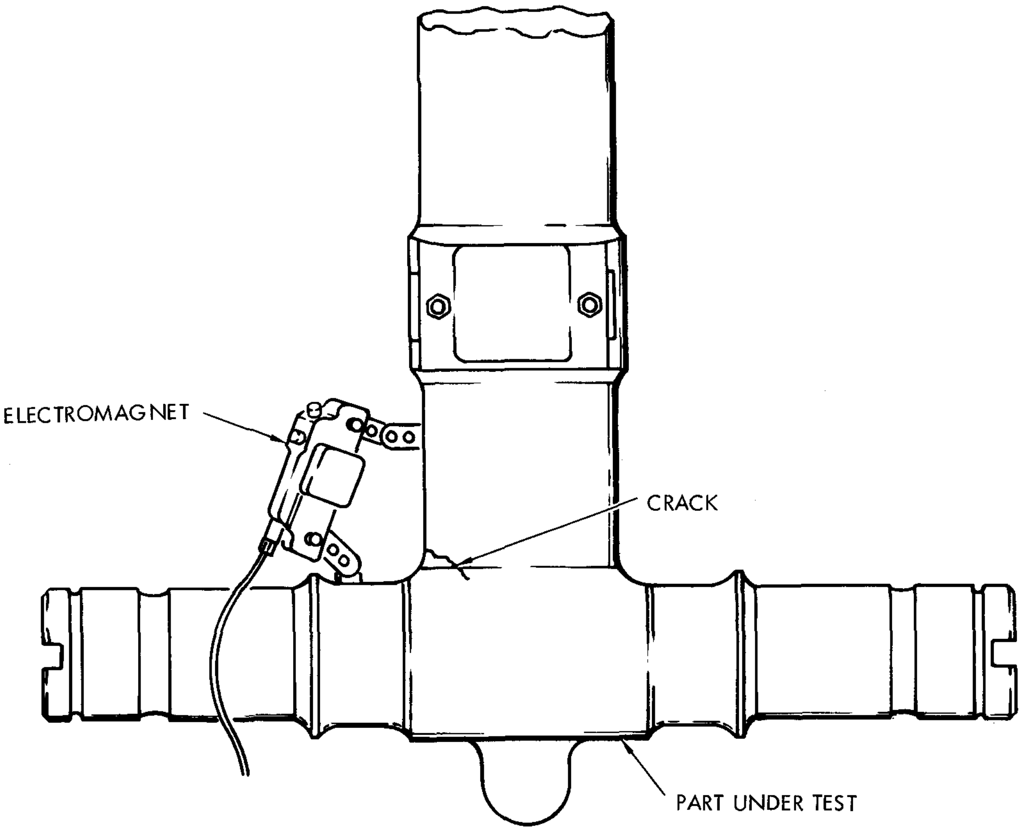
LONGITUDINAL LINES OF FORCE INDUCED IN A TEST PART BY A YOKE MAGNET


COIL
ELECTRICAL CURRENT
CRACK
MAGNETIC LINES OF FORCE (TO REVEAL DEFECTS IN DARK AREAS PART MUST BE ROTATED 90 DEGREES)

LONGITUDINAL MAGNETISM USED TO DETECT CRACKS IN A RADIAL PART

SMR151-58.1-0-003

Figure 58.1-4. Longitudinal Magnetization Applications



SMR151-58.1-0-004

Figure 58.1-5. Longitudinal Magnetization Application on Typical Landing Gear Part

58.1.4.7 (Cont).

c. Using automatic demagnetization equipment: Most large current units are equipped with an adjustable 30-point rotary tap switch that is used to select the demagnetization current. The most effective method of demagnetization is to select a current value of one step higher than the setting used to magnetize the particular component under test. Place the component in the coil and operate the demagnetization control switch. This will give a timed sequential reduction of the current, reducing the magnetic field to a minimum. If the residual field is not low enough, the demagnetization operation may be repeated.

58.1.4.8 MATERIALS. The materials listed in figure 58.0-4 are adaptable to various methods of magnetic particle inspection. The items are listed for convenience and equivalent substitutes may be used.

58.1.4.9 CONCENTRATION CONTROL. Magnetic particle suspension as delivered on the part requires testing at 8-hour intervals maximum by one of the methods specified in Specification MIL-I-6868. Contamination and water suspension tests shall be conducted at the intervals and by the method specified in Specification MIL-I-6868.

58.1.4.10 INSPECTION LIGHTS. The fluorescent particles used are visible under ultraviolet (black) light. The black light used should be of the same intensity and qualifications as described in the Fluorescent Penetrant Method.

58.1.4.11 TEMPERATURE LIMITATIONS. Most magnetic particle inspections are not affected by temperature except in the use of wet suspended particles. Near the freezing temperature, the magnetic particles may not be able to freely dispense within the suspension liquid. If the temperature is too hot the suspension liquid will evaporate, depositing the magnetic particles incorrectly.

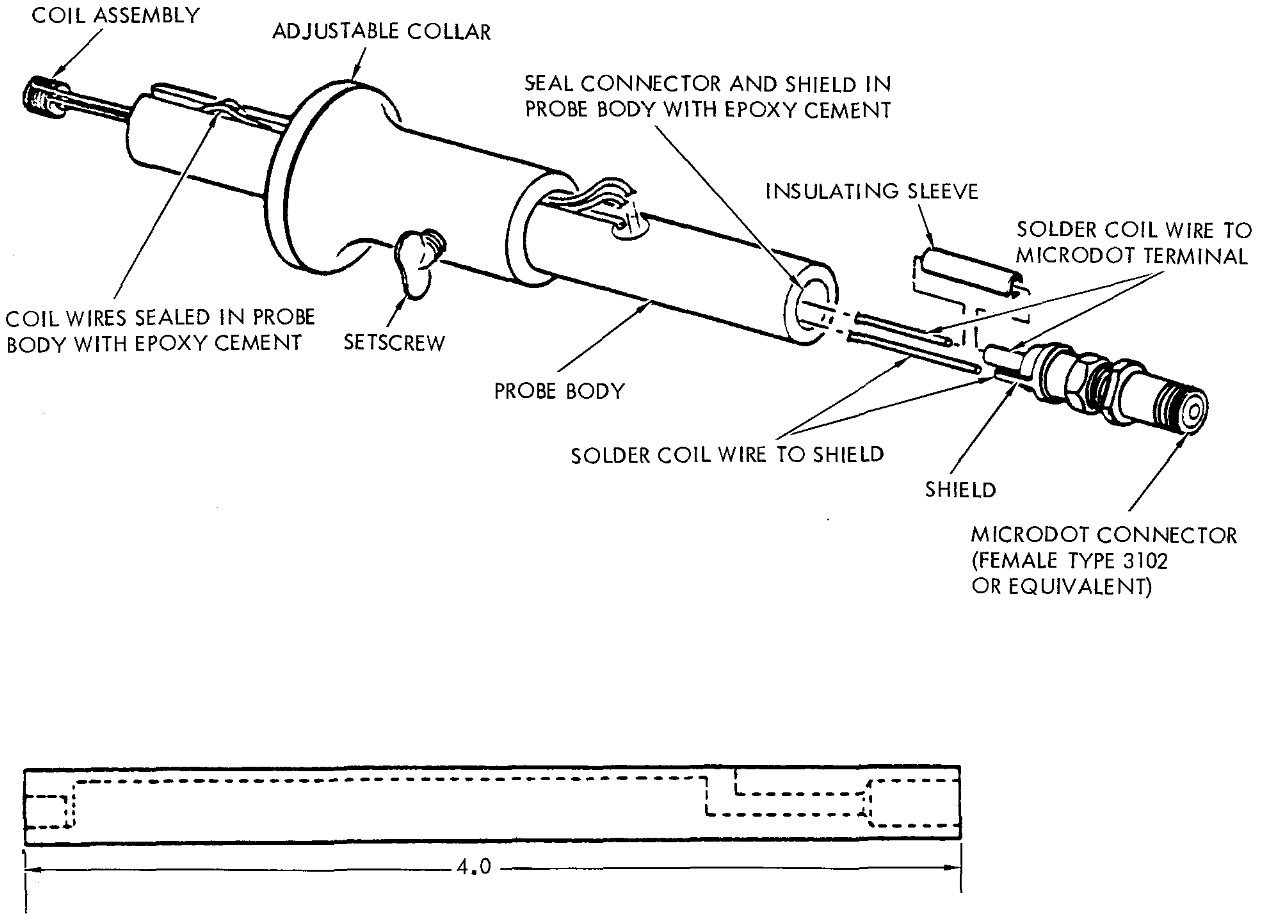
58.1.5 EDDY CURRENT METHOD.

58.1.5.1 PURPOSE. The eddy current method of testing is effective for detection of breaks or discontinuities in accessible metallic surfaces. In eddy currrent flaw detection and in other areas of eddy current material analysis, the electro-magnetic field which induces eddy current flow is produced by an alternating current in a small wire coil which is the eddy curent probe. The coil induces eddy currents by being in contact or close proximity to the test piece. The test piece must be a conductor of electricity. Eddy current inspection equipment consists of a probe and electronic instrument. The instrument contains a bridge circuit which measures the variations in the electric current flow and indicates these changes on a meter.

58.1.5.2 FLAW DETECTION METHODS. Typical eddy current probes are shown in figures 58.1-6 and 58.1-7. One probe shown is used for surface scanning and the other probe is used for hole examination. The principles of operation are the same for either type probe shown.

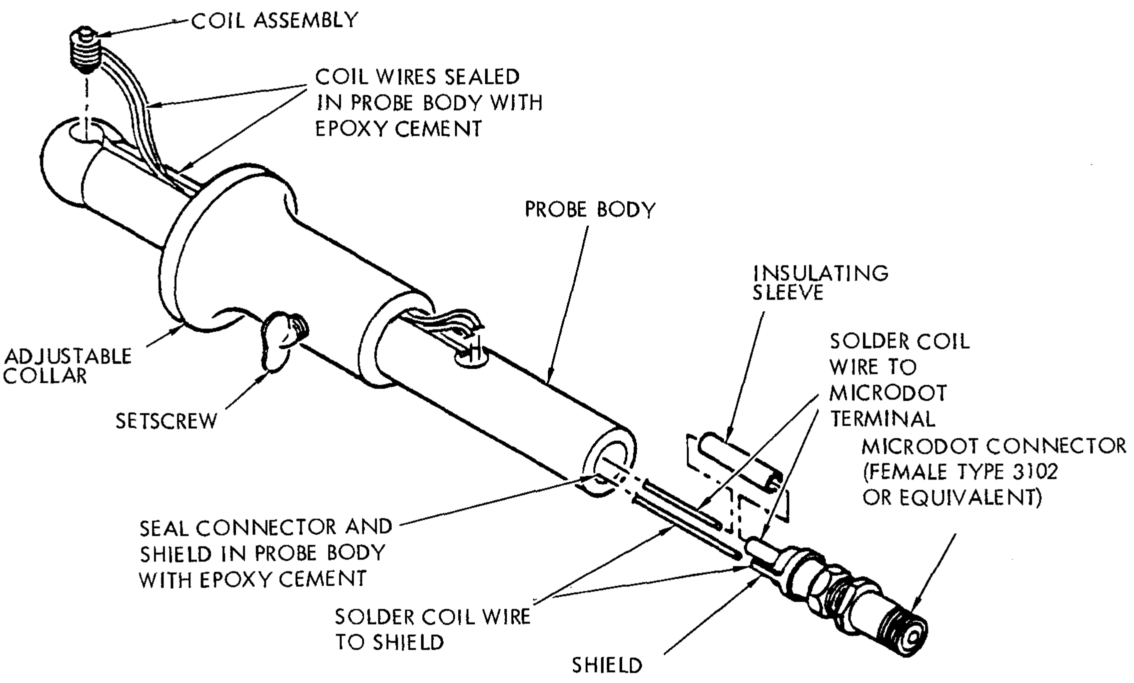
a. When the coil is energized, a small hemispherical magnetic field approximately of the same diameter as the coil is established. The characteristics of the coil-inductive reactance and ohmic resistance are balanced by adjustment of the bridge circuit in the test equipment. This balance is affected whenever material is placed within range of the hemispherical magnetic field. The degree of effect the material has upon the balance of the coil's characteristics varies proportionately to the conductivity or permeability of the material. The effect of the material may be nullified by further balance of the bridge while the probe is held in contact with sound material. The bridge circuit will then remain in balance as the surface is scanned, provided other disturbances (surface roughness defects, and conductivity variation) are not encountered.

b. The remaining disturbing influences are surface roughness, material defects, and changes in conductivity. Of these, surface roughness can be nullified by further balance adjustments, leaving only material defects or change in conductivity to be interpreted by the operator of the equipment. Since defects cause an abrupt deflection of the indicator, it is relatively easy to differentiate between abrupt deflections caused by defects and the smooth deflections caused by conductivity changes.


COIL ASSEMBLY
ADJUSTABLE COLLAR
SEAL CONNECTOR AND SHIELD IN PROBE BODY WITH EPOXY CEMENT
INSULATING SLEEVE
SOLDER COIL WIRE TO MICRODOT TERMINAL
COIL WIRES SEALED IN PROBE BODY WITH EPOXY CEMENT
SETSCREW
PROBE BODY
SOLDER COIL WIRE TO SHIELD
SHIELD
MICRODOT CONNECTOR (FEMALE TYPE 3102 OR EQUIVALENT)
4.0

SMR151-58.1-0-005

Figure 58.1-6. Typical Eddy Current Surface Probe


COIL ASSEMBLY
COIL WIRES SEALED IN PROBE BODY WITH EPOXY CEMENT
PROBE BODY
INSULATING SLEEVE
SOLDER COIL WIRE TO MICRODOT TERMINAL
MICRODOT CONNECTOR (FEMALE TYPE 3102 OR EQUIVALENT)
ADJUSTABLE COLLAR
SETSCREW
SEAL CONNECTOR AND SHIELD IN PROBE BODY WITH EPOXY CEMENT
SOLDER COIL WIRE TO SHIELD
SHIELD

SMR151-58.1-0-006

Figure 58.1-7. Typical Eddy Current Hole Probe

58.1.5.3 CALIBRATION OF MAGNAFLUX P/N ED-520. The Magnaflux P/N ED-520 (figure 58.1-8) is a portable, battery operated, variable frequency eddy currrent instrument which may be operated from self-contained rechargeable batteries or from 115 or 230 VAC, 50/60 Hz power. The unit has an operational range from 55 KHz to 200 KHz and is utilized for performance of inspection for crack detection in aluminum and steel. Calibrate the Magnaflux P/N ED-520 instrument as follows:

CAUTION

For maximum instrument stability and battery life, do not operate the ED-520 when the meter indication is below the red line in the test position.

a. Set the mode switch (1) to the TEST position. Meter indication should be above the red line. A meter reading below the red line indicates that recharging of batteries is required, while fully charged batteries will give a reading substantially above the red mark, approximately 450 on the meter scale.

b. Connect the probe or test coil to the appropriate front panel PROBE connector (7).

c. Turn the mode switch to the LO sensitivity position.

d. Set the LIFT-OFF/FREQ control (3) and the BALANCE control (2) to zero (fully counterclockwise).

e. Place the probe on the surface of the calibration standard (figure 58.1-9) which is representative of the metal to be inspected.

f. Rotate the BALANCE control clockwise until the meter needle comes on scale.

Note

If the meter needle does not come on scale by the time the BALANCE control is rotated fully clockwise, leave BALANCE control at full CW position and rotate the LIFT-OFF/FREQ control until the meter needle comes on scale.

Note

If the BALANCE control reaches the point where it can no longer be rotated in the direction required to keep the meter needle on scale, continue rotating the LIFT-OFF/FREQ control clockwise until the meter needle goes completely off scale and then returns to scale. This maneuver reverses the direction of the meter needle and the direction of rotation of the BALANCE control required to keep the meter needle on scale.

g. Rotate the LIFT-OFF/FREQ control clockwise while keeping the meter needle on scale with the BALANCE control until a null point is reached. A null point occurs when the meter needle reverses direction while the LIFT-OFF/FREQ control is being rotated in one direction.

h. After the null point is reached, bring the meter needle to mid-scale using the BALANCE control.

i. Place a piece of ordinary Scotch tape (0.001 to 0.003-inch thick) on one corner of the standard away from any defects (figure 58.1-10), and place the probe over the tape.

j. Using the BALANCE control, bring the meter needle to mid-scale.

k. Move the probe off the tape to an area on the standard without tape and note the meter needle deflection.

l. Place the probe back on the tape and rotate the LIFT-OFF/FREQ control counterclockwise for aluminum, and clockwise for steel, to move the needle approximately the same amount as deflection in step k. Use the BALANCE control to keep the meter needle on scale.

m. Repeat steps k and l until the meter needle reads the same when the probe is on and off the tape.

n. Turn mode switch to MED sensitivity position and repeat step m.

o. Turn mode switch to HI sensitivity position and repeat step m.

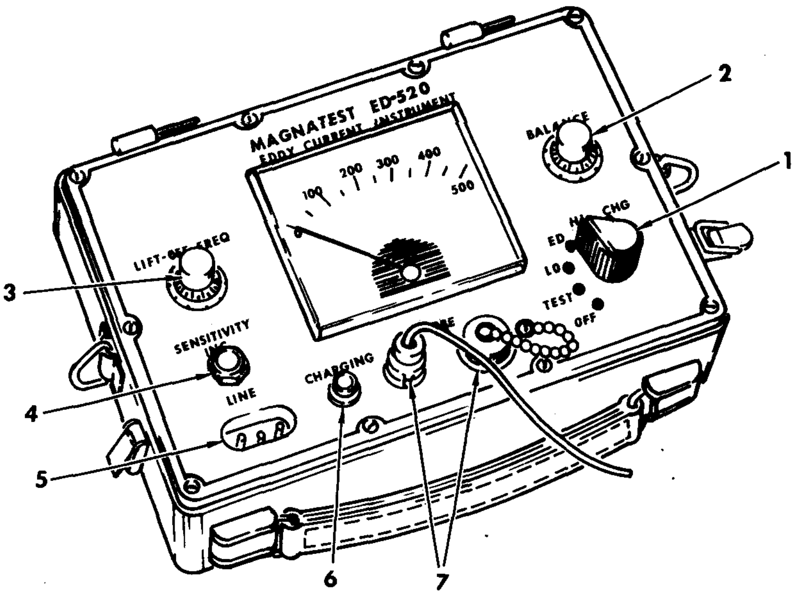
p. To determine whether the probe and instrument are functioning adequately, scan the 0.020-inch deep surface slot in the aluminum calibration standard and the 0.050-inch deep surface slot in the magnetic steel calibration standard with instrument in LO position. A clearly recognizable meter deflection of at least 10 increments of the full scale for aluminum and steel should result. If no deflection occurs of at least that specified above, the instrument and/or probe are not functioning properly.

q. With a properly operating instrument and probe adjusted to the correct LIFT-OFF/FREQ, scan the area of the calibration standard which approximates the area to be inspected in the specific procedure. For example, if a fillet radius inspection is specified, scan along the fillet radius of the calibration standard noting the meter deflection caused when the simulated cracks (transverse and longitudinal to the fillet radius) are scanned.

Note

When two or more probes are to be used in performing a specific inspection procedure, crack detector must be recalibrated at the time of changing each of the probes, regardless of frequency.

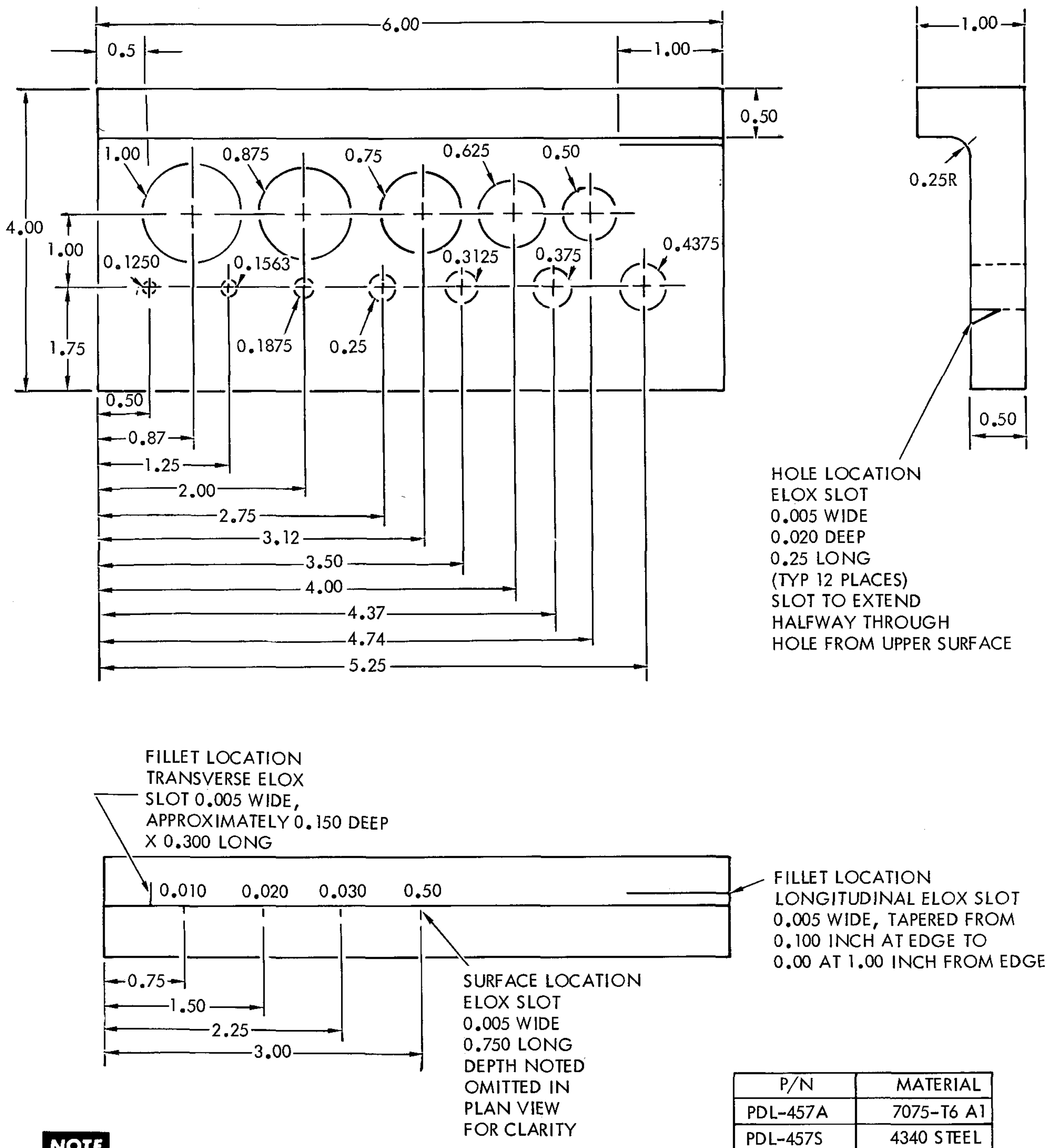
r. When the instrument is properly calibrated on the standard, place the probe on the part being inspected. It may be necessary to bring the needle back on scale with the balance control. Lock the LIFT-OFF/FREQ control in position and begin inspecting the part per the inspection procedure.



1. MODE SWITCH
2. BALANCE CONTROL
3. LIFT-OFF/FREQ CONTROL
4. SENSITIVITY CONTROL
5. LINE CONNECTOR
6. CHARGING LIGHT
7. PROBE CONNECTORS

SMR151-58.1-0-007

Figure 58.1-8. Eddy Current Inspection Instrument



HOLE LOCATION
ELOX SLOT
0.005 WIDE
0.020 DEEP
0.25 LONG
(TYP 12 PLACES)
SLOT TO EXTEND
HALFWAY THROUGH
HOLE FROM UPPER SURFACE

FILLET LOCATION
TRANSVERSE ELOX
SLOT 0.005 WIDE,
APPROXIMATELY 0.150 DEEP
X 0.300 LONG

FILLET LOCATION
LONGITUDINAL ELOX SLOT
0.005 WIDE, TAPERED FROM
0.100 INCH AT EDGE TO
0.00 AT 1.00 INCH FROM EDGE

SURFACE LOCATION
ELOX SLOT
0.005 WIDE
0.750 LONG
DEPTH NOTED
OMITTED IN
PLAN VIEW
FOR CLARITY

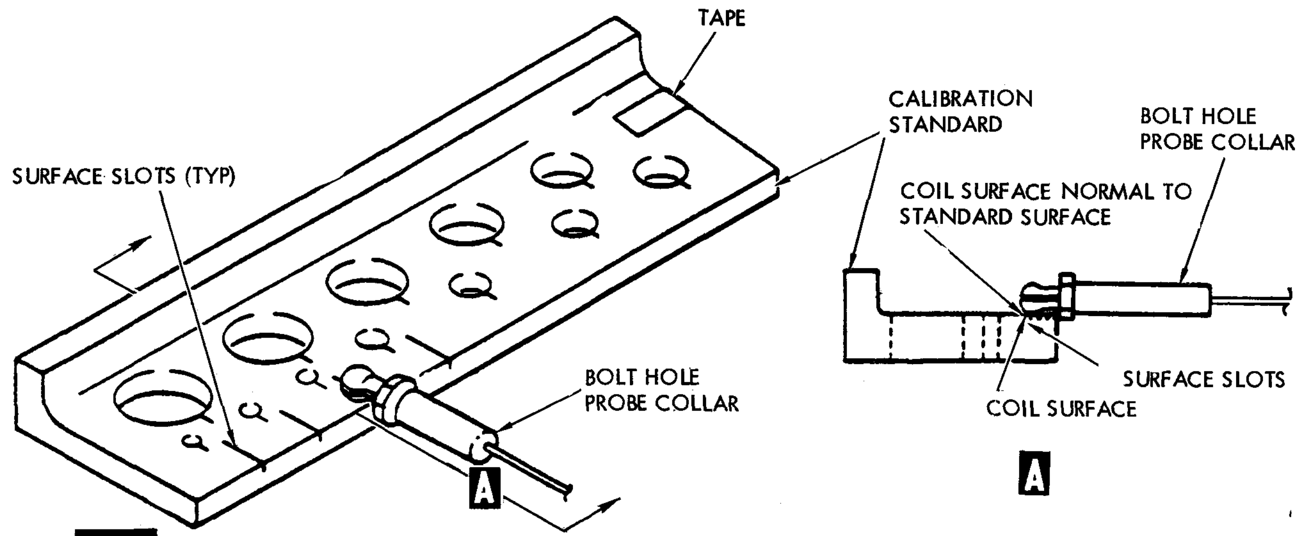
P/N	MATERIAL
PDL-457A	7075-T6 Al
PDL-457S	4340 STEEL

NOTE

SURFACE FINISH 63 RMS.

SMR151-58.1-0-009

Figure 58.1-9. Eddy Current Calibration Standard



NOTE

THE CORRECT POSITION TO CALIBRATE IS FOUND BY PLACING THE PROBE COLLAR FLUSH AGAINST THE EDGE OF THE STANDARD SO THAT THE PROBE COIL IS NORMAL TO THE SURFACE. TO DETERMINE IF THE COIL IS EXACTLY NORMAL, ROTATE THE PROBE SLIGHTLY (WHILE OBSERVING THE METER NEEDLE) UNTIL THE NEEDLE REACHES A MAXIMUM POINT AND BEGINS TO MOVE IN THE OPPOSITE DIRECTION. THE POSITION OF THE PROBE WHERE THE NEEDLE REVERSES DIRECTION IS THE CORRECT POSITION FOR CALIBRATION.

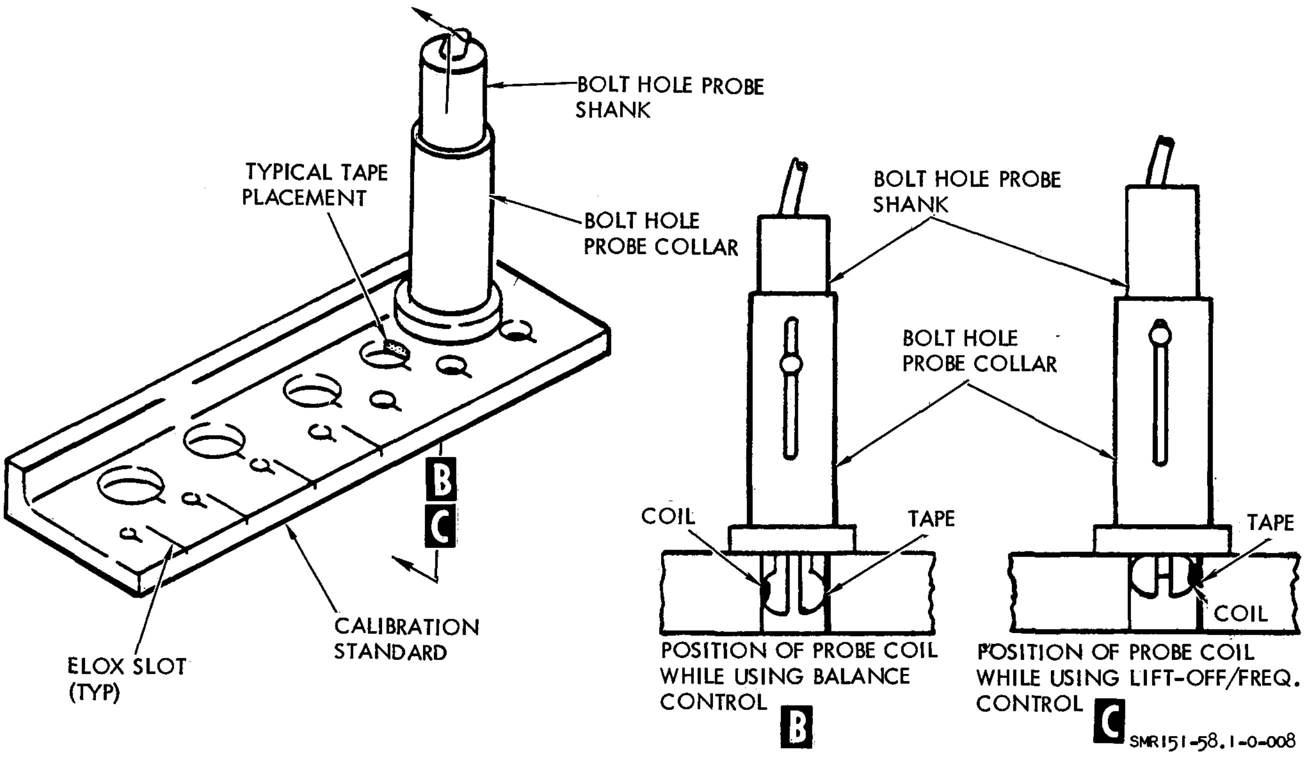


Figure 58.1-10. Eddy Current Calibration Setup

58.1.5.4 SCANNING TECHNIQUES. Eddy current inspection procedures throughout this manual contain instructions that are specifically defined in the following paragraphs.

Note

Shielded eddy current probes are now commercially available in general purpose and bolt hole models. The smaller magnetic field area minimizes the effects of edge distance and permeability effects of steel fasteners. Hole noise is also minimized in bolt holes. These probes are preferred for scanning around and between fasteners and along edges of parts. They may be used without probe guides or templates required in the following notes and paragraphs. They also require slower scanning speeds. Practice on standards prior to use on aircraft.

Note

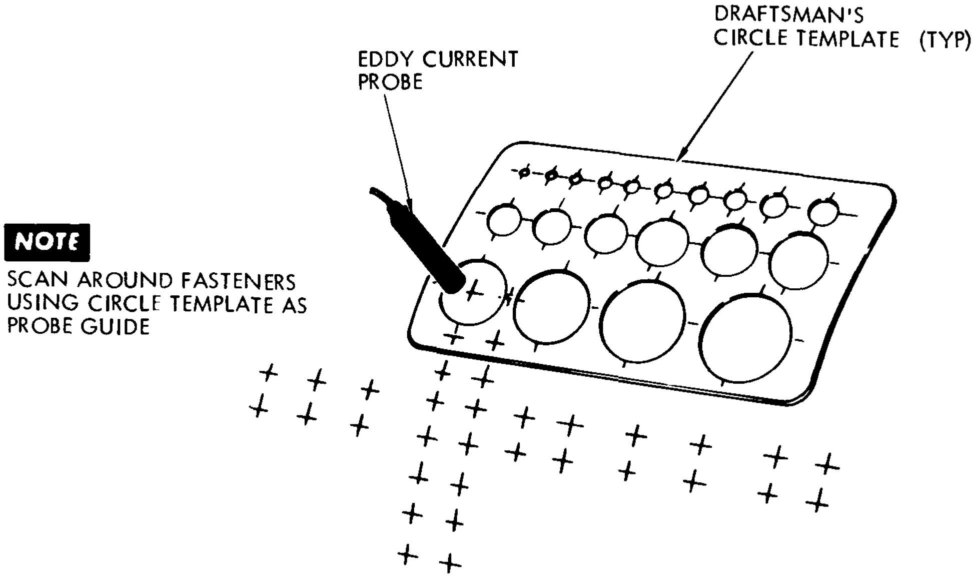
When using a 1/8-inch diameter general purpose probe with the ED-520 to inspect an area on an aluminum part near steel fasteners or steel bushing, a 1/8-inch clearance between the steel part and the probe must be maintained at all times. Using a small magnet will enable the operator to determine whether the fastener is steel. Whenever performing an inspection around steel fasteners, use a probe guide as illustrated in figure 58.1-11 or 58.1-12. When using the probe guides shown, the needle is brought to mid-scale with the BALANCE control while the probe is on the part. Slight deflections may be noted if the guide is not centered around the fastener.

Note

Because of the permeability characteristics present in magnetic steel alloys, greater care must be taken when inspecting steel components by eddy current methods. Permeability effects are not constant throughout these materials and therefore may result in slight instrument variations between adjacent areas as well as between different steel alloys. To minimize the effects of permeability it is necessary to calibrate the instrument to a high gain setting (HI on ED-520) and then decrease the setting to MED position when inspecting the part.

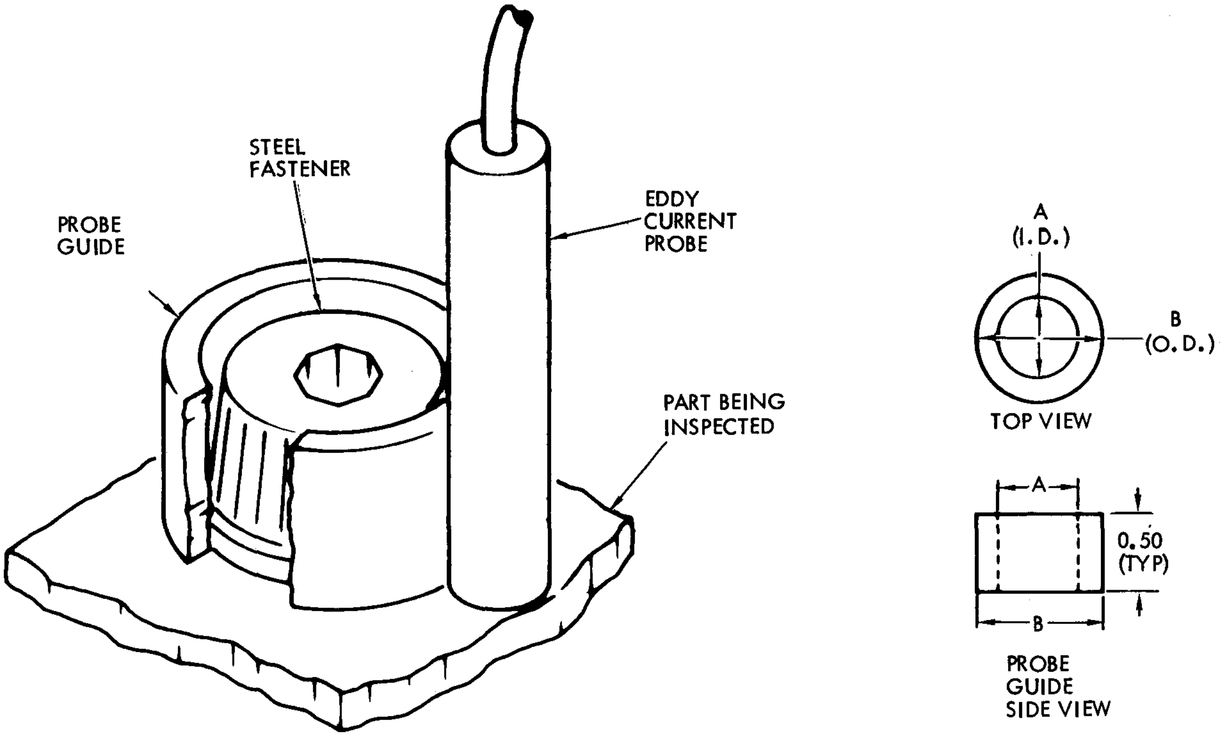
Note

When inspecting aluminum alloys, inspection may be performed with the instrument MODE selector switch in the MED or HI position, depending on the surface condition of the part being inspected. Rough surface conditions of radius inspection causes large needle fluctuations in the HI position.


DRAFTSMAN'S CIRCLE TEMPLATE (TYP)
EDDY CURRENT PROBE
NOTE
SCAN AROUND FASTENERS USING CIRCLE TEMPLATE AS PROBE GUIDE

SMR151-58.1-0-010

Figure 58.1-11. Probe Guide for Scanning Around Flush Mounted Fasteners



MATERIAL: PLASTIC, PHENOLIC, OR OTHER NON-CONDUCTIVE MATERIAL

PROBE GUIDE DIMENSIONS	
A. (I.D.)	B. (O.D.)
1/4	0.40
5/16	0.46
3/8	0.52
7/16	0.58
1/2	0.65
9/16	0.71
5/8	0.77
11/16	0.83
3/4	0.90

SMR151-58.1-0-011

Figure 58.1-12. Probe Guide for Scanning Around Protruding Fasteners

58.1.5.4 (Cont).

Note

> Special care must be taken by the operator to ensure that the probe position is normal to the inspection surface to avoid possible lift-off effect during inspection. The ability to hold the probe in a steady position will depend upon the competence and experience of the operator. Parts to be inspected by the eddy current method must be cleaned free of dirt, grease, and other contaminants to permit good contact between parts and probe.

a. Scanning around flush mounted fasteners. It is mandatory to maintain the probe-to-fastener spacing to obtain the best inspection when scanning around fasteners. It is recommended that a minimum of 1/8-inch spacing be maintained at all times when scanning around all fasteners (steel, monel, etc.) except aluminum or titanium. No minimum spacing is required for aluminum or titanium fasteners, but whatever spacing is used must be maintained uniformly. To facilitate inspections of flush mounted fasteners, a draftsman's circle template is to be used as a probe guide. (See figure 58.1-11.)

b. Scanning around protruding fasteners. After determining that fasteners are steel on a nonferrous part, a shielded probe or an appropriate guide is to be used to scan around or between fasteners. The shield probe or probe guide will provide sufficient spacing between the fastener and eddy current probe to preclude erroneous crack indications which would otherwise be caused by too close inspection around steel fasteners in nonferrous parts. Fit probe guide snugly around the fastener. (See figure 58.1-12.)

c. Scanning between fasteners. Scanning between fasteners is identical to scanning around fasteners, except that areas made inaccessible by part configuration or close fastener patterns are omitted. (See figure 58.1-13.) The phrase "scanning between fasteners" is used only if the eddy current probe can inspect all locations where cracks are probable.

d. Scanning fillet radii. This item requires multi-directional scanning of the inspection area, i.e., make several scans in the fillet radii in both the longitudinal and transverse directions where accessible. (See figure 58.1-14.)

e. Scanning bolt holes. (See figure 58.1-15.) Where an inspection procedure specifies a bolt hole scanning of layers of similar metals but does not designate the depth at which scans are to be made, the following general method is to be used. For bolt hole scanning of layers of dissimilar metals, scan depths will be provided in the specific procedure.

(1) Select and calibrate the appropriate bolt hole probe for the material to be inspected in accordance with the appropriate calibration procedure.

(2) Insert the bolt hole probe into the bolt hole where the probe coil first comes in contact with the metal. Place the probe collar gainst the part being inspected and mark where it stops on the shank of the bolt hole probe.

(3) Move the probe through the bolt hole until the meter indicates that the probe coil has been moved out of the hole. Place the probe collar aginst the part being inspected and mark where it stops on the shank of the bolt hole probe.

(4) Back the probe out of the hole to the location where the probe coil first contacted metal and advance the probe into the hole until the meter indicates that an interface between two of the mating parts has been reached. The precise location of the interface can be found by moving the probe back and forth across the interface until the depth at which the meter deflection changes direction has been found. Move the collar in place and mark on the shank of the probe the location of the interface. Repeat this step until all interfaces have been found and marked on the probe shank. Using these marks, inspect all the mating parts at depths of 1/8-inch increments by rotating the probe 360 degrees in both the clockwise and counterclockwise directions.

58.1.5.5 EQUIPMENT. The items listed in figure 58.0-4 are adaptable to various methods of eddy current inspection. The equipment is listed for convenience and equivalent substitutes may be used. Some surfaces designated for eddy current inspection cannot be inspected using a standard general purpose probe or a standard bolt hole probe. In such cases, alternate probes are provided, such as: right angle 1/8-inch diameter probe and miniature bolt hole probes of various sizes. The inspector will select the best probe configuration for inspecting the area in question.

58.1.5.6 INSPECTION PROCEDURE. When conducting eddy current inspection in fuel tank areas, the aircraft must be defueled and purged. Holes and surfaces to be inspected must be clean, free of dirt, grease, and loose paint. Remove all burrs and lapped material.

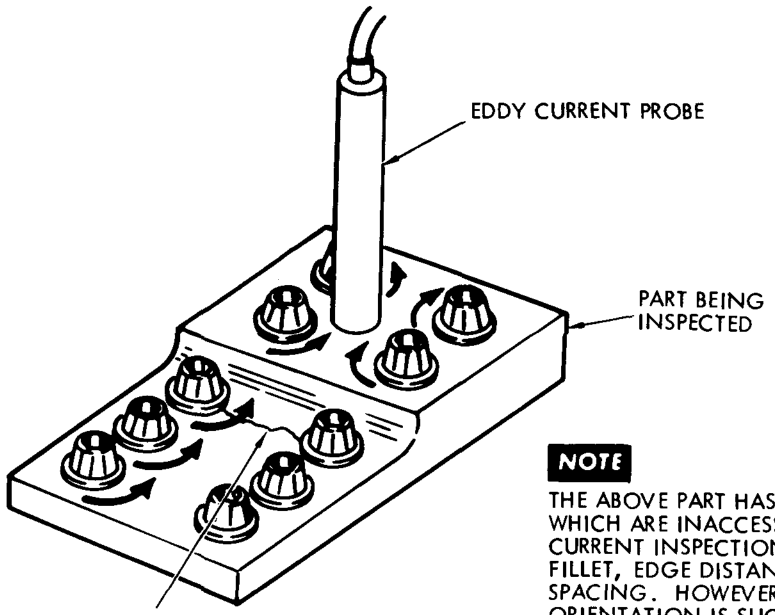
a. Calibrate eddy current instrument in accordance with the instructions contained in this section of the manual.

b. Inspect the airplane parts in accordance with specified inspection procedures and instructions contained in this section of the manual.

c. If necessary, minor adjustments of the BALANCE control can be made to move the needle to mid-scale when the coil is moved close to such things as faying surface, edge of hole, edge of part, or thickness change in part.

d. An indication is noted by a sharp deflection of the needle. To pinpoint the indication, move the coil back and forth, gradually decreasing the movement until a maximum needle deflection is obtained. The indication will be at the center of the coil.

58.1.5.7 TEMPERATURE LIMITATIONS. Most eddy current inspections are not affected by anything other than extreme high or low temperature. A decrease in electrical conductivity is obtained with increasing temperature and an increase in electrical conductivity results from decreasing temperatures. Temperature variations affect the inductance of the test coil. Differences between test coil temperature, test object temperature and test area temperature can lead to unstable eddy current readings. Therefore time should be allowed for temperature stabilization.

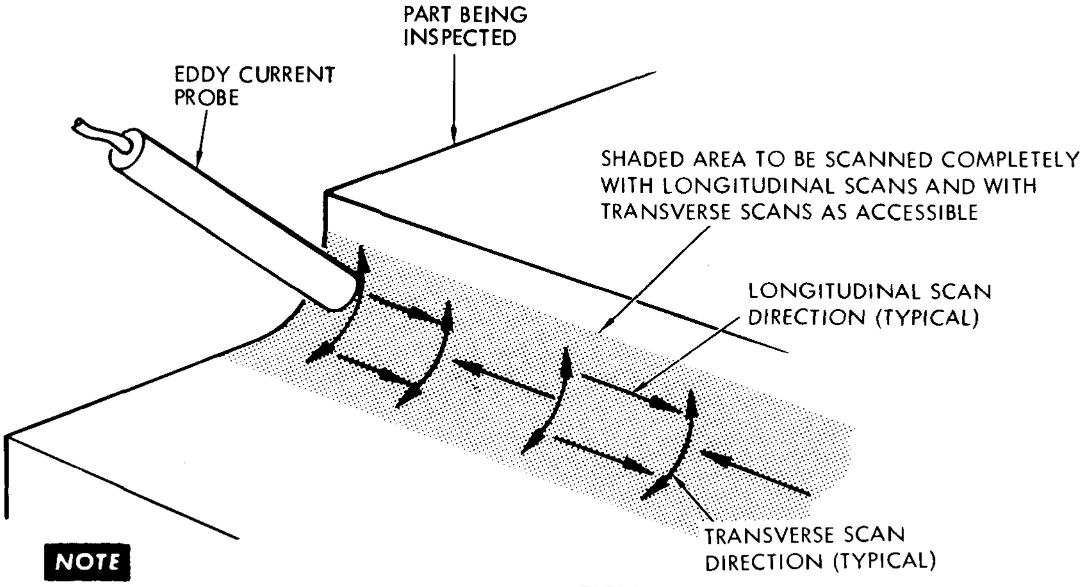


NOTE

THE ABOVE PART HAS CERTAIN AREAS WHICH ARE INACCESSIBLE TO EDDY CURRENT INSPECTION BECAUSE OF THE FILLET, EDGE DISTANCE, OR FASTENER SPACING. HOWEVER THE CRACK ORIENTATION IS SUCH THAT SCANNING BETWEEN THE FASTENERS IS ADEQUATE.

SMR151-58.1-0-012

Figure 58.1-13. Scanning Between Fasteners

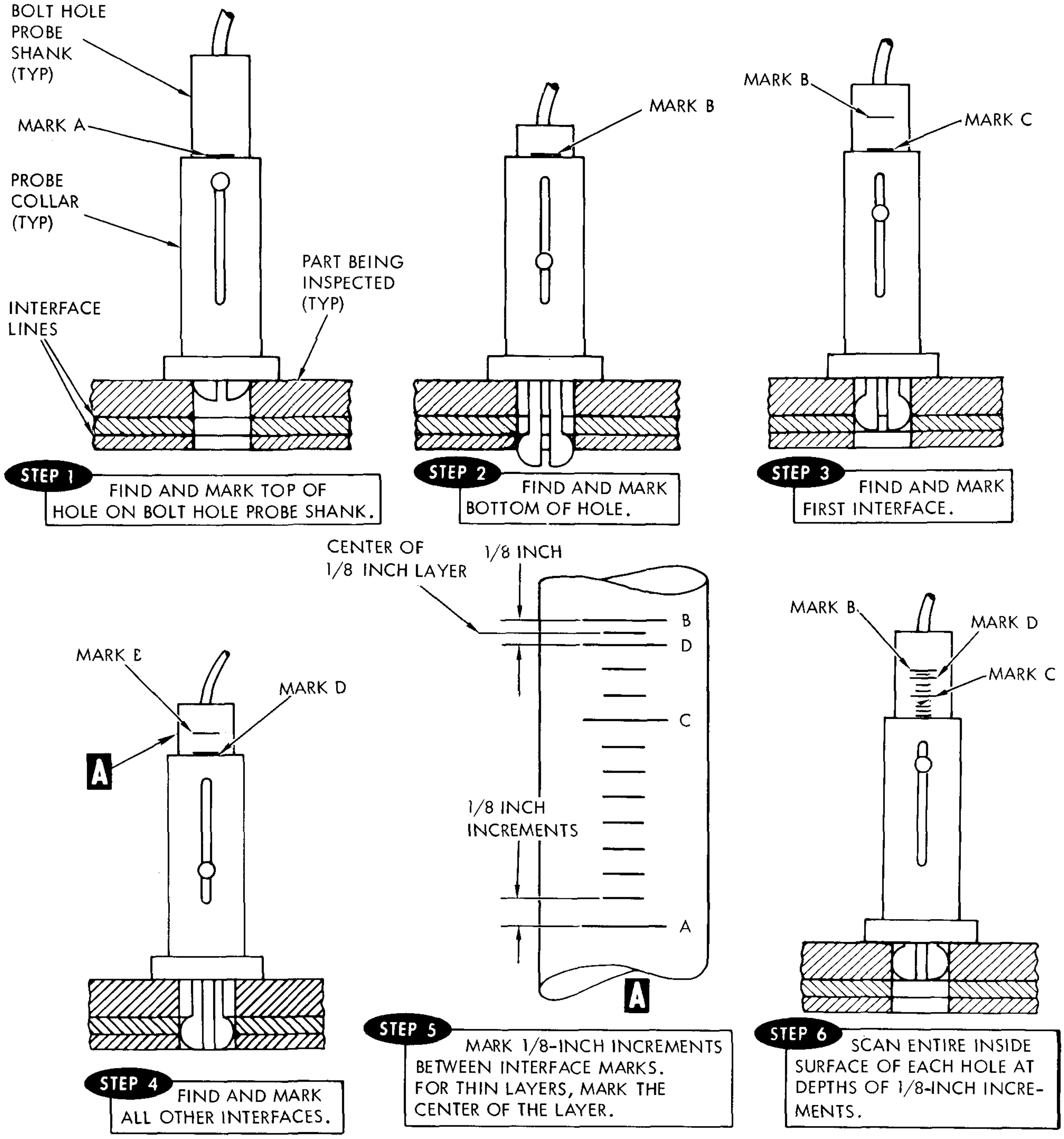

PART BEING INSPECTED
EDDY CURRENT PROBE
SHADED AREA TO BE SCANNED COMPLETELY WITH LONGITUDINAL SCANS AND WITH TRANSVERSE SCANS AS ACCESSIBLE
LONGITUDINAL SCAN DIRECTION (TYPICAL)
TRANSVERSE SCAN DIRECTION (TYPICAL)

NOTE

FILLET RADII MUST BE INSPECTED BY MULTI-DIRECTIONAL SCANS USING RADIUS PROBE OR 1/8-INCH DIAMETER GENERAL PURPOSE PROBE. THE SCANS, TRANSVERSE TO THE FILLET, ARE TO BE MADE ONLY WHERE THE SIZE AND SHAPE OF THE FITTING PERMIT, ESPECIALLY IN LARGER FILLET RADII.

SMR151-58.1-0-013

Figure 58.1-14. Scanning Fillet Radii



STEP 1 FIND AND MARK TOP OF HOLE ON BOLT HOLE PROBE SHANK.

STEP 2 FIND AND MARK BOTTOM OF HOLE.

STEP 3 FIND AND MARK FIRST INTERFACE.

STEP 4 FIND AND MARK ALL OTHER INTERFACES.

STEP 5 MARK 1/8-INCH INCREMENTS BETWEEN INTERFACE MARKS. FOR THIN LAYERS, MARK THE CENTER OF THE LAYER.

STEP 6 SCAN ENTIRE INSIDE SURFACE OF EACH HOLE AT DEPTHS OF 1/8-INCH INCREMENTS.

NOTE

1. THIS PROCEDURE IS TO BE USED ONLY FOR INSPECTING HOLES WHERE ALL LAYERS OF MATERIAL ARE THE SAME BASIC MATERIAL. WHERE DISSIMILAR METALS COMPRISE A SINGLE BOLT HOLE, THE SCAN DEPTHS WILL BE SHOWN ON THE SPECIFIC PROCEDURE.

2. INTERFACES CAN BE DETECTED BY MOVING THE PROBE THROUGH THE HOLE AND NOTING WHERE THE METER NEEDLE MOVES OFF-SCALE. THE POINT AT WHICH THE METER NEEDLE MOVEMENT CHANGES DIRECTION IS THE DEPTH INTO THE HOLE WHERE THE INTERFACE IS LOCATED.

SMR151-58.1-0-014

Figure 58.1-15. Scanning Bolt Holes

58.1.6 ULTRASONIC METHOD.

58.1.6.1 PURPOSE. Ultrasonic inspection uses high-frequency sound waves as a probing medium to provide information as to the state of various materials. This method is effective for inspection of most metals for surface and sub-surface defects. The method requires that at least one surface of the part be accessible for transducer contact in the vicinity of the area to be examined. The inspection is accomplished by inducing the ultrasound into the part by a contacting transducer and picking up reflections of this sound from within the part. The detected ultrasonic reflections are electronically displayed on an oscilloscope and interpreted for indications of defects. Accessory devices (wedges) are designed and fabricated to provide adequate transducer coupling to curved surfaces or to change the angle of the sound beam.

58.1.6.2 FLAW DETECTOR EQUIVALENCY. An ultrasonic flaw detector equivalency is determined by its capability to detect defects as presented by defined calibration standards. Each item of test equipment will have special features by design, with some degree of variations as to integrity for input/output signals. In some instances the portability and/or physical size will determine what test equipment will be used for an inspection. The equivalency of the equipment must be determined by actual test using the calibration standard for each category of testing. Figure 58.1-16 provides a comparison of each ultrasonic instrument's control settings currently in use for nondestructive inspections.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. If compensation for the difference in sensitivity cannot be accomplished, use the standard calibration or the calibration compensation required for previous similar fastener hole, whichever is higher, and inspect fastener hole, using the same relative positions as required on the calibration standard for top and bottom saw cuts.

Note

Some areas may have uneven paint coating caused by runs in the paint. These may be smoothed using Scotch Brite or 400 grit abrasive. Areas where the top coat is removed must be repaired after the inspection.

58.1.6.3 ULTRASONIC SENSITIVITY (GAIN) COMPENSATION. (See figure 58.1-17.) Sensitivity differences between the standard and part can be compensated for as follows:

a. Select a representative area in the standard and part which are approximately equal in thickness or length of sound travel and have similar reflective surfaces.

b. Couple the transducer to the selected area on the standard. Adjust the instrument so the initial pulse signal is displayed near the left side and the signal from the reflective surface near the right side of the CRT display. Maximize this signal by adjusting the transducer position. Adjust the instrument sensitivity (Gain) control to obtain a signal amplitude of 80 percent saturation on the CRT display. Record the sensitivity (Gain) control setting as value A.

c. Couple the transducer to the selected area on the part. Maximize the signal from the reflective surface by adjusting the transducer position. Adjust the instrument sensitivity (Gain) control to obtain a signal amplitude of 80 percent saturation on the CRT display. Record this sensitivity (Gain) control setting as value B.

d. Find the difference between value A and value B to obtain value C. When value A is greater than value B, value C is negative. When value A is less than value B, value C is positive.

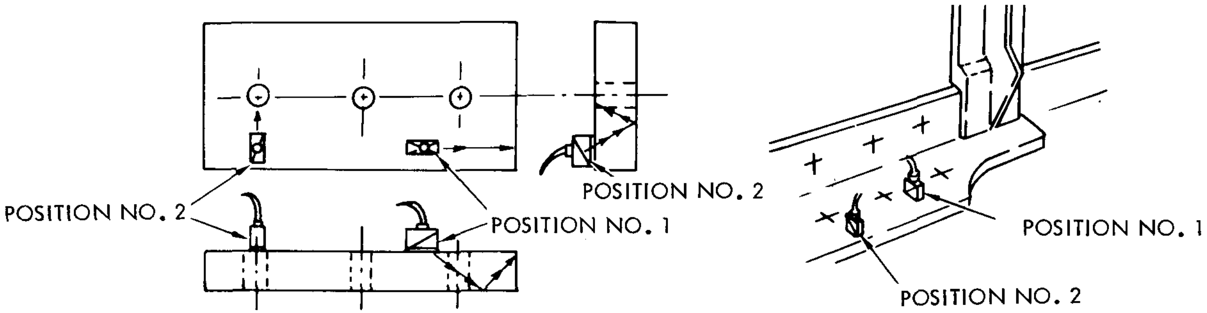
e. Calibrate the instrument as specified in the specific inspection procedure to obtain value D. Increase or decrease the value D sensitivity (Gain) control setting by value C to obtain value E.

f. Adjust instrument sensitivity (Gain) control setting for value E prior to inspecting part.

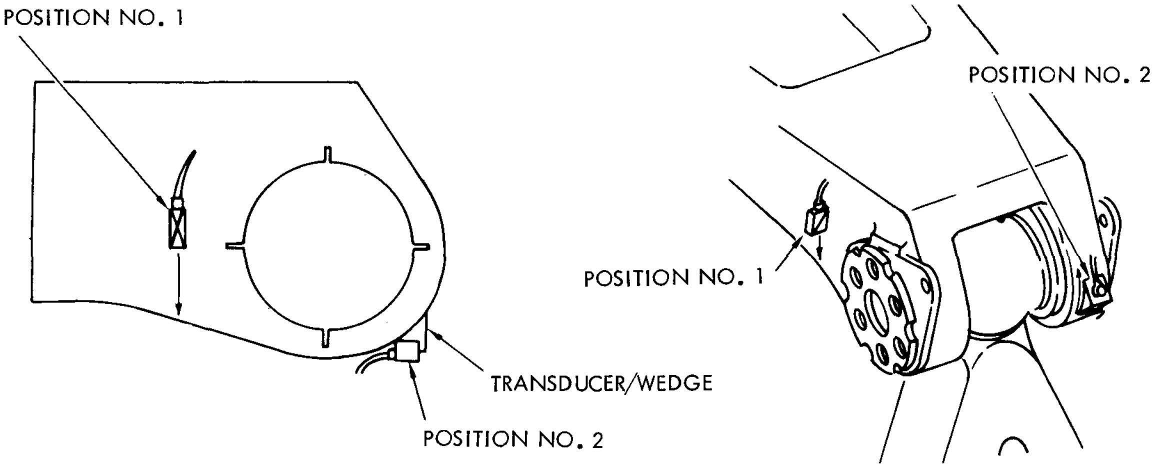
INSTRUMENT		SONIC MARK I	SONIC MARK IV	KRAUTKRAMER-BRANSON USL-32	UJ SPERRY
SWEEP LENGTH		1. MAT'L CAL dial 2. RANGE switch	1. RANGE/INCHES switch 2. MAT'L VELOCITY dial	1. SWEEP COARSE switch 2. FINE dial	SWEEP LG
SWEEP DELAY		DELAY dial	1. DELAY COARSE switch 2. DELAY FINE dial	DELAY dial	SWEEP DELAY
GAIN		1. GAIN dB COARSE switch 2. GAIN dB FINE switch 3. DAMPING DIAL	1. GAIN dB COARSE switch 2. GAIN dB FINE switch 3. GAIN TRIM dial 4. PULSE LENGTH dial (Damping)	1. GAIN COARSE switch 2. GAIN FINE switch 3. GAIN FINE uncalibrated dial 4. DAMP dial	DB GAIN
REJECT		REJECT dial	VID REJ dial	REJ dial	REJECT
FREQUENCY		FREQ. MHz switch	P/R FREQ. MHz switch	MHz switch	FREQUENCY switch
MODE	PULSE ECHO	1. NORMAL switch 2. T receptacle	1. NORMAL switch 2. T/R receptacle	1. SINGLE FIX DAMP switch 2. SINGLE DAMP switch 3. TRANSMIT receptacle	1. NORMAL 2. R/T
	THRU TRANS-MISSION	1. THRU TRANS switch 2. T&R receptacles	1. THRU TRANS switch 2. T & T/R receptacle	1. DUAL/THRU switch 2. TRANSMIT & RECEIVE receptacles	1. THRU 2. R/T
PULSE RATE		REP RATE switch on rear panel	REP RATE switch	Internal Adjustment	Back of instrument

Figure 58.1-16. Ultrasonic Equipment Comparison

SHEAR WAVE TRANSDUCER - USE A REPRESENTATIVE AREA ON STANDARD AND PART, USE THE REFLECTION FROM A KNOWN REFLECTING SURFACE WHICH IS AT APPROXIMATELY THE SAME DISTANCE SUCH AS THE TOP EDGE OF A HOLE, TOP EDGE OF THE SIDE OF THE PART OR OTHER KNOWN REFLECTING SURFACES.



LONGITUDINAL WAVE TRANSDUCER WITH WEDGE - USE A REPRESENTATIVE AREA ON STANDARD AND PART, USE A SHEAR WAVE TRANSDUCER TO OBTAIN A REFLECTION FROM A KNOWN REFLECTING SURFACE WHICH IS AT APPROXIMATELY THE SAME DISTANCE SUCH AS THE EDGE OF THE HOLE OR OTHER KNOWN REFLECTING SURFACES.



LONGITUDINAL WAVE TRANSDUCER - SELECT REPRESENTATIVE AREA ON STANDARD AND PART, USE THE REFLECTION FROM A KNOWN REFLECTING SURFACE WHICH IS AT APPROXIMATELY THE SAME DISTANCE SUCH AS THE SIDE OF A SIMILAR HOLE OR OTHER SIMILAR REFLECTING SURFACES.

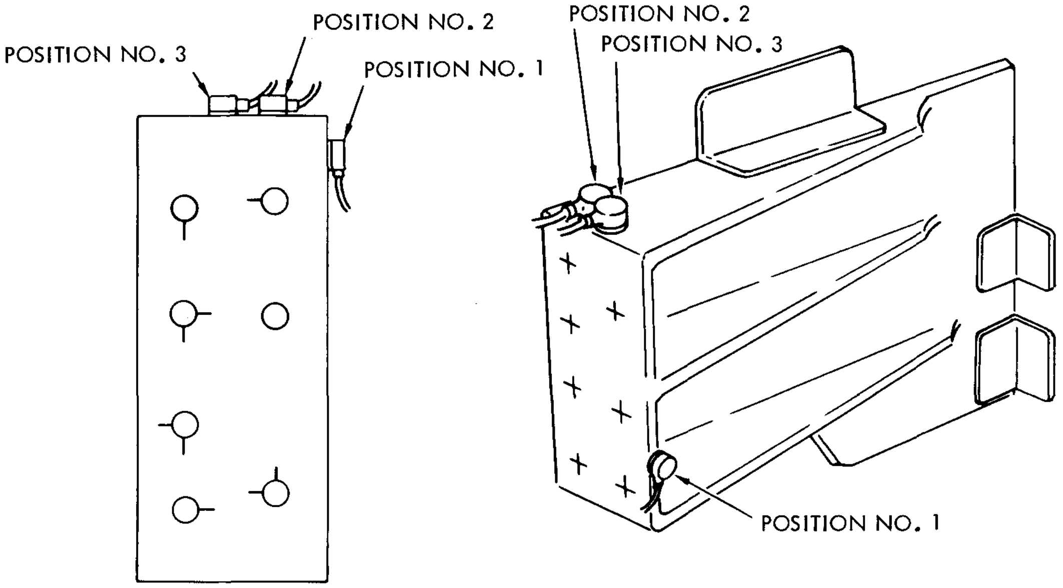


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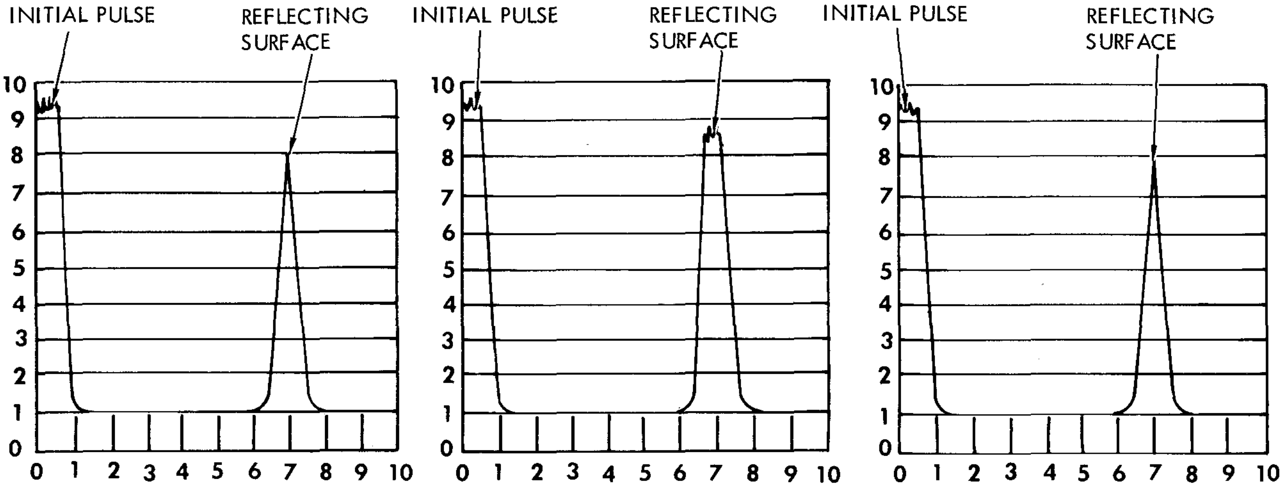
Figure 58.1-17. Computation for Ultrasonic Sensitivity (Gain) Compensation (Sheet 1 of 2)



TRANSDUCER ON STANDARD
REPRESENTATIVE AREA,
SIGNAL ADJUSTED TO 80 PERCENT
VALUE A = 12 db

TRANSDUCER ON PART
REPRESENTATIVE AREA,
SIGNAL UNADJUSTED

TRANSDUCER ON PART
REPRESENTATIVE AREA,
SIGNAL ADJUSTED TO 80 PERCENT
VALUE B = 10 db

COMPUTATION:

1. THE DIFFERENCE BETWEEN VALUE A AND VALUE B = VALUE C. VALUE C = 10 - 12 = - 2 db.
2. CALIBRATE INSTRUMENT FOR SPECIFIC INSPECTION PROCEDURE TO OBTAIN VALUE D.
3. COMBINE VALUE D AND VALUE C TO OBTAIN VALUE E.
4. IF VALUE D IS 26 db, THEN 26 db - 2 db = 24 db = VALUE E.
5. ADJUST INSTRUMENT SENSITIVITY (GAIN) CONTROL SETTING TO VALUE E AND INSPECT PART.

SMR151-58.1-0-015-2

Figure 58.1-17. Computation for Ultrasonic Sensitivity (Gain) Compensation (Sheet 2 of 2)

58.1.6.4 ULTRASONIC INSTRUMENT CALIBRATION. (See figure 58.1-18.) For ultrasonic inspection procedures using the Sonic Model FTS Mark I portable ultrasonic flaw detecter, use the following procedure for calibrating the instrument. The calibration may be performed using the longitudinal or shear wave transducer and calibration standard listed for this procedure or by substituting the transducer and calibration standard listed for the specific inspection procedure.

a. Equipment:

(1) Portable Ultrasonic Flaw Detector, Sonic Model FTS Mark I.

(2) Transducer, 5.0 MHz, 0.250-inch diameter, Longitudinal Wave, Automation Industries P/N 57A2214 with connecting cable or as specified in the inspection procedure.

(3) Transducer, 5.0 MHz, 0.250 × 0.250, 45-degree, Shear Wave, Mastercraft Enterprises Inc. P/N HMT 7-221 with connecting cable or as specified in the inspection procedure.

(4) Couplant, light oil.

(5) Calibration standard, as illustrated or as specified in the inspection procedure.

b. Preliminary settings:

(1) Set POWER/OFF switch to POWER and observe low battery indicator light is out.

(2) Connect coaxial cable and transducer to "R" receptable on the instrument front panel.

(3) Set controls as follows:

REP RATE (Rear Panel)	3K
FILTER (Rear Panel)	MED
MAT'L CAL	MIN (Fully Counterclockwise)
GAIN dB FINE	Mid-Range (8)
GAIN dB COARSE	20 dB
REJECT	4
FREQ-MHz	5
DAMPING	Mid-Range (5) or Medium
NORMAL/THRU TRANS	NORMAL
DELAY	Adjust control to position the leading edge of the initial pulse at the extreme left side of the CRT sweep baseline.

c. Detailed settings:

(1) Apply a thin film of couplant to the surface of the calibration standard and couple the tranducer so that it is aimed directly at the hole. Adjust the RANGE (Coarse) and DELAY controls to display the signal from the hole near the center of the CRT (Cathode Ray Tube) display.

(2) Slide the transducer so that it is aimed directly at the sawcut and note that the sawcut signal position is immediately to the right of the hole signal. Adjust the transducer position to maximize the sawcut signal amplitude and adjust the GAIN dB COARSE and FINE controls to obtain 80 percent saturation on the CRT display. A slight adjustment of the following controls may be required to achieve the desired CRT display:

RANGE (Coarse), DELAY, and MAT'L CAL	Adjust to display initial pulse and sawcut signal as illustrated.
GAIN dB COARSE and FINE	Adjust sawcut signal to achieve 80 percent saturation on the CRT display.
REJECT	Adjust to decrease any excessive noise on the sweep baseline.
DAMPING	Adjust to reduce pulse flare (ringing) of the initial pulse while maintaining the largest possible signal amplitude from the sawcut.

(3) Slide the transducer to the opposite side of the hole from the sawcut and observe the loss of signal.

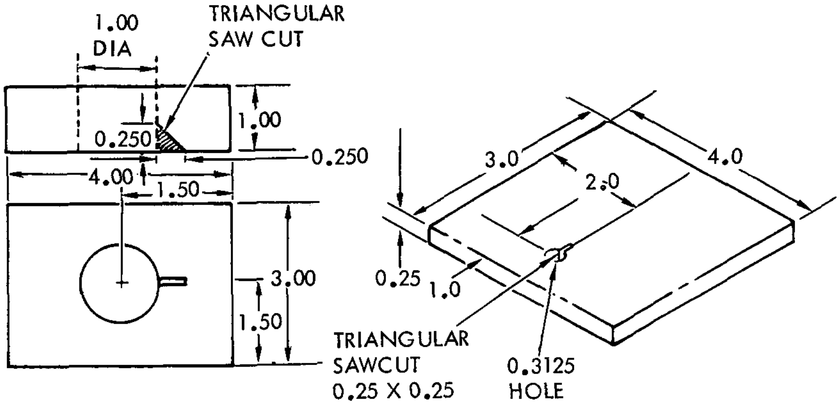
(4) Scan across the hole, sawcut, and "no sawcut" area and observe the CRT presentation as the signal shifts.

WARNING

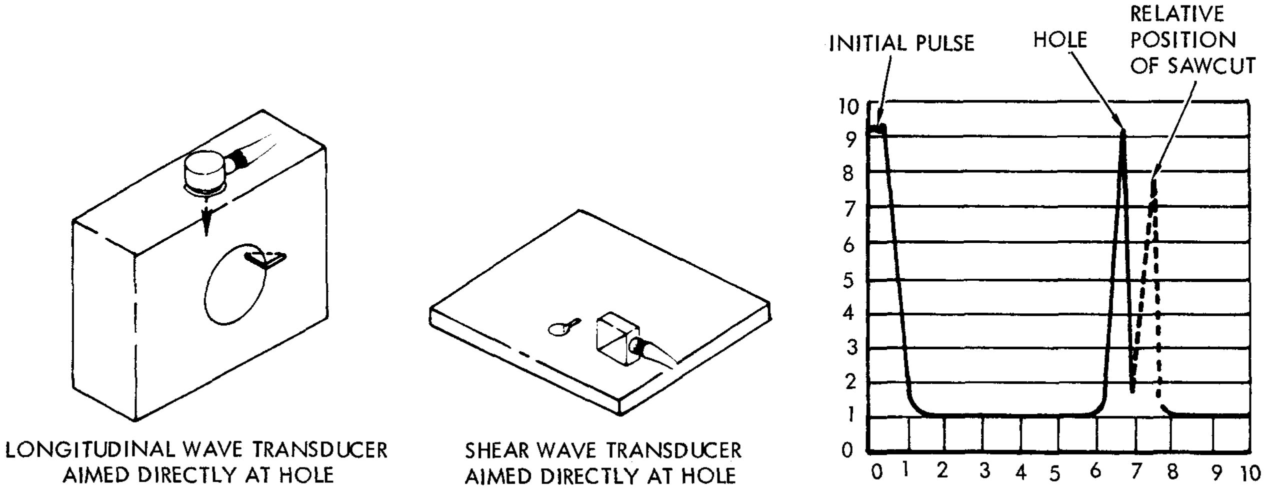
The use of radiation in nondestructive testing presents potential hazards to operating and adjacent personnel unless all safety precautions are observed.

58.1.7 RADIOGRAPHIC (X-RAY) METHOD.

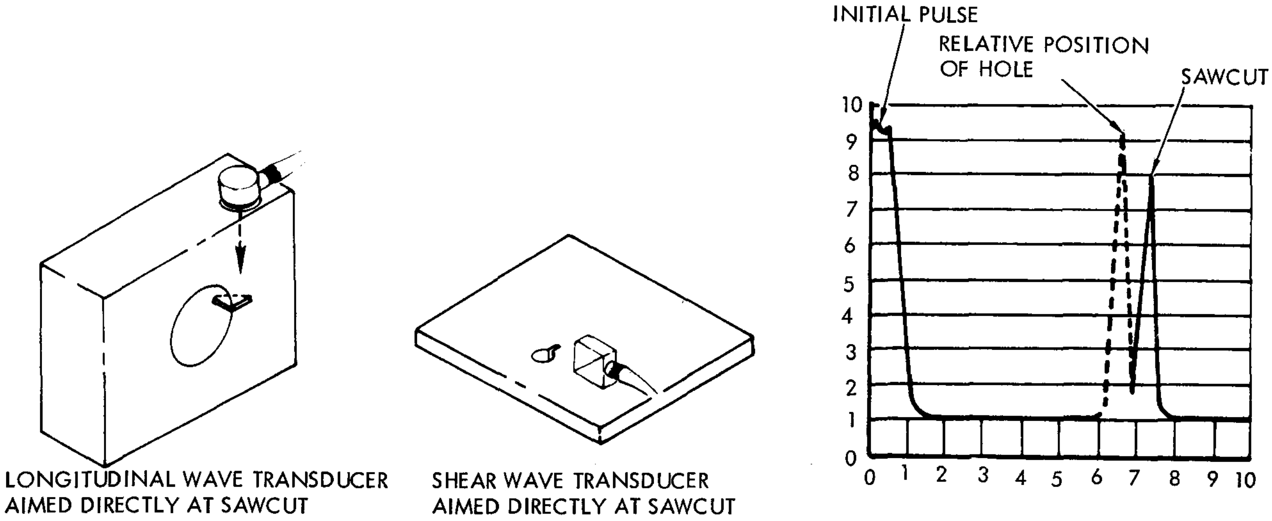
58.1.7.1 PURPOSE. X-ray inspection is used to show internal and external structural details of all types of parts and material. This method is used for the inspection of airframe structure for defects otherwise inaccessible for other methods of nondestructive inspection, or to verify conditions indicated by another method. Inspection is accomplished by passing the X-ray beam through the part or assembly to expose a radiographic film. The processed film shows the structural details of the part by variations in film density. The radiograph is interpreted for indication of defects.



TYPICAL CALIBRATION STANDARDS
(7075 ALUMINUM)



TYPICAL TRANSDUCER POSITION AND CRT PRESENTATION - HOLE



TYPICAL TRANSDUCER POSITION AND CRT PRESENTATION - SAWCUT

SMR151-58.1-0-016

Figure 58.1-18. Ultrasonic Instrument Calibration

58.1.7.2 LIMITATIONS. The two primary limitations of X-ray inspection are: the defect must be large enough to produce a visible image on the X-ray film and the plane of the X-ray beam must be oriented to be in line with the plane of a possible defect.

58.1.7.3 RADIOGRAPHIC TECHNIQUES. The problem of radiographing a section having several thicknesses, such as an air frame, resolves itself into one of using the available film-required density and contrast. In other words, the kilovoltage and exposure should be so adjusted that the image of the thinnest part has the acceptable density on one film, and the thickest has the acceptable density on the other film.

58.1.7.4 SET UP. Position the X-ray tube, film, and film identification as specified in each individual procedure.

a. Locally fabricated templates of cloth or lightweight metal material are recommended for use in locating primary beam alignment spots. The alignment spot is the point on the structure at which the X-ray tube target is aimed before exposure of the film. Measurement on successive airplanes to determine the alignment spot for a particular exposure is eliminated by use of the template. One edge or side of the template is placed on a point of known location; the opposite point or corner of the template is the location of the alignment spot. Layout and fabrication of the template are possible after the location of the alignment spot is determined by measurement.

b. Support of the X-ray tube is not shown in the details of the radiographic technique. The X-ray tube may be supported by a maintenance platform outside the airplane. The tube can be supported by an adjustable stand or table inside the airplane. Always secure the X-ray tube to some fixed object to prevent damage by an accidental fall.

58.1.7.5 ADJUSTMENTS. Settings specified in individual radiographic procedures in this chapter were established to provide quality radiographs and the densities specified in the procedure. It may be necessary to vary the MA, Time, and KV settings due to differences in equipment, film, and method of processing in order to achieve the density specified and quality necessary. Therefore, the MA and Time settings should be construed as guidelines and KV may be lowered if necessary (not raised); however, the density specified must be maintained.

58.1.7.6 FILM SUBSTITUTION. When films specified in individual procedures are not available, other vendor films may be substituted as shown in figure 58.1-19. When a procedure specifies particular films, finer grain films may be substituted for the specified films, or used with the specified films in multiple loading. It may be necessary to make some adjustments in exposure data when substituting films.

58.1.7.7 FILM DENSITY. Median film density is a density value or range of film density values which are considered essential for accurate film interpretation. The median film density values given in individual procedures in this chapter have a tolerance of -0.3 to +1.5 for solid metal structure. The median film density is required in all critical areas specified in the defects paragraph and procedure figure. Where multiple film loading is used, the median film density is required in each critical area on at least one of the films.

58.1.7.8 FILM LOADING. Unless specified in the individual procedure, film location in the film pack is optional. When not specified in the individual procedure, lead screens are not required.

58.1.7.9 FILM IDENTIFICATION. Each piece of X-ray film will be identified (via lead letters or lead tape during exposure) with the following information: procedure number (or work specification item number), upper, lower, left, right, forward, or aft as applicable, exposure number and film number of each exposure (example: E20-44. This would be for exposure 20 and film number 44); serial number (tail) or aircraft; date of film exposure.

58.1.7.10 RADIOGRAPHIC INSPECTION DATA FORM. A sample radiographic inspection form is shown in figure 58.1-20. A form similar to this should be used in reporting radiographic inspections of critical areas on the airframe. Suitable entry in the airplane maintenance log should be made.

58.1.7.11 AIRPLANE RECORDS. X-ray film will be filed by airplane serial number and date of inspection (i.e., special inspection, etc.). Film will be retained on file for one year after the next inspection.

58.1.7.12 POLAROID FILM. Polaroid film, Type 52, may be used in an X-ray procedure to confirm alignment. Polaroid film must not be substituted for film identified in Radiographic Inspection Data blocks for specific procedures.

CAUTION

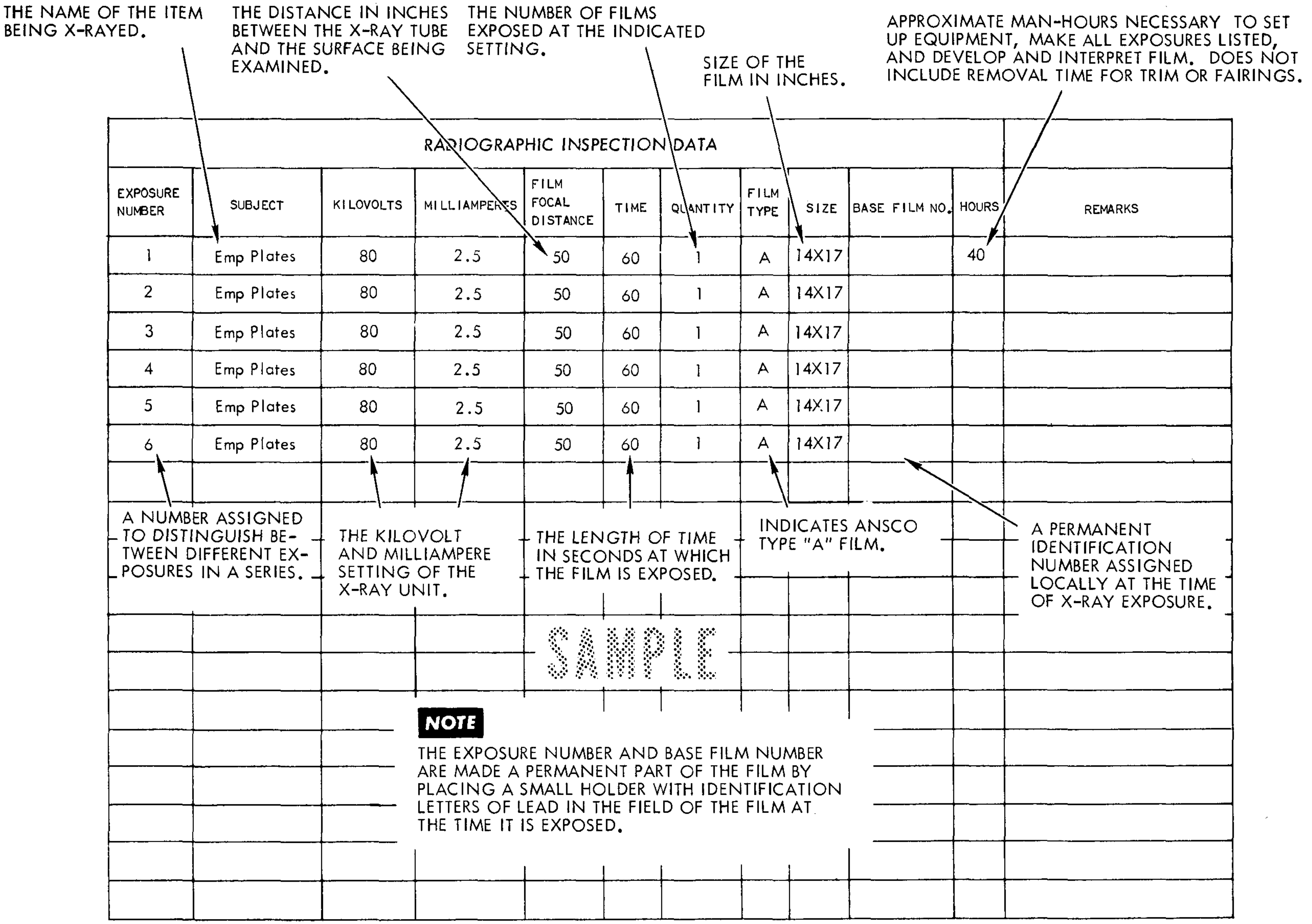
Inexperienced personnel should not attempt to interpret the results of the nondestructive evaluations. Misinterpretation could result in serviceable parts being rejected and defective parts being accepted and possibly resulting in an unsafe condition.

58.1.8 ACCEPTANCE/REJECTION CRITERIA.

It is essential that inspection personnel be skilled and well trained in the field of nondestructive evaluation to enable them to adequately report the results. This chapter of the manual provides a guide for performing nondestructive evaluation. Both primary and back-up NDE procedures are recommended where applicable. For example, the indications revealed by the primary procedure can be confirmed by performing the back-up procedure. The results of both procedures should be documented for acceptance/rejection review. Repair or replacement criteria of parts determined to be damaged will be in accordance with the applicable repair section of this manual.

FILM		SOURCE IDENTIFICATION		
CLASS	DESCRIPTION	KODAK	DUPONT	GAVAERT
I	Intermediate speed Ultra fine grain Highest contrast	R	45	D2
II	Slow speed Very fine grain High contrast	M	55	D4
—	Intermediate speed Fine grain Medium contrast	T	65	D5
III	Moderate speed Fine grain Medium contrast	AA	70	D7
For the purpose of this chapter, the film listed for each class are equivalent. Results to be expected when using film from the various sources may vary slightly.				

Figure 58.1-19. Equivalent Film Class and Source.



THE NAME OF THE ITEM BEING X-RAYED.

THE DISTANCE IN INCHES BETWEEN THE X-RAY TUBE AND THE SURFACE BEING EXAMINED.

THE NUMBER OF FILMS EXPOSED AT THE INDICATED SETTING.

SIZE OF THE FILM IN INCHES.

APPROXIMATE MAN-HOURS NECESSARY TO SET UP EQUIPMENT, MAKE ALL EXPOSURES LISTED, AND DEVELOP AND INTERPRET FILM. DOES NOT INCLUDE REMOVAL TIME FOR TRIM OR FAIRINGS.

RADIOGRAPHIC INSPECTION DATA											
EXPOSURE NUMBER	SUBJECT	KILOVOLTS	MILLIAMPERES	FILM FOCAL DISTANCE	TIME	QUANTITY	FILM TYPE	SIZE	BASE FILM NO.	HOURS	REMARKS
1	Emp Plates	80	2.5	50	60	1	A	14X17		40	
2	Emp Plates	80	2.5	50	60	1	A	14X17			
3	Emp Plates	80	2.5	50	60	1	A	14X17			
4	Emp Plates	80	2.5	50	60	1	A	14X17			
5	Emp Plates	80	2.5	50	60	1	A	14X17			
6	Emp Plates	80	2.5	50	60	1	A	14X17			

A NUMBER ASSIGNED TO DISTINGUISH BETWEEN DIFFERENT EXPOSURES IN A SERIES.

THE KILOVOLT AND MILLIAMPERE SETTING OF THE X-RAY UNIT.

THE LENGTH OF TIME IN SECONDS AT WHICH THE FILM IS EXPOSED.

INDICATES ANSCO TYPE "A" FILM.

A PERMANENT IDENTIFICATION NUMBER ASSIGNED LOCALLY AT THE TIME OF X-RAY EXPOSURE.

SAMPLE

NOTE

THE EXPOSURE NUMBER AND BASE FILM NUMBER ARE MADE A PERMANENT PART OF THE FILM BY PLACING A SMALL HOLDER WITH IDENTIFICATION LETTERS OF LEAD IN THE FIELD OF THE FILM AT THE TIME IT IS EXPOSED.

SMR151-58.1-0-1

Figure 58.1-20. Sample Radiographic Inspection Data Form

SECTION 58.2

WING GROUP

Contents Page

GENERAL DESCRIPTION 58-53

Inspection Procedures 58-53

58.2.1 GENERAL DESCRIPTION.

The wing group consists of the box structure, fixed leading edge, movable leading edge (leading edge flaps), trailing edge, trailing edge flap, aileron, tip, fairings, and an external fuel tank. See Chapter 57 for complete description of the wing structure and detail components.

58.2.1.1 INSPECTION PROCEDURES. Wing Group areas and components to be inspected are tabulated in the Wing Group Inspection Index (See Table 58.2-1). Each inspection procedure has been assigned an inspection index number which is cross-referenced to the relevant paragraph and page number.

Table 58.2-1. Wing Group Inspection Index

INDEX NO.	INSPECTION TITLE	PARAGRAPH NO.	PAGE NO.
W-1	Wing-to-Fuselage Attachment Bolts, Fittings, and Mainframes.	58.2.2	58-54
W-2	Wing Lower Root Rib Shear Fitting.	58.2.3	58-58
W-3	MLG Aft Support Spar-to-Fuselage Attachment Connecting Linkage and Attachment Bolts.	58.2.4	58-60
W-4	MLG Aft Support Spar and Bearing Support Cap.	58.2.5	58-72
W-5	MLG Actuator and Auxiliary Spar Outboard Support Fitting.	58.2.6	58-74
W-6	Trailing Edge Flap Actuator Support, Flap Sta. 19 and 58.	58.2.7	58-76
W-7	External Fuel Tank to Wing Attachment Bolts and Fittings.	58.2.8	58-78
W-8	Wing Panel Risers.	58.2.9	58-83
W-9	Aileron Counterweight Region.	58.2.10	58-92
W-10	Wing Lower Surface Spanwise Splices.	58.2.11	58-94A
W-11	Wing Panel to Rear Beam Lower Cap Spanwise Splice, BBS75 to BBS148.	58.2.12	58-94C
W-12	Wing Panel to Mid Beam Lower Cap Spanwise Splice, BBS55 to BBS187.	58.2.13	58-94J
W-13	Wing Lower Panel Drain Holes, Left and Right.	58.2.14	58-94P
W-14	Wing Upper and Lower Panel to Root Rib Attachment.	58.2.15	58-94U
W-15	Wing Upper and Lower Root Rib Fitting Attachment.	58.2.16	58-94AB

58.2.2 WING TO FUSELAGE ATTACHMENT BOLTS, FITTINGS, AND MAINFRAMES.

58.2.2.1 DESCRIPTION. (See figure 58.2-1-4/0.) The wing is attached to the fuselage by wing joint fittings secured to the box structure at the wing root rib. These fittings are secured by bolts to five fuselage frames. The wing joint fittings are fabricated from 7075-T6 aluminum alloy extrusions and the fuselage mainframes are fabricated from 7075-T6 aluminum forgings. The fittings and frames are finished with zinc chromate primer. The attachment bolts are fabricated from high strength alloy steel and are cadmium plated.

58.2.2.2 DEFECTS. Cyclic wing loads transmitted from the wing joint fittings to the fuselage mainframes by way of the attachment bolts may cause cracking in the wing joint fittings, the mainframes, and the attachment bolts. Cracks in the wing joint fittings and mainframes will initiate in the fastener holes and propagate outward in random directions. Circumferential and longitudinal cracks will originate in the thread root areas, in the shank sections, and in the radii beneath the heads and/or in the heads of the attachment bolts.

Note

Both primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.2.2.3 PRIMARY NDE PROCEDURE NO. 1 — EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1/2-inch diameter, Ideal Specialty Co. P/N 6200-1/2-BH-BNC or equivalent.

(3) Probe, Bolt Hole, 5/8-inch diameter, Ideal Specialty Co. P/N 6200-5/8-BH-BNC or equivalent.

(4) Probe, Bolt Hole, 3/4-inch diameter, Ideal Specialty Co. P/N 6200-3/4-BH-BNC or equivalent.

(5) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

WARNING

The wing must be supported in a neutral stress position prior to removal of wing attachment bolts in order to maintain the integrity of the wing and to preclude movement between mating surfaces when the bolts to be inspected are removed.

(2) Jack and shore the airplane to provide support of the wing in a neutral position per instructions in Sections 7.0 and 7.1 of this manual.

(3) Remove belly fairings as required to provide access to the wing joint fitting-to-fuselage frame attachments at fuselage stations 410, 430, 450, 470, and 490. Refer to Section 52.3 of this manual.

c. Access: Access to the wing-to-fuselage attachment bolts for removal and to the fastener hole areas to be inspected is from beneath the airplane through the access openings.

d. Preparation of part:

CAUTION

Do not remove more than one upper and one lower wing-to-fuselage attachment bolt at each fuselage station at any time. One upper and one lower attachment bolt at each station must be fully installed at all times. Failure to comply could result in difficulty in reinstalling bolts and/or damage to structure.

(1) Remove one upper and one lower wing-to-fuselage attachment bolt at each fuselage station. This operation may be performed on both the left and right sides simultaneously. A total of twenty bolts, ten left side and ten right side, may be removed for inspection at any one time.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface areas as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

Note

The outboard end of the fastener holes in the wing root rib fittings are curved at the barrel nut cavity. The fore and aft surfaces extend farther outboard than the upper and lower surfaces. A downward deflection of the instrument needle will occur when scanning at the upper and lower quadrants of the hole. This will not indicate a defect.

f. Inspection: Using appropriate diameter bolt hole probe, scan the entire inner surface of each fastener hole in the sequence illustrated through all layers of material. Refer to Section 58.1 of this manual for bolt hole scanning of straight wall fastener holes. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

58.2.2.3 (Cont).

g. Mark and report indicated defects.

h. If no defect is indicated, proceed with Primary NDE Procedure No. 2.

58.2.2.4 PRIMARY NDE PROCEDURE NO. 2 — MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

c. Access: Same as for Primary NDE Procedure No. 1.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field: 700 amperes.

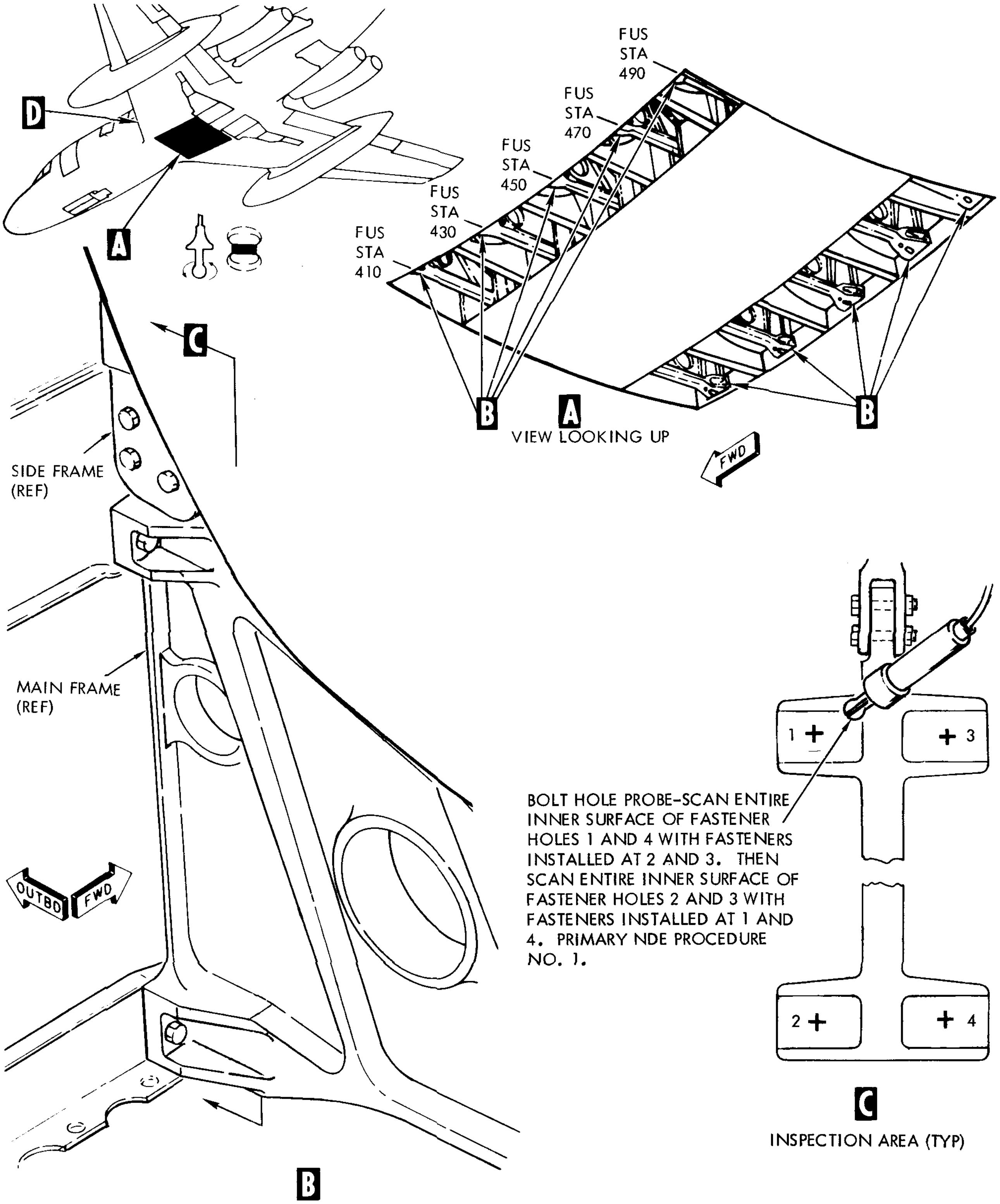
(2) Longitudinal field: 2000 amperes.

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects.

58.2.2.5 BACKUP NDE PROCEDURE. None required.

58.2.2.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual. (Refer to Section 51.0 of this manual for wing to fuselage attachment bolt torque values.)

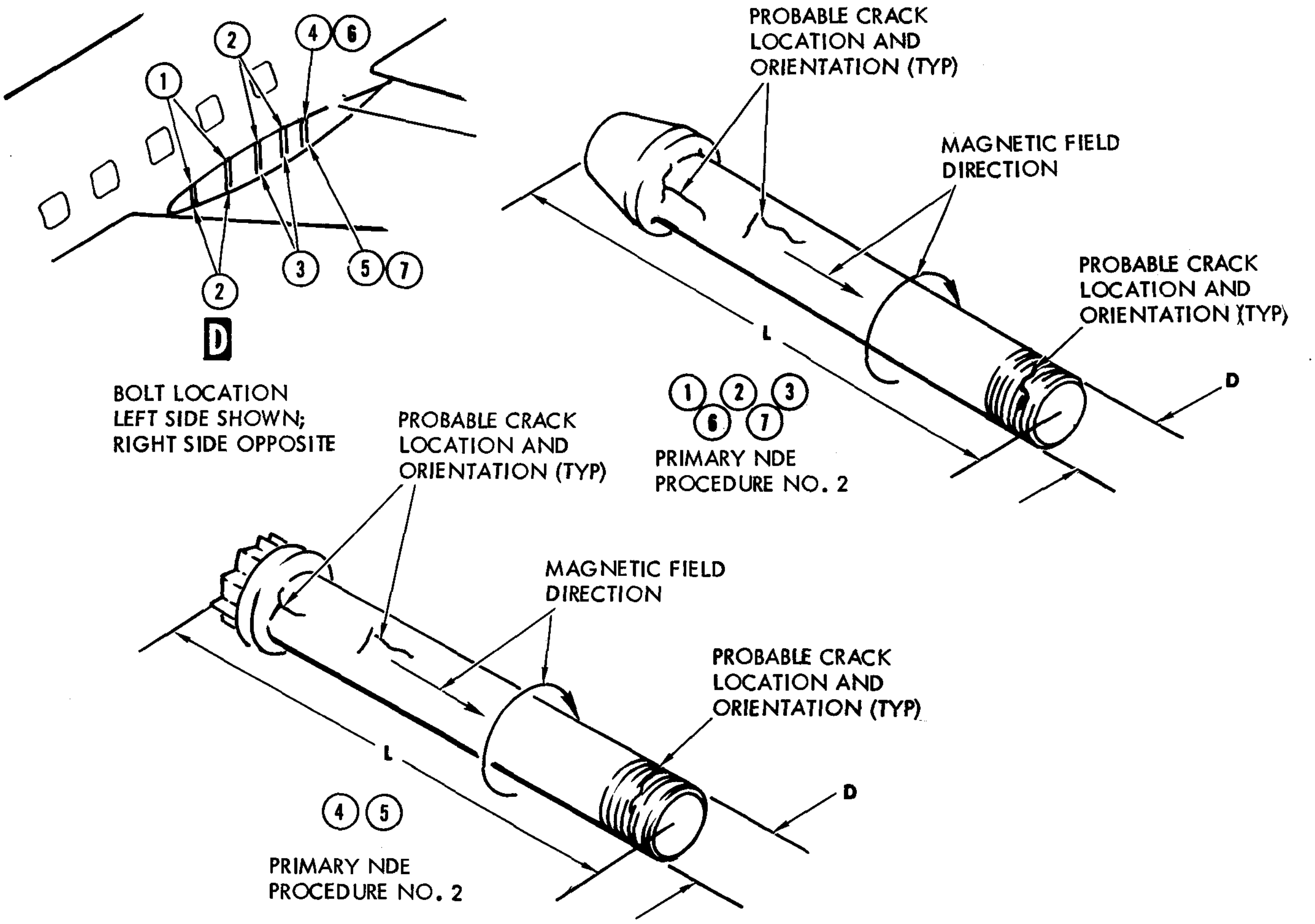


BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF FASTENER HOLES 1 AND 4 WITH FASTENERS INSTALLED AT 2 AND 3. THEN SCAN ENTIRE INNER SURFACE OF FASTENER HOLES 2 AND 3 WITH FASTENERS INSTALLED AT 1 AND 4. PRIMARY NDE PROCEDURE NO. 1.

(LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE)

SMR151-58.2-0-001-1

Figure 58.2-1-4/0. Wing to Fuselage Attachment Bolts, Fittings, and Mainframes (Sheet 1 of 2)


①
②
④⑥
②
③
⑤⑦
D
BOLT LOCATION
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
①②③
⑥⑦
PRIMARY NDE PROCEDURE NO. 2
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
④⑤
PRIMARY NDE PROCEDURE NO. 2

BOLT	D (IN)	L (IN)	MATERIAL	FINISH
① MS20008-20	0.50	2.56	△1	△2
② MS20010-22	0.63	2.94	△1	△2
③ MS20012-28	0.75	3.56	△1	△2
④ MS21250-10022 △4	0.63	2.89	△3	△2
⑤ MS21250-12028 △4	0.75	3.50	△3	△2
⑥ MS20010-22 △5	0.63	2.94	△1	△2
⑦ MS20012-28 △5	0.75	3.56	△1	△2

NOTE

△1 ALLOY STEEL, INTERNAL WRENCHING

△2 CADMIUM PLATED

△3 ALLOY STEEL, 12-POINT EXTERNAL WRENCHING

△4 5056 AND UP AND AIRPLANES MODIFIED BY SB329-157.

△5 5001 THROUGH 5055 NOT MODIFIED BY SB329-157.

SMR151-58.2-0-001-2

Figure 58.2-1-4/0. Wing to Fuselage Attachment Bolts, Fittings, and Mainframes (Sheet 2 of 2)

58.2.3 WING LOWER ROOT RIB SHEAR FITTING.

58.2.3.1 DESCRIPTION (See figure 58.2-2-4/2) The lower root rib shear fittings serve to connect the left and right wing main ribs to the lower attach fittings at Wing Station 44.5. These fittings are located at five main wing to fuselage mate points. The fittings are machined from 7075-T6 aluminum forgings and finished with zinc chromate primer. Fittings installed by Service Bulletin 329-190B may be 7075-T6 aluminum forging or 7075-T73511 aluminum bar.

58.2.3.2 DEFECTS. Stress corrosion cracks may occur at the end of the machined slots in the barrel end of the fitting. For fittings installed by Service Bulletin 329-190B, stress corrosion cracks may occur in the base of the fitting through the vertical attachment hole.

58.2.3.3 PRIMARY NDE PROCEDURE — EDDY CURRENT.

a. NDI equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM 100SS or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM 102AS or equivalent.

(4) Probe, Surface, General Purpose, Flexible, 8 inches long, VM Products P/N VM 100P-8 or equivalent.

(5) Calibration Standard, aluminum (See figure 58.1-9.)

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane:

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove belly fairings to provide access to the lower wing to fuselage attach area at fuselage station 410, 430, 450, 470 and 490. Refer to Section 52.3 of this manual.

c. Access: The inspection area is accessible from beneath the airplane through the access opening. For fittings installed by Service Bulletin 329-190B, it may be desirable to temporarily remove the vertical fastener.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the surface area of the fitting as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

Note

Existing cracks may have been previously stop-drilled. All areas must be inspected, even when previously stop drilled. Proceed to Backup NDE Procedure for inspection of previously stop drilled area.

f. Inspection:

(1) Scan around the top surface of the barrel and up the fitting on both sides of the slot. For fittings installed by Service Bulletin 329-190B, a scan of the base of the fitting around the vertical fastener hole is sufficient.

(2) Scan the inside area of the barrel as far down as access permits in a circumferential direction. For fittings installed by Service Bulletin 329-190B, a scan of the base of the fitting around the vertical fastener hole is sufficient.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.2.3.4 BACKUP NDE PROCEDURE — FLUORESCENT PENETRANT.

a. NDI equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

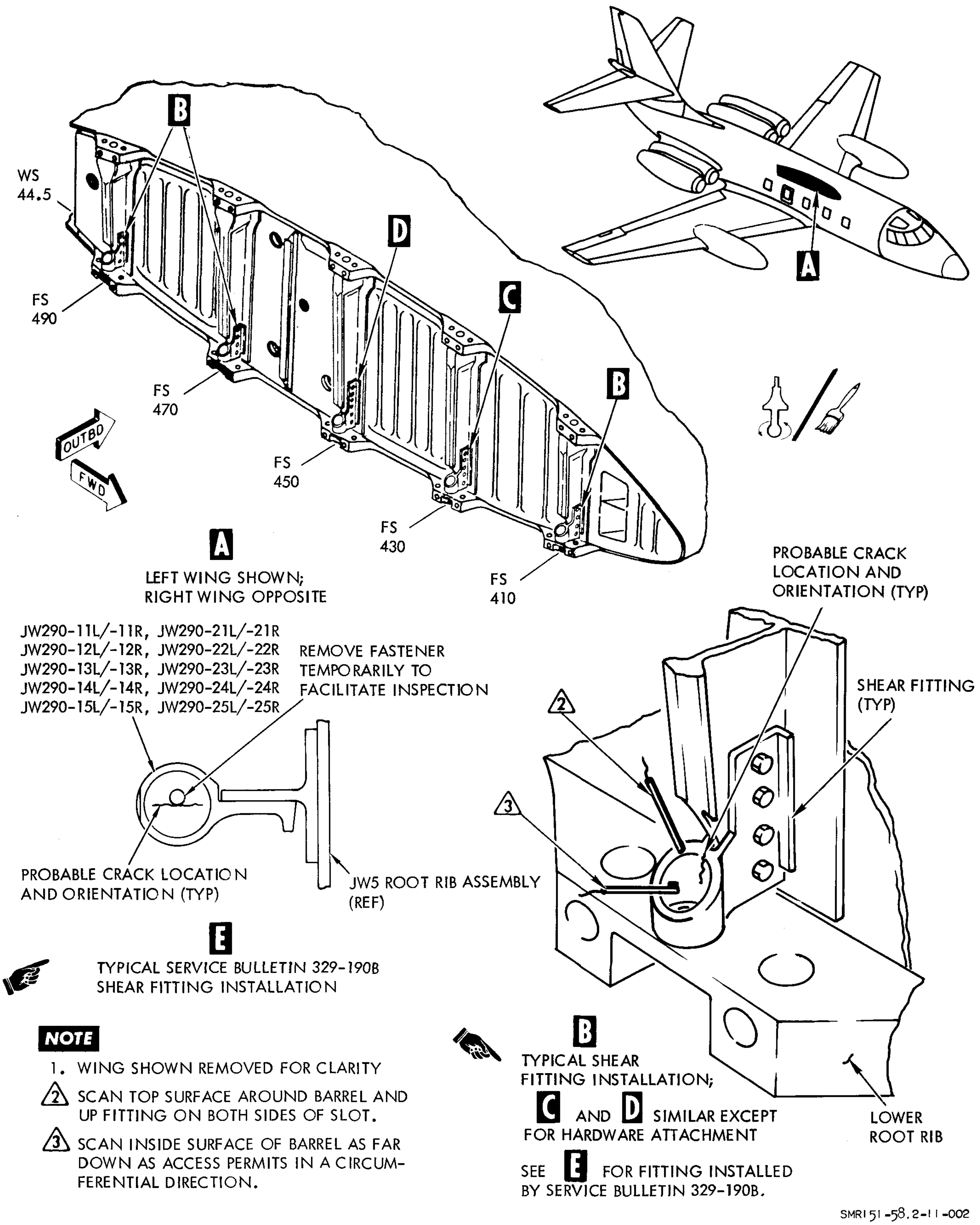
d. Preparation of part: Refer to Section 58.0 of this manual for cleaning instructions. Strip finish and clean local suspected defective area.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. Refer to Section 59.2 of this manual for defect evaluation/repair requirements.

58.2.3.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


B
WS 44.5
D
FS 490
C
FS 470
B
OUTBD
FWD
FS 450
FS 430
A
FS 410
A
LEFT WING SHOWN; RIGHT WING OPPOSITE
JW290-11L/-11R, JW290-21L/-21R
JW290-12L/-12R, JW290-22L/-22R
JW290-13L/-13R, JW290-23L/-23R
JW290-14L/-14R, JW290-24L/-24R
JW290-15L/-15R, JW290-25L/-25R
REMOVE FASTENER TEMPORARILY TO FACILITATE INSPECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
JW5 ROOT RIB ASSEMBLY (REF)
E
TYPICAL SERVICE BULLETIN 329-190B SHEAR FITTING INSTALLATION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
SHEAR FITTING (TYP)
2
3
B
TYPICAL SHEAR FITTING INSTALLATION; C AND D SIMILAR EXCEPT FOR HARDWARE ATTACHMENT
SEE E FOR FITTING INSTALLED BY SERVICE BULLETIN 329-190B.
LOWER ROOT RIB
SMR151-58.2-11-002

NOTE

1. WING SHOWN REMOVED FOR CLARITY
2. SCAN TOP SURFACE AROUND BARREL AND UP FITTING ON BOTH SIDES OF SLOT.
3. SCAN INSIDE SURFACE OF BARREL AS FAR DOWN AS ACCESS PERMITS IN A CIRCUMFERENTIAL DIRECTION.

Figure 58.2-2-4/2. Wing Lower Root Rib Shear Fitting

58.2.4 MLG AFT SUPPORT SPAR-TO-FUSELAGE ATTACHMENT CONNECTING LINKAGE AND ATTACHMENT BOLTS.

58.2.4.1 DESCRIPTION. (See figure 58.2-3-6/4.) The MLG (main landing gear) aft support spar is attached to the fuselage bulkhead frame at fuselage station 526 by linkage consisting of the auxiliary beam attach fitting, the fuselage frame attach fitting, and the auxiliary beam attachment tie. The auxiliary beam attach fitting is secured to the fuselage frame attach fitting, MLG aft support spar, and the auxiliary beam attachment tie by three high strength steel bolts. The MLG aft support spar is fabricated from a 7075-T6 aluminum forging and finished with wash primer, zinc chromate primer and Corogard 14. The fuselage, frame attach fitting, auxiliary beam attach fitting, and auxiliary beam attachment tie are fabricated from 7075-T6 aluminum forgings, sulfuric acid anodized, and finished with zinc chromate primer. All members contain steel bushings in the lug and clevis ends.

58.2.4.2 DEFECTS. Cyclic loads from the MLG aft support spar transmitted to the fuselage frame through the connecting linkage may cause fatigue cracks to initiate in the lug and clevis areas of the connecting linkage and in the attachment bolts. Cracks in the connecting linkage will originate at the bushed fastener hole and radiate outward to the lug or clevis edge in a direction normal to the load paths. Circumferential or longitudinal cracks will originate in the thread root areas, in the shank sections, in the radii beneath the heads and/or in the heads of the attachment bolts.

Note

Primary NDE Procedure No. 1 is for inspecting the lower end of the auxiliary beam attach fitting and the outboard end of the auxiliary beam attachment tie.

58.2.4.3 PRIMARY NDE PROCEDURE NO. 1 — RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure complicance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Jack and shore the airplane to provide support of the wing in a neutral position per instructions in Section 7.0 and 7.1 of this manual.

(3) Remove trailing edge fairing door to provide access to the MLG aft support spar attachment linkage. Refer to Section 52.3 of this manual.

c. Access: Access to the inspection areas will be from the under side of the wing/fuselage through the main landing gear open bay areas and with the trailing edge fairing door removed.

d. Preparation of part: Remove the bolts securing the MLG aft support spar to the outboard lug and the auxiliary beam attachment tie and trailing edge fairing support fitting to the inboard lug at the lower end of the auxiliary beam attach fitting. (Bolts ② and ③, sheet 8 of the illustration.)

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Inspect bolts ② and ③ removed in step d. in accordance with Primary NDE Procedure No. 4.

h. Mark and report suspected defects.

i. To confirm suspected defects, perform the Backup NDE Procedure.

j. If no defects are suspected, reinstall or replace bolt ② prior to proceeding to Primary NDE Procedure No. 2 when required.

k. If no additional inspections are required, reinstall or replace bolts ② and ③ and proceed to System Securing.

Note

Primary NDE Procedure No. 2 is for inspecting the inboard end of the MLG aft support spar.

58.2.4.4 PRIMARY NDE PROCEDURE NO. 2 — ULTRASONIC

a. NDE equipment:

(1) Portable Ultrasonic Flaw Detector, Sonic Mark I or equivalent.

58.2.4.4 (Cont).

(2) Transducer, 5.0 MHz, 45 degree Shear Wave, Mastercraft Enterprises Inc. P/N HMT 7-221 or equivalent.

(3) Cable, 6-foot, Microdot/BNC Connectors, Mastercraft Enterprises P/N HMT 7-41-BM or equivalent.

(4) Calibration Standard, Lockheed P/N PDL 899, as illustrated.

(5) Couplant, light oil.

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

c. Access: Same as for Primary NDE Procedure No. 1.

d. Preparation of Part:

(1) If installed, remove the bolt securing the MLG aft support spar to the outboard lug at the lower end of the auxiliary beam attach fitting. (Bolt ②, sheet 8 of the illustration.)

(2) Refer to Section 58.0 for cleaning instructions. Clean the surface areas of the clevis at the inboard end of the MLG aft support spar as required to permit good contact between part and transducer.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer to side A of the standard in Position No. 1 as illustrated.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT. Adjust sweep to display signal from the fastener hole near the center of the CRT.

(3) Slide transducer to Position No. 2 and note that the signal from the nearside sawcut is near the right side of the CRT.

(4) Maximize the signal from the sawcut by adjusting the transducer position and adjust the signal amplitude to obtain 98% saturation on the CRT display.

(5) Apply couplant and couple transducer to side B of the standard in Position No. 3 as illustrated and note position of the far side saw cut signal.

(6) Apply couplant and couple transducer to side A of the standard in Position No. 4 as illustrated and note position of the fastener hole signal as obtained in Position No. 1 on side A.

(7) Rotate the transducer to Position No. 5 and note the loss of signal.

(8) The CRT signal from the sawcut simulates a signal from a crack at the fastener hole.

Note

Attenuation by sealant, paint finishes, material differences, etc. may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 of this manual.

f. Inspection:

(1) Apply couplant to part and couple transducer to inspection area as illustrated.

(2) Scan the outer surface on both sides of the MLG aft spar support clevis completely around the fastener holes in both a clockwise and counterclockwise direction for nearside and farside defects.

(3) Any signal received similar to and at approximately the same location on the CRT display as the nearside and/or farside sawcut signals observed during calibration will indicate a probable crack in the part.

g. Inspect bolt ② if removed in step d. (1) in accordance with Primary NDE Procedure No. 4.

h. Mark and report suspected defects.

i. To confirm suspected defects, perform the Backup NDE Procedure.

j. If no defects are suspected, reinstall or replace bolt ② prior to proceeding to Primary NDE Procedure No. 3 when required.

k. If no additional inspections are required, reinstall or replace bolt ② and proceed to System Securing.

Note

Primary NDE Procedure No. 3 is for inspecting the fuselage frame attach fitting and the upper end of the auxiliary beam attach fitting.

58.2.4.5 PRIMARY NDE PROCEDURE NO. 3 – EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM 100SS or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM 102AS or equivalent.

(4) Calibration Standard, aluminum (See figure 58.1-9.)

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1 and, in addition, remove the upper wing fillet fairing.

c. Access: Same as for Primary NDE Procedure No. 1 and, in addition, from the top side of the wing/fuselage with the upper wing fillet fairing removed.

58.2.4.5 (Cont).

d. Preparation of part:

(1) Remove the bolt securing the upper end of the auxiliary beam attach fitting to the fuselage frame attach fitting. (Bolt ①, sheet 8 of the illustration.)

(2) Refer to Section 58.0 for cleaning instructions. Clean accessible surface areas of the auxiliary beam attach fitting clevis and the fuselage frame attach fitting as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Inspect all exposed surfaces by scanning in a clockwise and counterclockwise direction completely around bushed fastener holes and in both directions on part edges. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Inspect bolt ① removed in step d. (1) in accordance with Primary NDE Procedure No. 4.

h. Mark and report suspected defects.

i. To confirm suspected defects, perform the Backup NDE Procedure.

j. If no defects are suspected, reinstall or replace bolt ① and proceed to System securing.

Note

Primary NDE Procedure No. 4 is for inspecting the connecting linkage attachment bolts removed during Primary NDE Procedures No. 1, No. 2, and/or No. 3.

58.2.4.6 PRIMARY NDE PROCEDURE NO. 4 — MAGNETIC PARTICLE

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as for Primary NDE Procedure No. 3.

c. Access: Same as for Primary NDE Procedure No. 3.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of Part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

NAS 1112-36D - 700 amperes

JW 289-1 - 700 amperes

MS 21250-05024 - 500 amperes

(2) Longitudinal Field:

NAS 1112-36D - 1800 amperes

JW 289-1 - 1800 amperes

MS 21250-05024 - 1000 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields as illustrated. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. If no defects are indicated, proceed to System Securing.

58.2.4.7 BACKUP NDE PROCEDURE — EDDY CURRENT.

a. NDE equipment: Same as for Primary NDE Procedure No. 3.

58.2.4.7 (Cont).

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

c. Access: Same as for Primary NDE Procedure No. 1.

d. Preparation of part:

Note

> Do not disassemble connecting linkage if suspected defect is in an exposed surface area.

(1) If suspected defect is in an unexposed surface area, disassemble the connecting linkage only as required to expose the suspected defective surface area.

(2) Refer to Section 58.0 for cleaning instructions. Clean the suspected defective area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 this manual for calibration of the crack detector.

Note

> The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Inspect suspected defective areas only. Ref to Section 58.1 of this manual for eddy current scanning tech ques. A sharp meter deflection, as noted during calibration, w indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.2.4.8 SYSTEM SECURING. Clean areas inspected, resto finishes, reinstall removed components, and perform operation checkouts as required in accordance with applicable sections this manual. Refer to Section 51.0 of this manual for the ML aft support spar to auxiliary beam attach fitting bolt ② torq value.

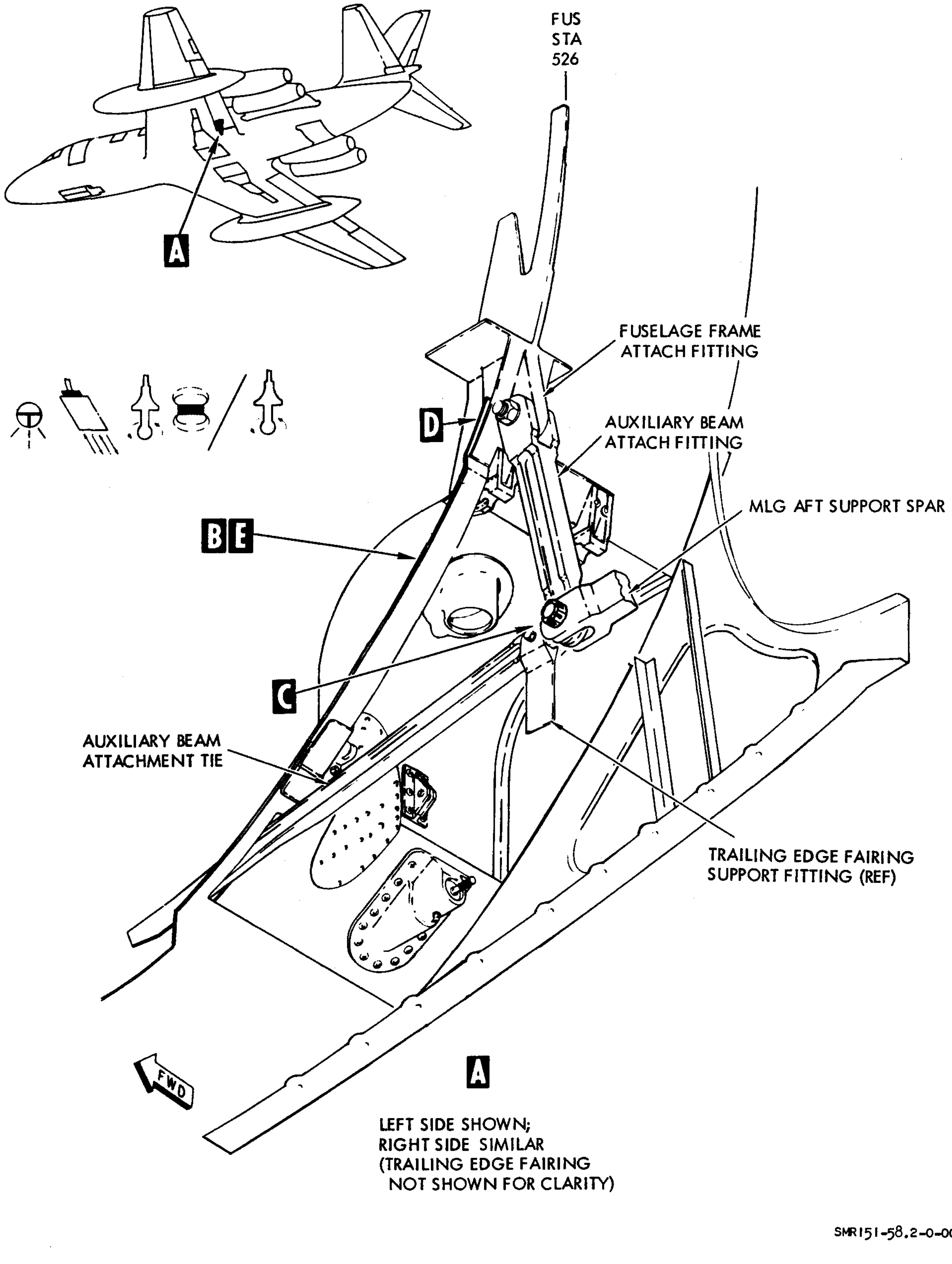
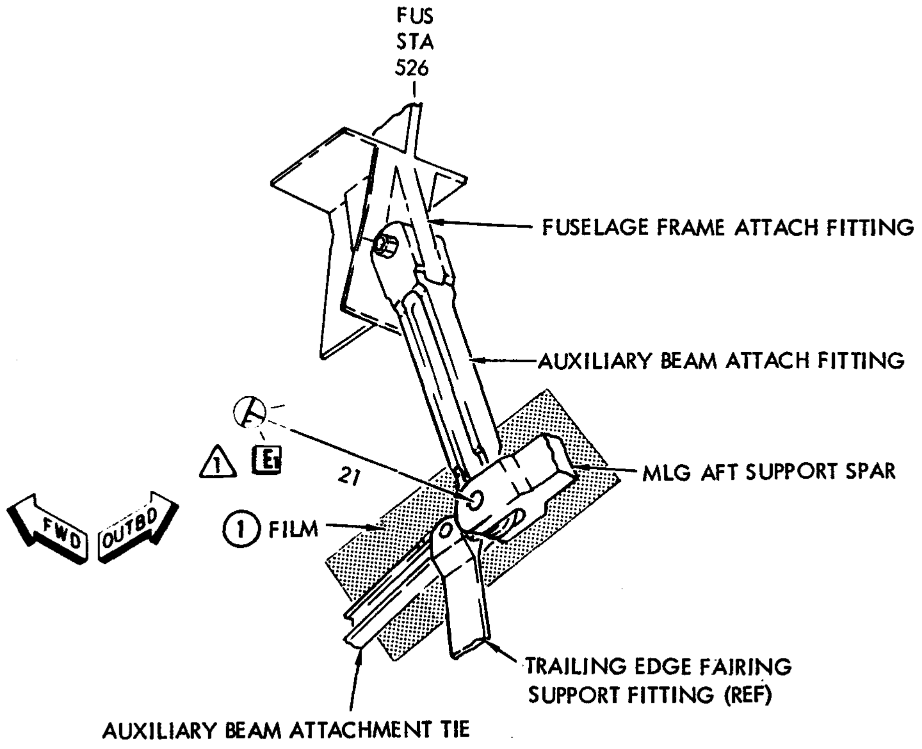


Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 1 of 8)


FUS
STA
526
FUSELAGE FRAME ATTACH FITTING
AUXILIARY BEAM ATTACH FITTING
△1
E
21
MLG AFT SUPPORT SPAR
FWD
OUTBD
① FILM
TRAILING EDGE FAIRING SUPPORT FITTING (REF)
AUXILIARY BEAM ATTACHMENT TIE

B

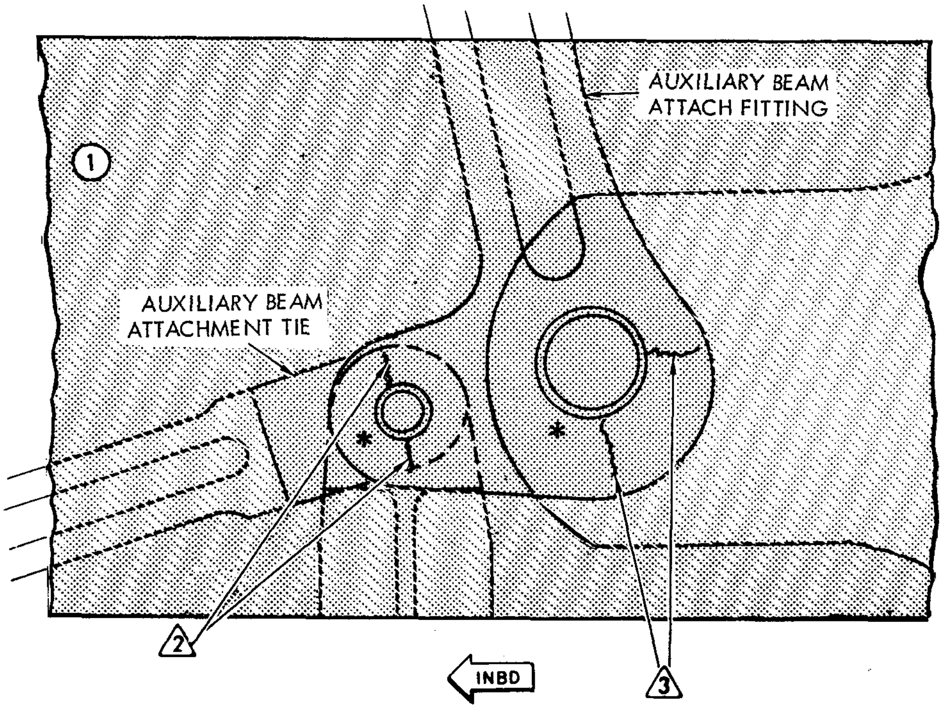
TUBE AND FILM PLACEMENT-
PRIMARY NDE PROCEDURE NO. 1

NOTE

△1 CENTER BEAM ON INBOARD END OF MLG AFT SUPPORT SPAR CLEVIS

SMR151-58.2-0-003-2

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 2 of 8)



NOTE

△2 PROBABLE CRACK LOCATION, AND ORIENTATION - AUXILIARY BEAM ATTACH FITTING INBOARD LUG AND/OR AUXILIARY BEAM ATTACHMENT TIE.

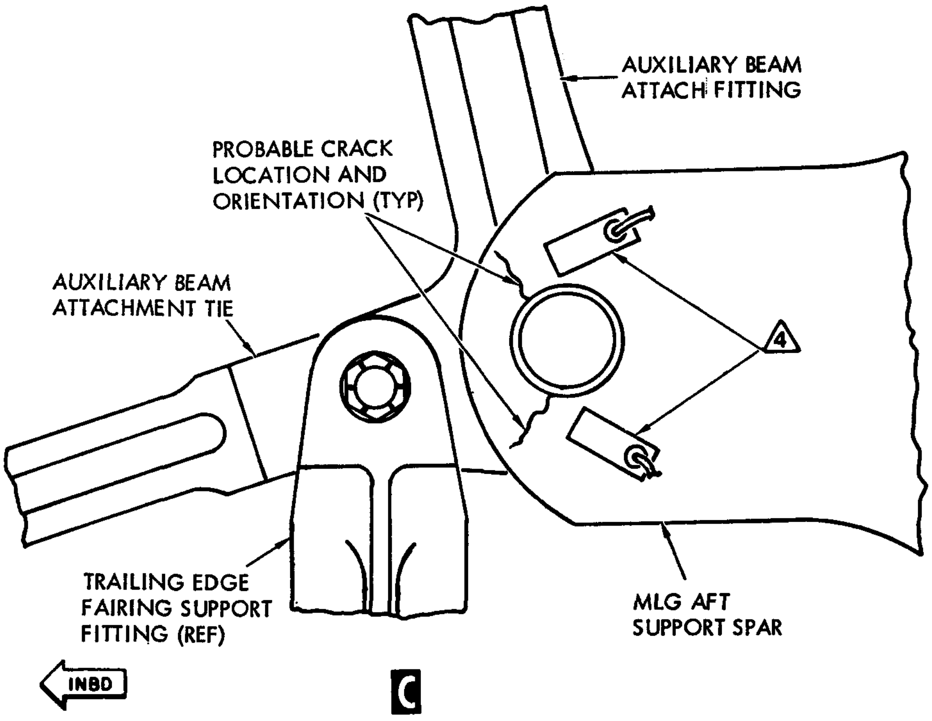
△3 PROBABLE CRACK LOCATION AND ORIENTATION - AUXILIARY BEAM ATTACH FITTING OUTBOARD LUG.

* CHECK MEDIAN FILM DENSITY AT NOTED POINTS.

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD (INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	120	5	21	75	NDT 45 NDT 55	4.5 X 17	EXPOSE ①

SMR151-58.2-0-003-3

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 3 of 8)


AUXILIARY BEAM ATTACH FITTING
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
AUXILIARY BEAM ATTACHMENT TIE
4
TRAILING EDGE FAIRING SUPPORT FITTING (REF)
MLG AFT SUPPORT SPAR
INBD

C

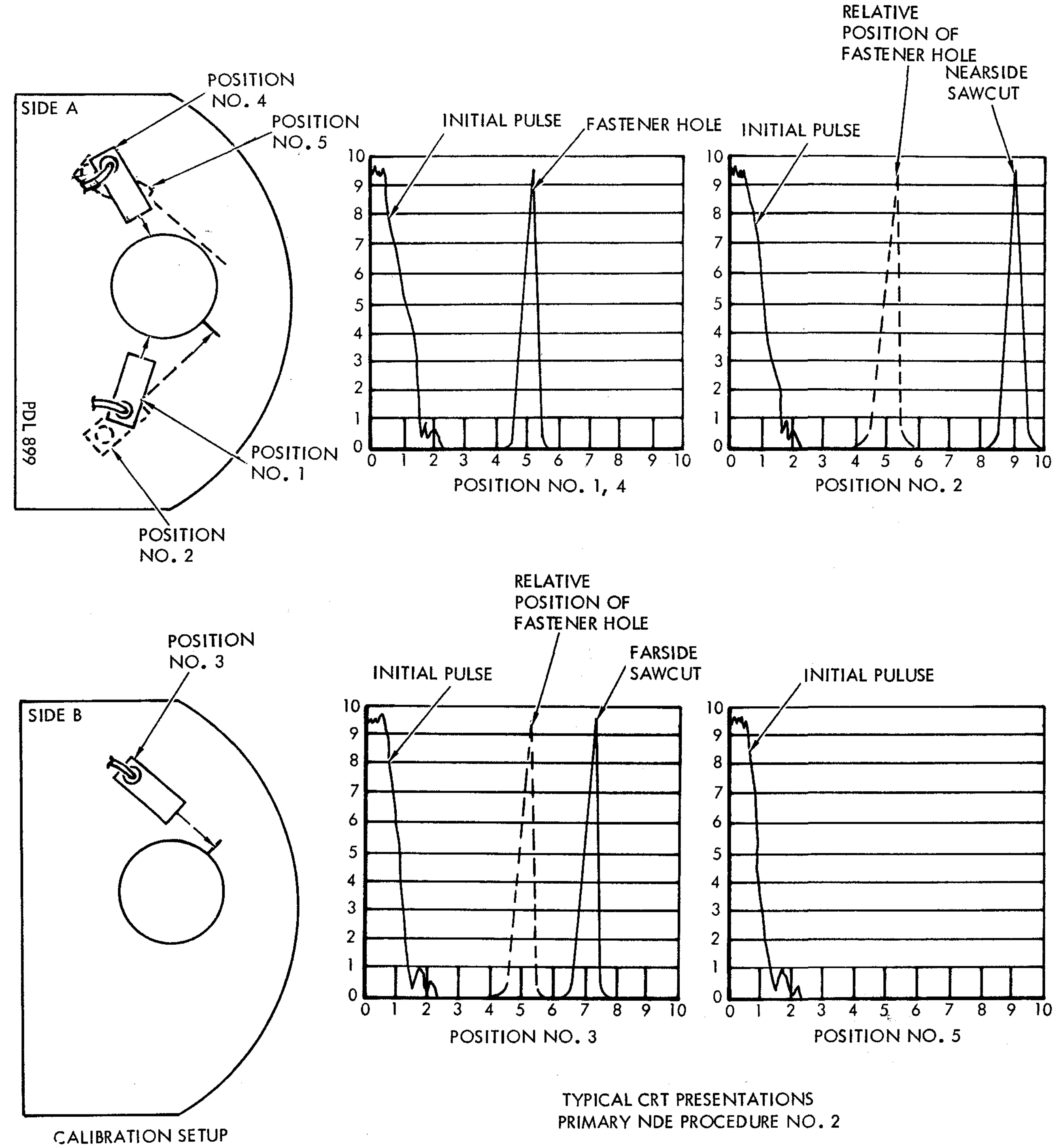
VIEW LOOKING AFT,
MLG AFT SUPPORT SPAR
INSPECTION - PRIMARY
NDE PROCEDURE NO. 2

NOTE

4 TRANSDUCER-SCAN FOR NEARSIDE AND FARSIDE DEFECTS IN FORWARD AND AFT SIDES OF CLEVIS. SCAN COMPLETELY AROUND FASTENER HOLE IN CLOCKWISE AND COUNTER-CLOCKWISE DIRECTIONS.

SMR151-58.2-0-003-4

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 4 of 8)

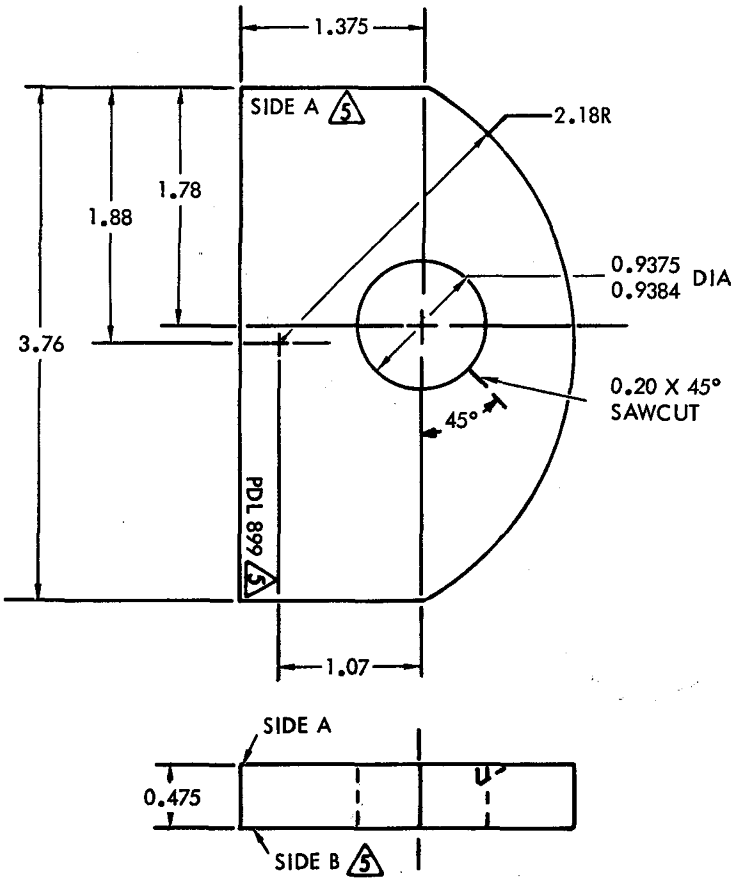


CALIBRATION SETUP

TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 2

SMR151-58.2-0-003-5

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 5 of 8)



CALIBRATION STANDARD, P/N PDL 899
(7075-T6 ALUMINUM OR EQUIVALENT)

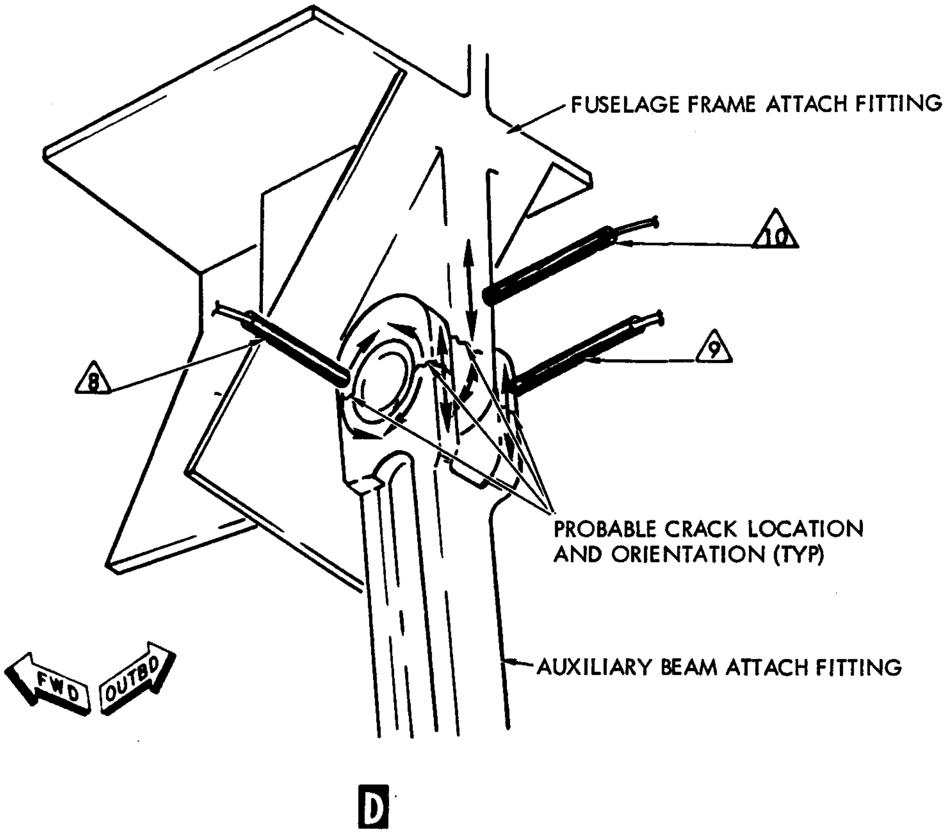
5 SCRIBE ON PART.

6. FINISH - WASH PRIMER, ZINC CHROMATE PRIMER, COROGARD 14

7. REFER TO SECTION 58.0 FOR ADDITIONAL FABRICATION INFORMATION.

SMR151-58.2-0-003-

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 6 of 8)


FUSELAGE FRAME ATTACH FITTING
10
9
8
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
AUXILIARY BEAM ATTACH FITTING
FWD
OUTBD
D

AUXILIARY BEAM AND FUSELAGE ATTACH FITTINGS INSPECTION - PRIMARY NDE PROCEDURE NO. 3

NOTE

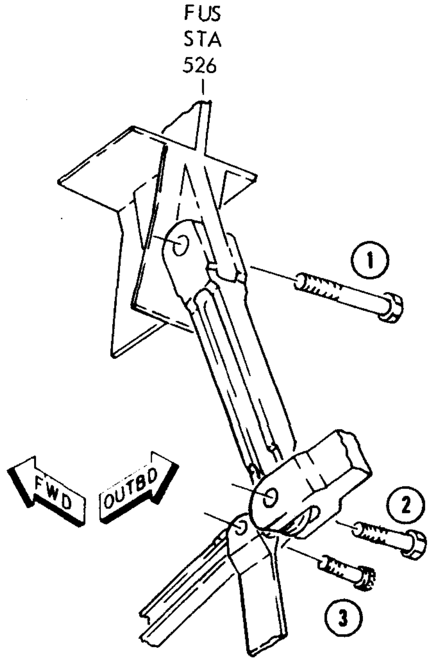
8 SURFACE PROBE-SCAN OUTER SURFACES OF CLEVIS IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION COMPLETELY AROUND BUSHED FASTENER HOLE ON BOTH FORWARD AND AFT SIDES.

9 SURFACE PROBE-SCAN ACCESSIBLE EDGES OF CLEVIS IN BOTH DIRECTIONS ON BOTH FORWARD AND AFT SIDES.

10 SURFACE PROBE-SCAN ACCESSIBLE EDGE OF FITTING IN BOTH DIRECTIONS.

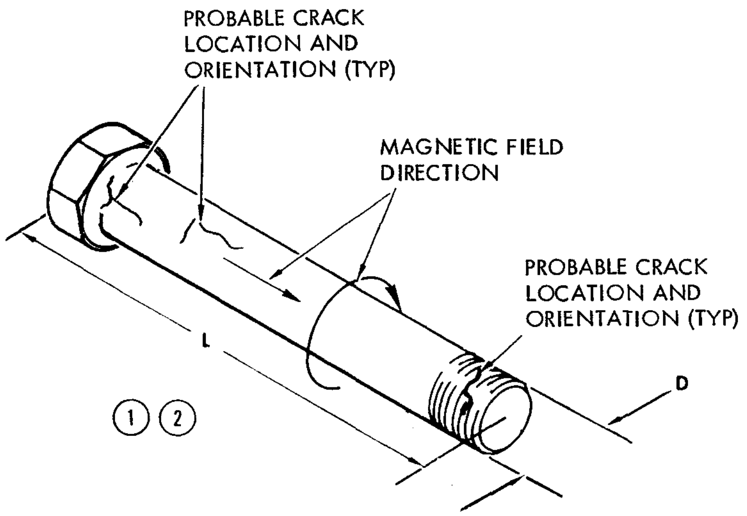
SMR151-58.2-0-003-7

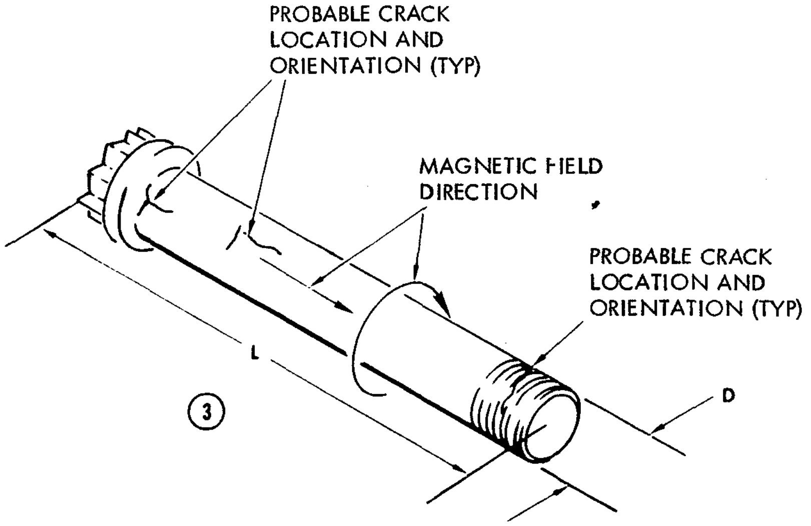
Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 7 of 8)



E

ATTACHMENT BOLT INSPECTION-
PRIMARY NDE PROCEDURE NO. 4


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
① ②


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
③

BOLT	D (IN)	L (IN)	MATERIAL	FINISH
① NAS1112-36D	0.75	3.2	△11	△12
② JW289-1 △14	0.75	2.9	△11	△12
③ MS21250-05024	0.31	2.4	△13	△12

NOTE

△11 ALLOY STEEL, 160,000 UTS, HEX HEAD
△12 CADMIUM PLATED
△13 ALLOY STEEL, 180,000 UTS, 12-POINT EXTERNAL WRENCHING
△14 SHOULDER BOLT

SMR151-58.2-0-003-8

Figure 58.2-3-6/4. MLG Aft Support Spar to Fuselage Attachment Connecting Linkage and Attachment Bolts (Sheet 8 of 8)

58.2.5 MLG AFT SUPPORT SPAR AND BEARING SUPPORT CAP.

58.2.5.1 DESCRIPTION. (See figure 58.2-4-4/2.) The main landing gear aft support spar is fabricated from a 7075-T6 aluminum forging and the aft bearing support cap is fabricated from 7075-T6 or 7075-T651 (preferred spare) aluminum plate. The spar and cap surfaces are finished with wash primer, zinc chromate primer, and Corogard 14. The bearing support cap is secured to the spar by eight fasteners.

58.2.5.2 DEFECTS. Cyclic gear loads resulting from taxiing, takeoffs, and landings may cause cracks to initiate at the fastener holes in the aft support spar and the bearing support cap. These cracks will propogate approximately vertically or horizontally from the fastener holes.

58.2.5.3 PRIMARY NDE PROCEDURE — EDDY CURRENT

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1/2-inch diameter, Ideal Specialty Co. P/N 6200-1/2-BH-BNC or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove access panels as required to gain access to aft side of the aft support spar. Refer to Section 52.3 of this manual.

(3) Jack airplane to remove load from landing gear per Section 7.0 of this manual.

c. Access: Access to the inspection area can be gained from underneath the airplane with appropriate access panels removed.

d. Preparation of part:

CAUTION

Do not remove more than four fasteners from the cap-to-spar attachment at any one time. Four fasteners in the pattern illustrated must be fully installed at all times. Failure to comply could result in damage to structure.

(1) Remove fasteners numbered 1, 3, 6, and 8 as illustrated leaving fasteners numbered 2, 4, 5, and 7 installed. Following bolt-hole inspection, reinstall fasteners numbered 1, 3, 6, and 8 and remove fasteners numbered 2, 4, 5, and 7. Following bolt-hole inspection, reinstall fasteners 2, 4, 5, and 7.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface areas as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

f. Inspection: Scan the entire inner surface of each fastener hole through all layers of material. Refer to Section 58.1 of this manual for bolt hole scanning of straight wall fastener holes. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.5.4 BACKUP NDE PROCEDURE — FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Remove the main landing gear strut assembly per the instructions in Section 32.0 of this manual.

c. Access: Access to the inspection area can be gained following removal of the main landing gear strut assembly.

Note

Preparation of airplane for the Backup NDE Procedure requires removal of the main landing gear strut assembly. This procedure includes removal of the aft bearing support cap.

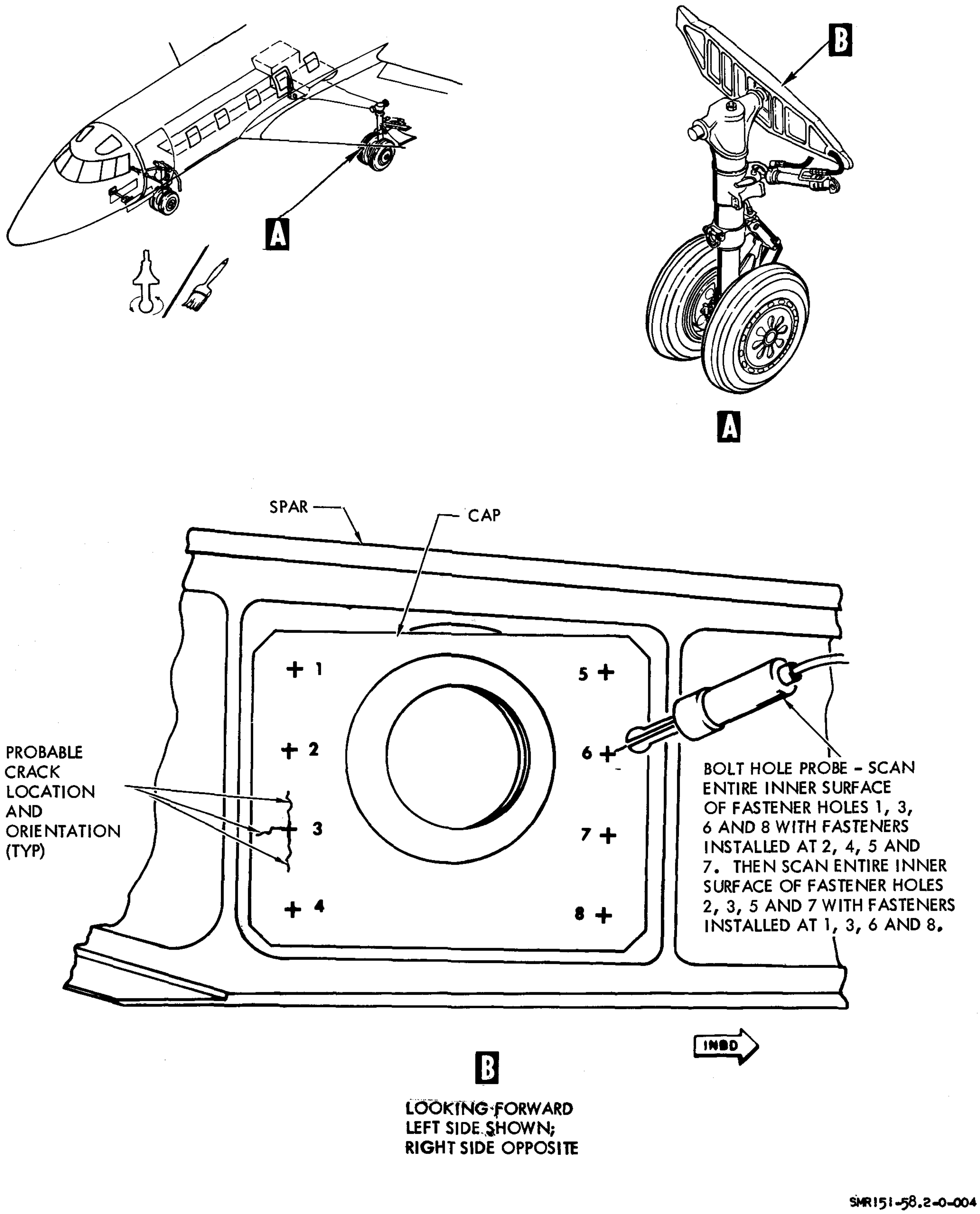
d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective areas in the spar and/or the cap.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.2.5.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



BOLT HOLE PROBE - SCAN ENTIRE INNER SURFACE OF FASTENER HOLES 1, 3, 6 AND 8 WITH FASTENERS INSTALLED AT 2, 4, 5 AND 7. THEN SCAN ENTIRE INNER SURFACE OF FASTENER HOLES 2, 3, 5 AND 7 WITH FASTENERS INSTALLED AT 1, 3, 6 AND 8.

Figure 58.2-4-4/2. MLG Aft Support Spar and Bearing Support Cap

58.2.6. MLG ACTUATOR AND AUXILIARY SPAR OUTBOARD SUPPORTING FITTING.

58.2.6.1. DESCRIPTION. (See figure 58.2-5-2/0.) The left and right main landing gear actuator attach fitting attaches to the wing rear beam structure outboard of the strut attachment point and provides the outboard connecting point for the side brace actuator cylinder and auxiliary spar. The fittings are machined from 7075 aluminum forgings, heat treated to T-6, shot peened, and finished with zinc chromate primer.

58.2.6.2. DEFECTS. Stress corrosion cracking may occur in the exposed cavities in the fitting.

Note

Either Primary NDE Procedure or Alternate Primary NDE Procedure must be performed.

58.2.6.3. PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, V.M. Products P/N VM202-1/8 or equivalent.

(3) Calibration Standard, Aluminum. (See figure 58.1-9.)

b. Preparation of airplane.

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Disconnect and lower the outboard end of the actuating cylinder in accordance with Section 32.0.

c. Access: Access to the fitting is from beneath the airplane up through the access door while the airplane is in the normal parked condition.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions prior to surface eddy current scanning.

e. Instrument settings/calibration: Refer to Section 58.1 for calibration of the crack detector.

f. Inspection: Surface scan the entire upper and outboard radius in the lower cavity of the fitting. Scan in both transverse and longitudinal directions. Refer to Section 58.1 for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in part being inspected.

g. Mark and report indicated defects.

h. To confirm suspected defects, perform the backup NDE Procedure. If no defect is suspected, proceed to system securing.

58.2.6.4. ALTERNATE PRIMARY NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean the inspection area in the upper and forward radius inside the lower cavity of the fitting.

e. Instrument Settings/Calibration: Not applicable.

Note

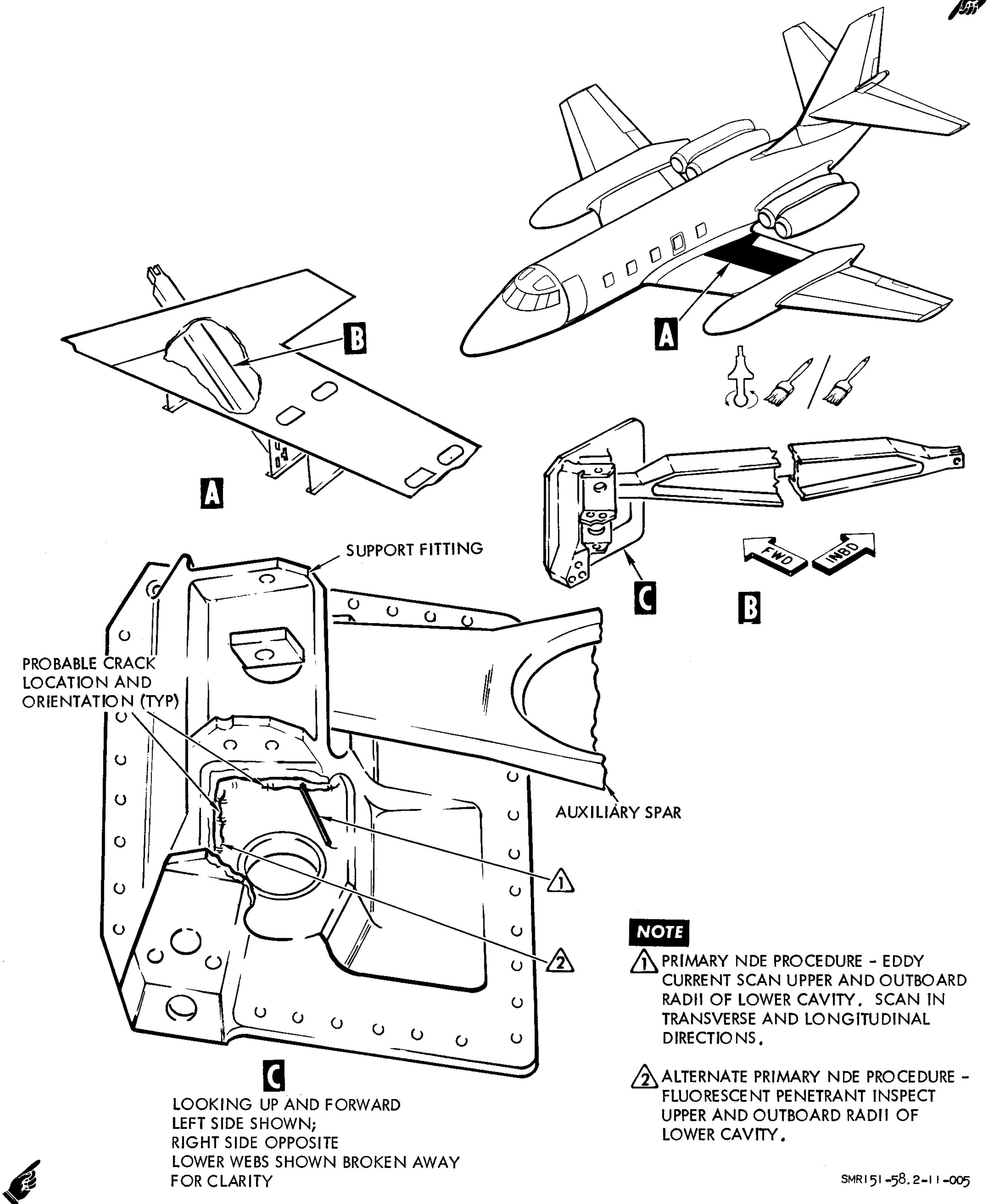
Existing cracks may have been previously stop-drilled.

f. Inspection: Inspect the entire upper and outboard radius in the lower cavity of the fitting. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. If existing cracks or new cracks are found, refer to Secltion 59.2 for structural requirements for cracks in the fitting. If no defects are found, proceed to System Securing.

58.2.6.5. BACKUP NDE PROCEDURE. If primary NDE procedure, eddy current, is performed, the alternate primary NDE procedure, fluorescent penetrant, must be performed as a backup. If alternate primary NDE procedure, fluorescent penetrant, is performed, no backup is required.

58.2.6.6. SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



LOOKING UP AND FORWARD
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE
LOWER WEBS SHOWN BROKEN AWAY
FOR CLARITY

NOTE

1 PRIMARY NDE PROCEDURE - EDDY CURRENT SCAN UPPER AND OUTBOARD RADII OF LOWER CAVITY. SCAN IN TRANSVERSE AND LONGITUDINAL DIRECTIONS.

2 ALTERNATE PRIMARY NDE PROCEDURE - FLUORESCENT PENETRANT INSPECT UPPER AND OUTBOARD RADII OF LOWER CAVITY.

SMR151-58.2-11-005

Figure 58.2-5-2/0. MLG Actuator and Auxiliary Spar Outboard Support Fitting

58.2.7 TRAILING EDGE FLAP ACTUATOR SUPPORT, FLAP STA 19 AND 58.

58.2.7.1 DESCRIPTION (See figure 58.2-6-4/2.) The trailing edge flap actuator supports are attached to the wing rear spar and support the actuating mechanism for the trailing edge flaps. The supports are fabricated from 7075-T6 aluminum forgings, receive chemical film protective treatment, and are finished with zinc chromate primer.

58.2.7.2 DEFECTS. The trailing edge flap actuator supports at flap stations 19 and 58 are susceptible to stress-corrosion cracking in the aft lug area. These cracks will propogate horizontally from the bushed holes aft to the end of the lugs.

58.2.7.3 PRIMARY NDE PROCEDURE — EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM 102AS or equivalent.

(3) Calibration standard, aluminum — (See figure 58.1-9.)

b. Preparation of airplane:

(1) Extend the trailing edge flaps to the full down position in accordance with Section 27.4 of this manual.

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(2) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(3) Disconnect the flap support link from the flap actuator support and rotate the link aft sufficiently to clear the flap actuator support lugs.

c. Access: Access to the inspection area can be gained while the airplane is in the normal parked condition and the trailing edge flaps are in the full down position.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the inner and outer surfaces of the aft lug areas as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

f. Inspection: Inspect the inner and outer surfaces of the aft lug areas by scanning in a clockwise and counterclockwise direction completely around the outer diameter of the bushing.

Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.7.4 BACKUP NDE PROCEDURE — FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure. In addition, remove the suspected defective flap actuator support in accordance with instructions in Section 27.4 of this manual.

c. Access: Access to the inspection area requires removal of the suspected defective flap actuator support.

d. Preparation of part:

(1) Remove the bushings from the aft lug areas of the flap actuator support in accordance with Section 59.1 of this manual.

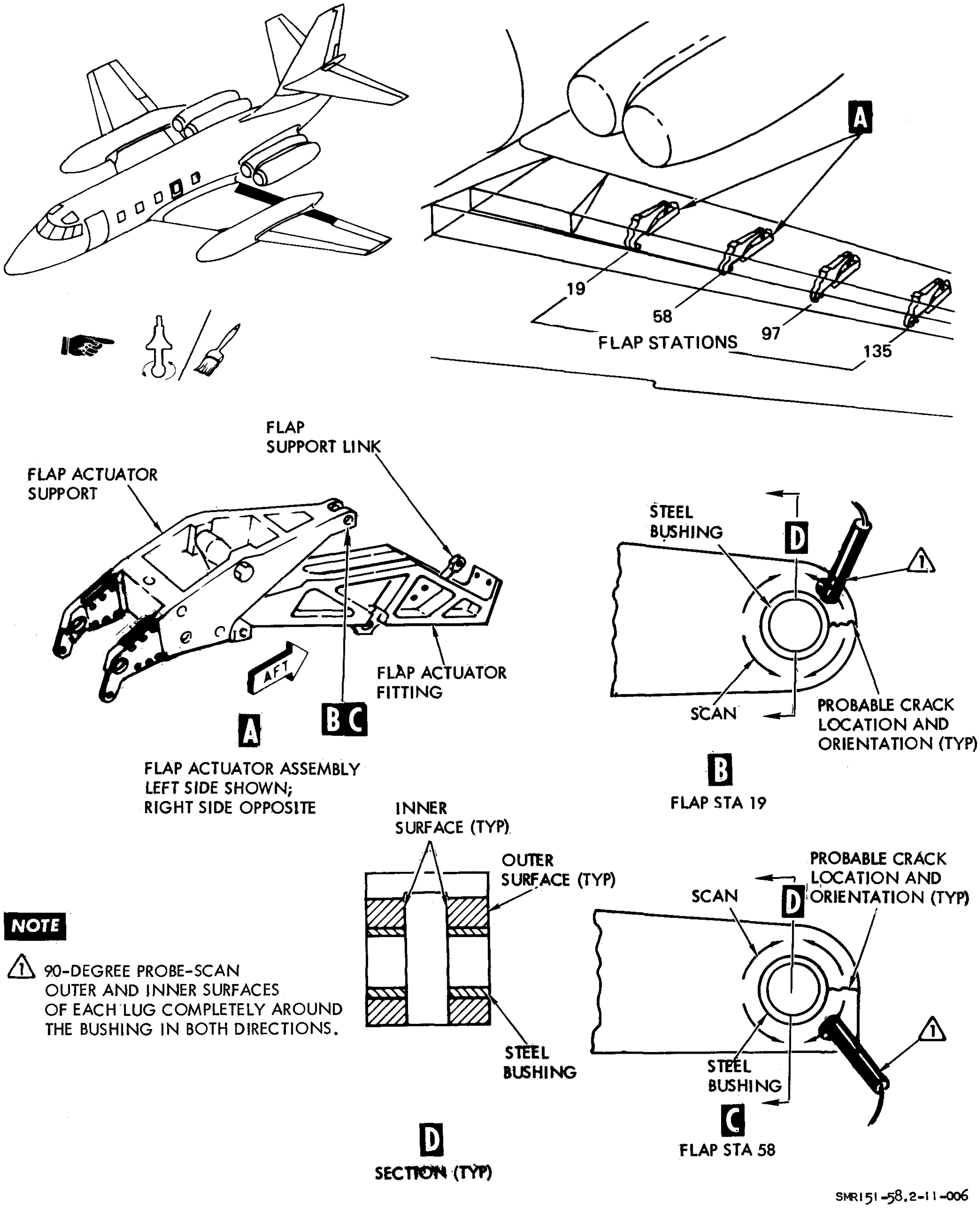
(2) Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective area in the flap actuator support.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.2.7.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

⚠1 90-DEGREE PROBE-SCAN OUTER AND INNER SURFACES OF EACH LUG COMPLETELY AROUND THE BUSHING IN BOTH DIRECTIONS.

Figure 58.2-6-4/2. Trailing Edge Flap Actuator Support, Flap Sta 19 and 58

58.2.8 EXTERNAL FUEL TANK TO WING ATTACHMENT BOLTS AND FITTINGS.

58.2.8.1 DESCRIPTION (See figure 58.2-7-4/2) The external fuel tank is attached to the wing front and rear beams at Wing Station 154. The tank has three major sections: the forward section, the aft section and the bottom fairing. The forward section extends on the top and bottom of the wing to the rear beam where it joins with the aft section. The attachment fittings on the wing front and rear beams and the forward bulkhead of the external fuel tank forward section are fabricated from 7075-T6 aluminum alloy extrusions. The attachment fittings on the aft ends of the external fuel tank forward section are fabricated from 2014-T6 aluminum alloy forgings. All fittings are finished with zinc chromate primer. The attachment bolts are fabricated from high strength alloy steel and are cadmium plated.

58.2.8.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the fasteners and the attachment fittings. Cracks in the fittings will initiate in the fastener hole and propagate outward in random direction. Circumferential and longitudinal cracks will originate in the thread root areas, in the shank sections, in the radius beneath the heads and in the heads of the attachment bolts.

Note

Both Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.2.8.3 PRIMARY NDE PROCEDURE NO. 1 — EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1/4-inch diameter, Ideal Specialty Co. P/N 6200-1/4 BH-BNC or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products, P/N VM102AS or equivalent.

(4) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM100SS or equivalent.

(5) Calibration Standard, aluminum. (See figure 58.1-9.)

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane:

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

c. Access: The attachment fittings are accessible with the external fuel tank removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the fastener hole and/or surface areas of the attachment fittings on the external fuel tank and the wing front and rear beam areas as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Front Beam Attachment: Using the 90-degree or pencil probe, scan around the bushed holes in a clockwise and counterclockwise direction on the inboard and outboard sides of the external fuel tank attach fittings on the wing front beam. Using the 90-degree probe, scan outer and inner surfaces completely around the bushed holes of each lug of each external tank forward attach fitting.

(2) Rear Beam Attachment: Using the bolt hole probe, scan the entire inner surface of both fastener holes in the external fuel tank fittings and the mating holes on the aft side of the wing rear beam.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed with Primary NDE Procedure No. 2.

58.2.8.4 PRIMARY NDE PROCEDURE NO. 2 — MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB1619 or equivalent.

58.2.8.4 (Cont).

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The attachment bolts are accessible with the external fuel tank removed.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contiminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

JWT 56-1 - 500 amperes

NAS 1104-14 - 250 amperes

(2) Longitudinal field:

JWT 56-1 - 2100 amperes

NAS 1104-14 - 1700 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. If no defects are indicated, proceed to System Securing.

58.2.8.5 BACKUP NDE PROCEDURE – FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

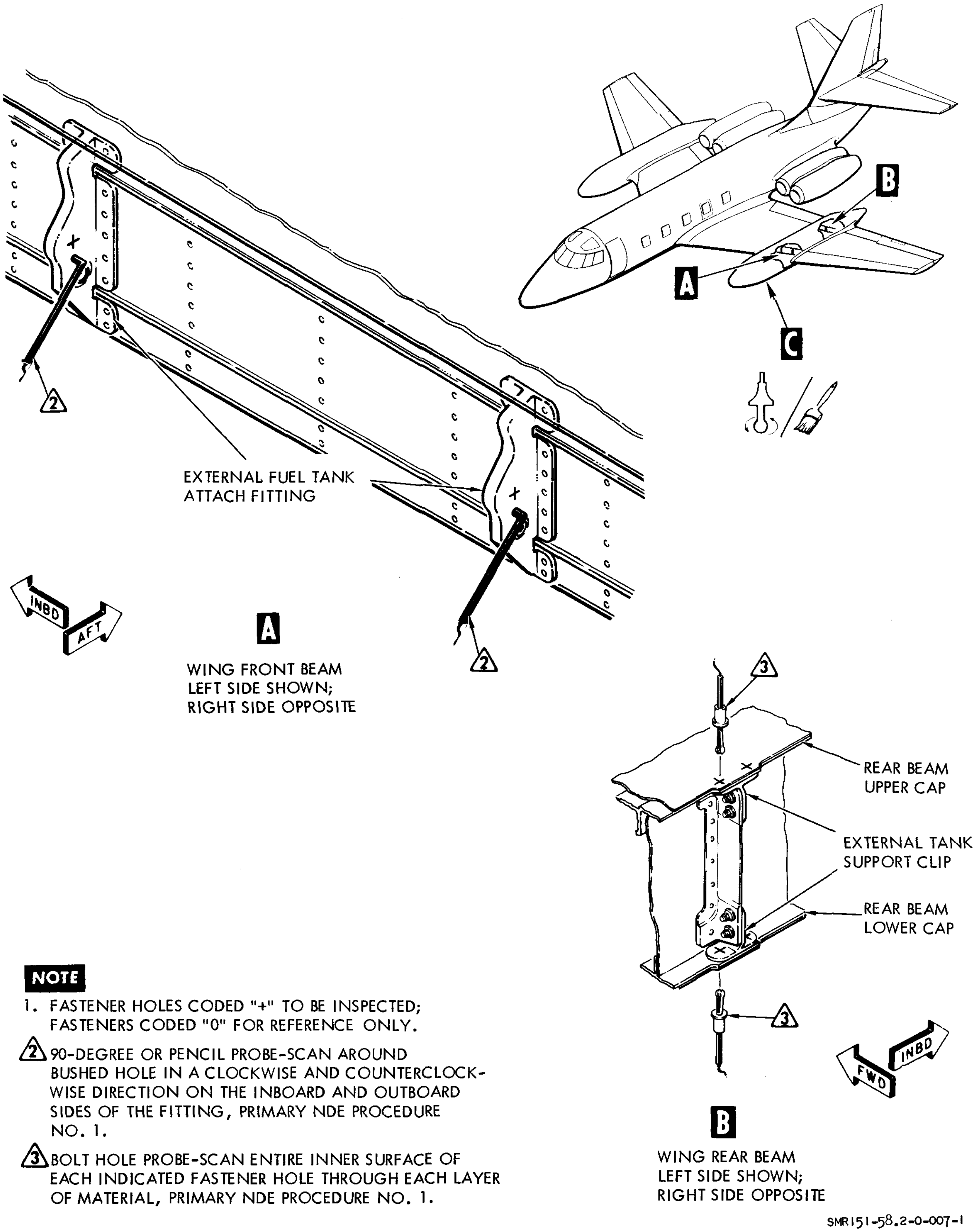
d. Preparation of part: If installed, remove the bushing in accordance with Section 59.1 of this manual. Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspectiton: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.2.8.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



WING FRONT BEAM
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

WING REAR BEAM
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

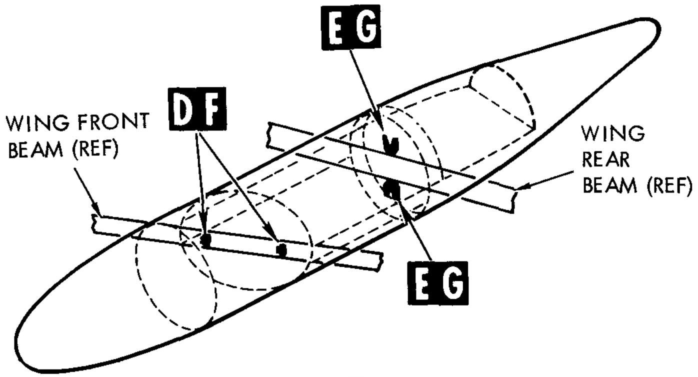
NOTE

1. FASTENER HOLES CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

△2 90-DEGREE OR PENCIL PROBE-SCAN AROUND BUSHED HOLE IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION ON THE INBOARD AND OUTBOARD SIDES OF THE FITTING, PRIMARY NDE PROCEDURE NO. 1.

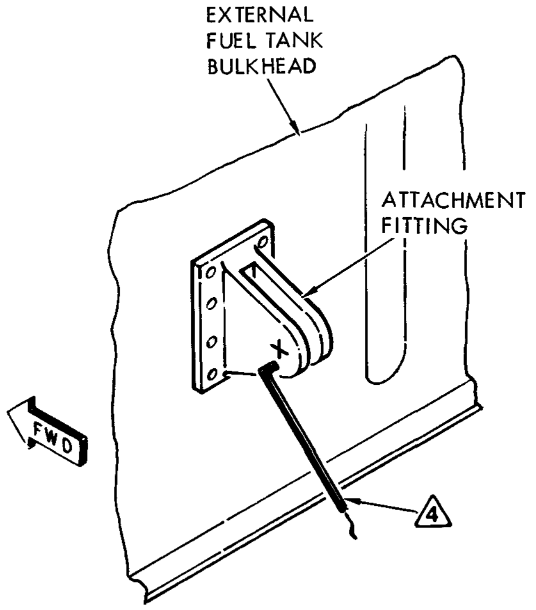
△3 BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF EACH INDICATED FASTENER HOLE THROUGH EACH LAYER OF MATERIAL, PRIMARY NDE PROCEDURE NO. 1.

Figure 58.2-7-4/2. External Fuel Tank to Wing Attachment Bolts and Fittings (Sheet 1 of 3)



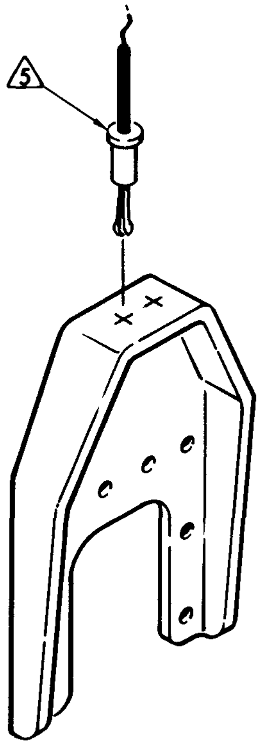
C

EXTERNAL FUEL TANK
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



D

EXTERNAL TANK FORWARD
ATTACH FITTING
(TYP 2 PLACES)



E

EXTERNAL TANK AFT
ATTACH FITTING
(TYP 2 PLACES)

NOTE

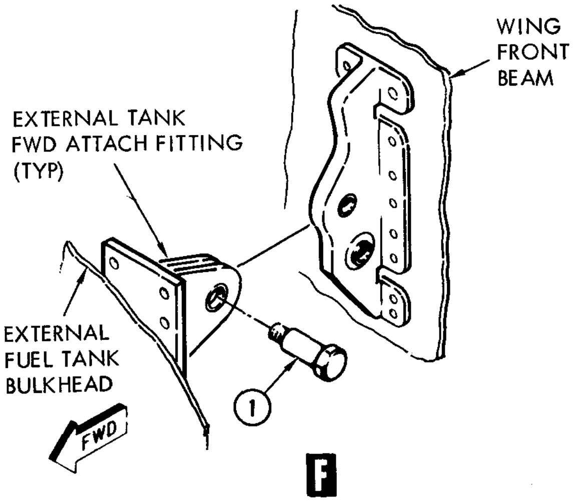
1. FASTENER HOLES CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

△4 90-DEGREE PROBE-SCAN OUTER AND INNER SURFACES OF EACH LUG COMPLETELY AROUND THE BUSHED HOLE IN A CLOCKWISE AND COUNTER-CLOCKWISE DIRECTION, PRIMARY NDE PROCEDURE NO. 1.

△5 BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF EACH INDICATED FASTENER HOLE, PRIMARY NDE PROCEDURE NO. 1.

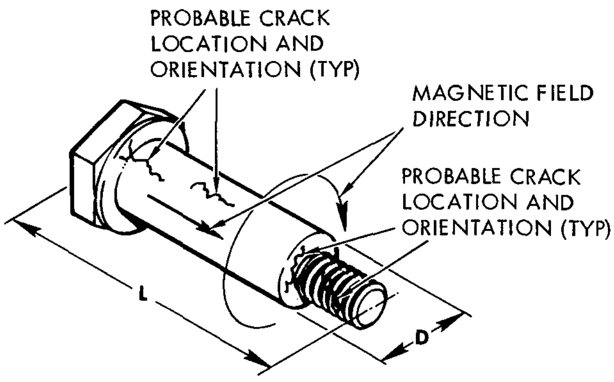
SMR151-58.2-0-007-2

Figure 58.2-7-4/2. External Fuel Tank to Wing Attachment Bolts and Fittings (Sheet 2 of 3)

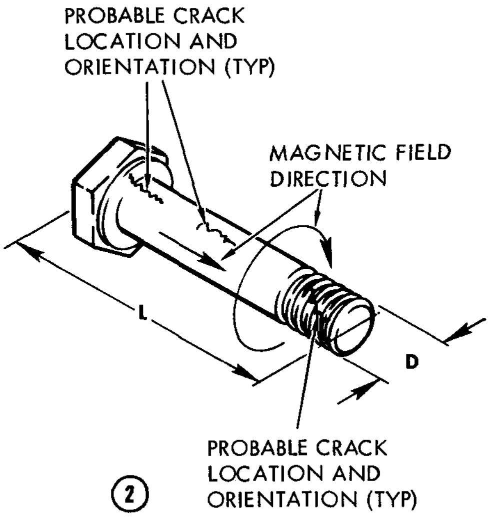


F

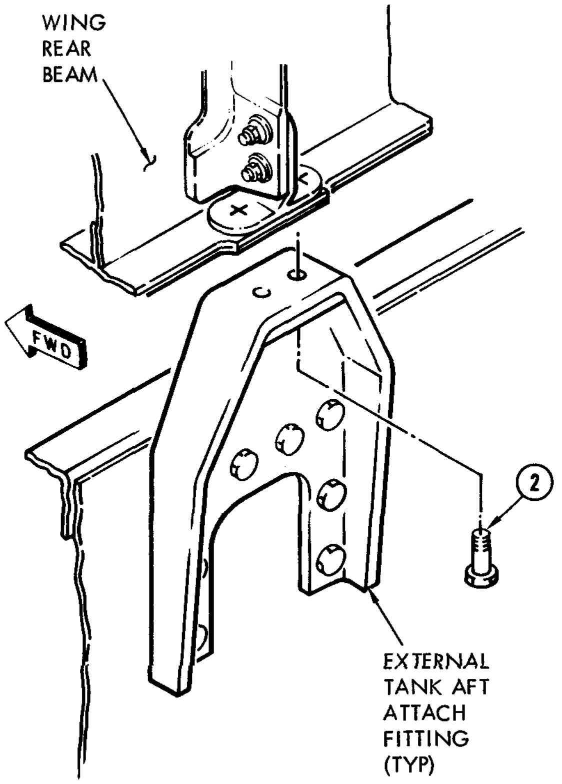
FORWARD ATTACHMENT BOLT INSPECTION
PRIMARY NDE PROCEDURE NO. 2



①



②



G

AFT ATTACHMENT BOLT INSPECTION-
PRIMARY NDE PROCEDURE NO. 2

	BOLT	D (IN)	L (IN)	MATERIAL	FINISH
1	JWT56-1	0.499	2.17	△6	△7
2	NAS1104-14	0.250	1.32	△6	△7

NOTE

△6 ALLOY STEEL, HEX HEAD.

△7 CADMIUM PLATED

SMR151-58.2-0-007-3

Figure 58.2-7-4/2. External Fuel Tank to Wing Attachment Bolts and Fittings (Sheet 3 of 3)

58.2.9 WING PANEL RISERS.

58.2.9.1 DESCRIPTION. (See figure 58.2-8-5/2.) The extruded wing panels having integral "I" section risers form the upper and lower surfaces of the wings. These panels are 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162), or 7075-T76511 (replacement panels) extrusions and are machined and chemically milled to gradually reduce the cross sectional area as they extend outboard. The lower surface panels are attached to the shear beams and ribs with flush rivets and hi-loks, and are sealed completely to form the lower boundary of the integral tanks. The upper surface panels are attached to the ribs and shear beams with flush screws, and are removable. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.9.2 DEFECTS. Stress corrosion cracks may occur in the root area of the integral wing panel risers at the inboard and outboard ends of the risers and propagate in a spanwise direction.

58.2.9.3 PRIMARY NDE PROCEDURE NO. 1 — ULTRASONIC.

a. NDE equipment:

(1) Ultrasonic Flaw Detector, Sonic Model FTS Mark I, or equivalent.

(2) Transducer, Longitudinal Wave, 5.0 MHz, 0.250 inch diameter, Pitch-catch, Krautkramer-Branson Aerotech Model FDU, with cable, or equivalent.

(3) Transducer, Longitudinal Wave, 5.0 MHz, 0.250 inch diameter, Pitch-catch, Krautkramer-Branson Aerotech Model FDU, with cable, modified as illustrated, or equivalent.

(4) Couplant, water.

(5) Spray bottle for couplant, local purchase.

(6) Calibration standard, Lockheed P/N PDL954 as illustrated.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the upper and lower wing-to-fuselage fillet fairings from the wing front beam to the wing rear beam.

c. Access: Access to the inspection area can be gained while the airplane is in the normal parked condition and with appropriate fairing panels removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the wing surfaces as required to permit good contact between part and transducer.

Note

Calibration may be used for either the standard or modified transducer.

Note

Instrument must be operated in the "Thru Transmission" mode when using a Pitch/Catch transducer.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer to the calibration standard in Position No. 1 as illustrated.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT display. Adjust sweep to display signal from the panel back surface near the center of the CRT.

(3) Slide the transducer to Position No. 2 and note that the signal from the sawcut is near the center of the CRT, immediately to the right of the panel back surface signal.

(4) Maximize the signal from the sawcut by adjusting the transducer position and adjust the signal amplitude to obtain 98 percent saturation on the CRT display.

(5) Slide the transducer to Position No. 3 and note the loss of signal.

(6) Re-scan Positions No. 1, No. 2 and No. 3 to note the signals received from each area and the shift of the signal on the CRT when moving across the riser.

(7) The CRT signal from the sawcut simulates a signal from a crack in the riser.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 in this manual.

58.2.9.3 (Cont).

Note

Know where the risers begin and end because inspection beyond these points gives misleading results which could be interpreted as crack indications. Know where on the CRT to look for the crack indication and what to ignore. Scan slowly so that crack indications are not overlooked. Make certain there is no friction between the search unit and the inspection surface. Be careful not to mistake the area beyond the end of the riser as a cracked riser. The indications are similar. Be careful not to apply excess couplant. A thin film is all that is required to remove any friction during scanning. Excess couplant will produce erratic results. Be careful not to orient the search unit incorrectly with respect to the line of the riser. This will greatly affect amplitude, and a crack could be overlooked.

Note

All risers are spaced 4.0 inches apart on parallel lines to the mid beam butt joint.

Note

The area under the external fuel tanks, under the boot overlap on the lower surface, and under the external doubler on the lower surface aft of the leading edge flaps will be considered inaccessible and not requiring inspection.

f. Inspection:

(1) Risers:

(a) Using the standard or modified transducer, apply couplant and inspect 48 inches of the inboard end and 36 inches of the outboard end of each riser on the upper and lower surface where access permits. The inboard riser locations are 2, 3, 4, 6, 7, 8, 10, 11, 12, 14, 15, 16, 18, 19, 22, 23, 25, 26, 27, 29, 30, 31, 33, 34, 35, 37, and 38. Outboard risers are 3, 6, 18, 23, 25, 26, and 27.

(b) Slowly scan in a fore and aft direction across the vertical leg of the riser with the transducer connectors aligned at right angles to the direction of the riser.

(c) A signal similar to the signal received from the sawcut during calibration will indicate a probable crack in the riser.

Note

The modified transducer is designed to clear the mismatch that often exists along a spanwise splice and provide proper coupling between transducer and panel surface.

(2) Spanwise splice risers:

(a) Using the modified transducer, apply couplant and inspect 48 inches of the inboard end and 36 inches of the outboard end of each spanwise splice riser on the upper and lower surface where access permits. The inboard spanwise splice riser locations are 5, 13, 17, 24, 28, and 36. Outboard spanwise splice riser locations are 5, 28, and 36.

(b) Slowly scan in a fore and aft direction to and from the splice joint with the transducer connectors aligned at right angles to the direction of the spanwise splice riser.

(c) A signal similar to the signal received from the sawcut during calibration will indicate a probable crack in the spanwise splice riser.

g. Mark and report suspected defects.

h. For disposition of suspected defects, refer to Chapter 5 in this manual. If no defects are suspected, proceed to System Securing.

58.2.9.4 PRIMARY NDE PROCEDURE NO. 2 — ULTRASONIC.

a. NDE equipment: Same as Primary NDE Procedure No. 1

b. Preparation of airplane: In addition to Primary NDE Procedure No. 1, perform the following:

(1) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(2) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

CAUTION

The wing must be supported in a neutral stress position prior to removal of the upper surface wing panels in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(3) Remove the integral fuel tank top access panel in accordance with Section 28.2 of this manual.

58.2.9.4 (Cont).

c. Access: Access to the inspection areas is gained with the external fuel tanks and integral fuel tank top access panels removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the inspection areas as required to permit good contact between part and transducer.

Note

Calibration may be used for either the standard or modified transducer.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer to the calibration standard in Position No. 1 as illustrated.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse at the left side of the CRT display. Adjust sweep to display signal from the flange back surface near the left side of the CRT.

(3) Slide the transducer to Position No. 2 and note that the signal from the panel back surface is near the right side of the CRT.

(4) Slide the transducer to Position No. 3 and note that the signal from the saw cut is near the right side of the CRT immediately to the left of the panel back surface signal.

(5) Maximize the signal from the sawcut by adjusting the transducer position and adjust the signal amplitude to obtain 98 percent saturation on the CRT display.

(6) Re-scan Positions No. 1, No. 2, and No. 3 to note the signals received from each area and the shift of the signal on the CRT when moving across the riser.

(7) The CRT signal from the sawcut simulates a signal from a crack in the riser.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 in this manual.

Note

The ultrasonic instrument must be calibrated in accordance with instructions in Primary NDE Procedure No. 1 to perform the external inspection and in accordance with instructions in this Primary NDE Procedure to perform the internal inspection.

f. Inspection:

(1) External inspection. Inspect the risers and spanwise splice risers from the external panel surface as follows:

(a) Inspect inboard 48 inches of all risers and spanwise splice risers on the upper and lower surfaces except risers 9, 32, and 39 in accordance with instructions in Primary NDE Procedure No. 1.

(b) Inspect outboard 36 inches of all risers and spanwise splice risers on the upper and lower surfaces except risers 7 through 12, 29 through 34, and 39 in accordance with instructions in Primary NDE Procedure No. 1.

(2) Internal inspection. Inspect the risers and spanwise splice risers from the internal riser flange surfaces as follows:

(a) Using the standard or modified transducer, apply couplant and inspect the areas not inspected during the external inspection.

(b) Slowly scan in a fore and aft direction across the vertical leg of the riser with the transducer connectors aligned at right angles to the direction of the riser.

(c) A signal similar to the signal received from the sawcut during calibration will indicate a probable crack in the riser.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.2.9.5 BACKUP NDE PROCEDURE — FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from the approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 2.

c. Access: Same as Primary NDE Procedure No. 2.

d. Preparation of part:

(1) If required, remove the angle(s) attached to the risers with the suspect defect.

(2) Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective area of the riser.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. For disposition of indicated defects, refer to Chapter 5 in this manual.

58.2.9.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

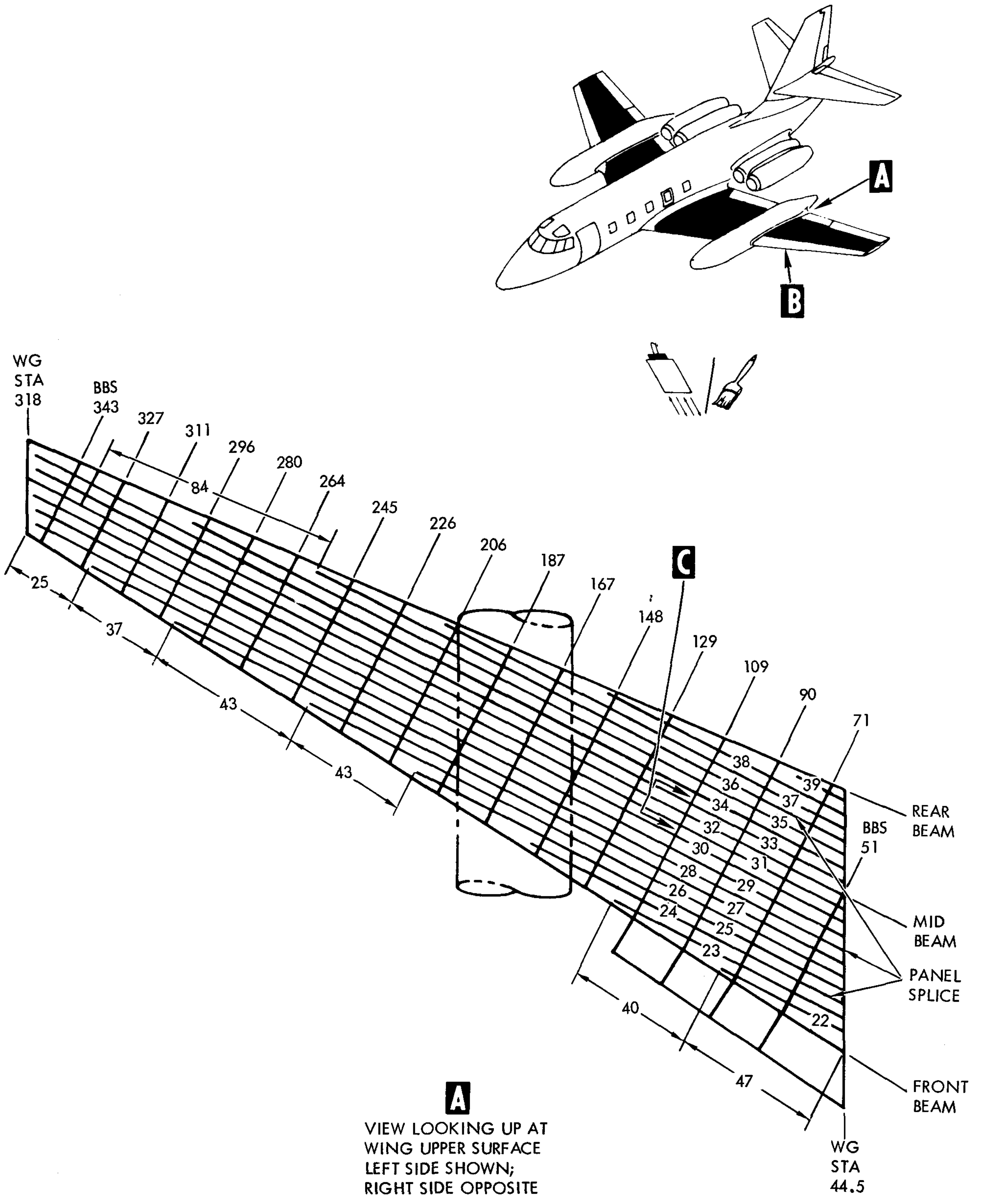


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Figure 58.2-8-5/2. Wing Panel Risers (Sheet 1 of 6)

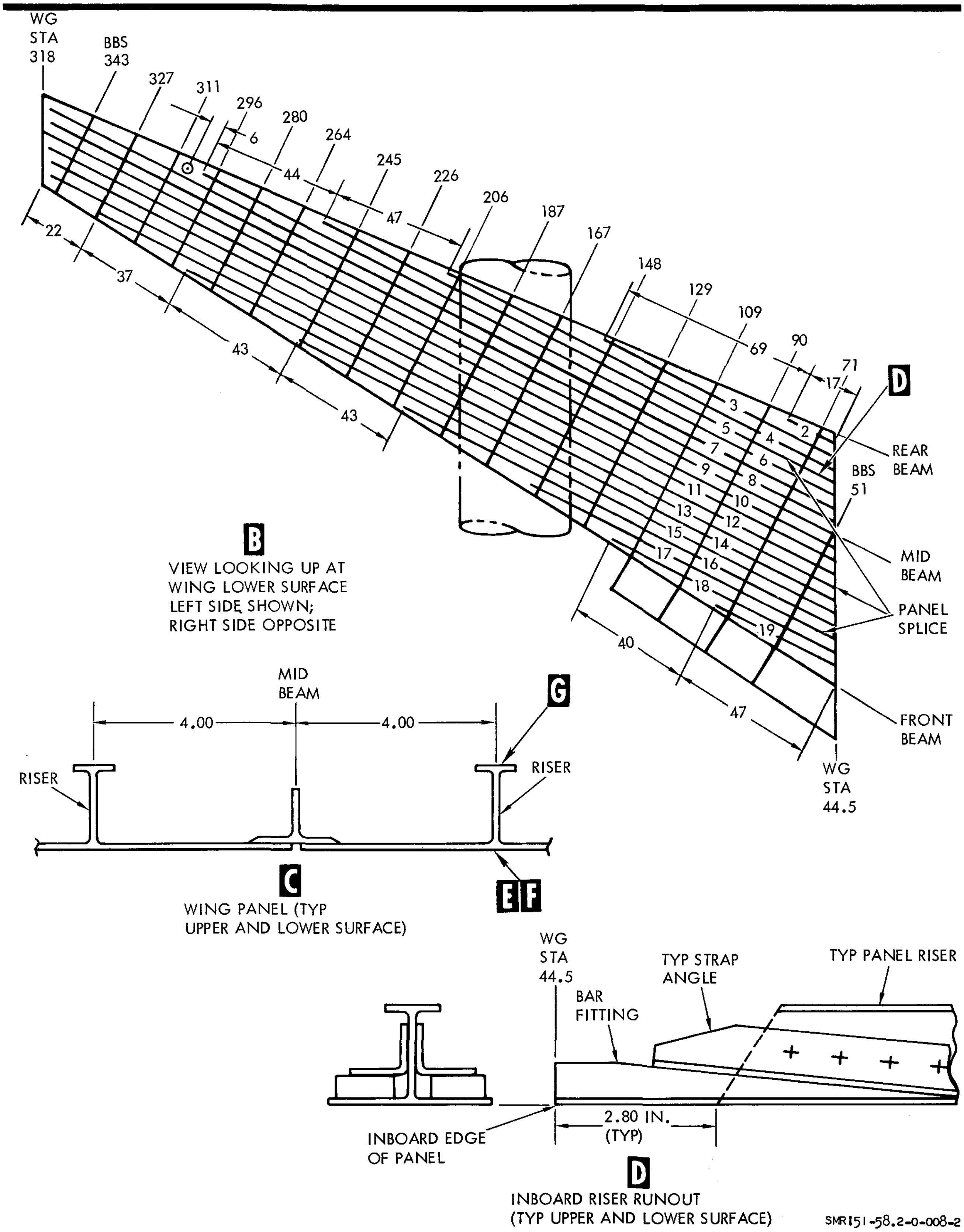
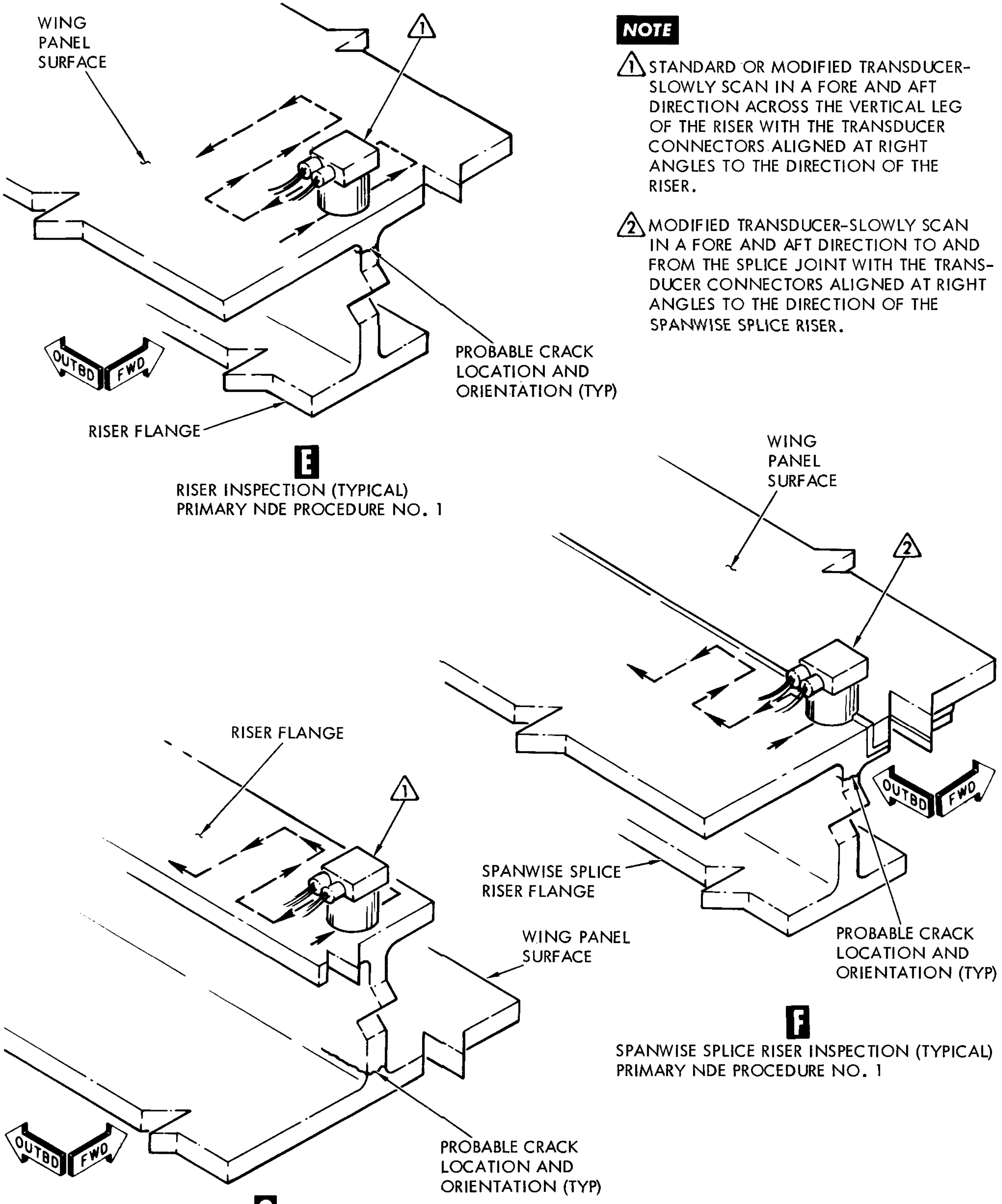


Figure 58.2-8-5/2. Wing Panel Risers (Sheet 2 of 6)

NOTE

1. STANDARD OR MODIFIED TRANSDUCER-SLOWLY SCAN IN A FORE AND AFT DIRECTION ACROSS THE VERTICAL LEG OF THE RISER WITH THE TRANSDUCER CONNECTORS ALIGNED AT RIGHT ANGLES TO THE DIRECTION OF THE RISER.

2. MODIFIED TRANSDUCER-SLOWLY SCAN IN A FORE AND AFT DIRECTION TO AND FROM THE SPLICE JOINT WITH THE TRANSDUCER CONNECTORS ALIGNED AT RIGHT ANGLES TO THE DIRECTION OF THE SPANWISE SPLICE RISER.


WING PANEL SURFACE
1
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
OUTBD
FWD
RISER FLANGE
E
WING PANEL SURFACE
2
OUTBD
FWD
SPANWISE SPLICE RISER FLANGE
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
F
RISER FLANGE
1
WING PANEL SURFACE
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
OUTBD
FWD
G

E RISER INSPECTION (TYPICAL)
PRIMARY NDE PROCEDURE NO. 1

F SPANWISE SPLICE RISER INSPECTION (TYPICAL)
PRIMARY NDE PROCEDURE NO. 1

G INTERNAL INSPECTION (TYPICAL)
PRIMARY NDE PROCEDURE NO. 2

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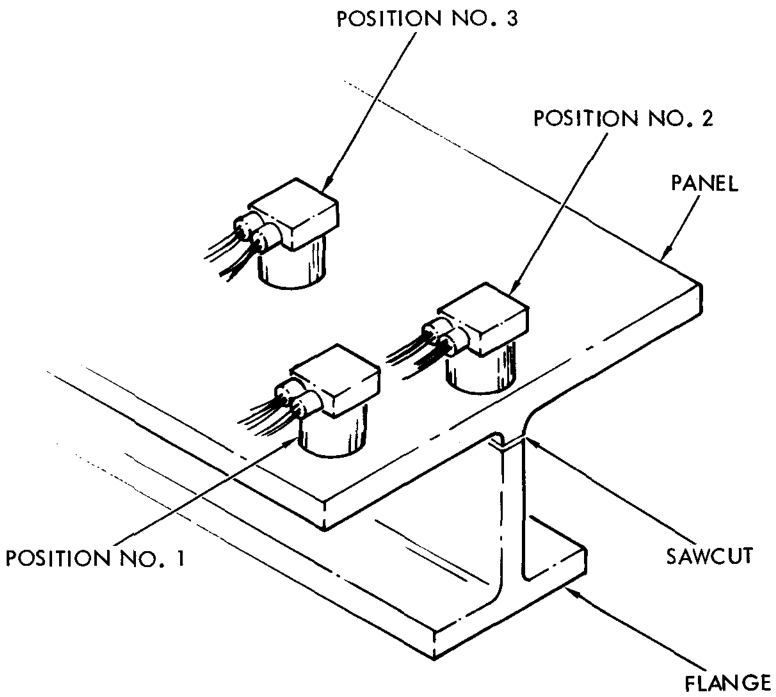
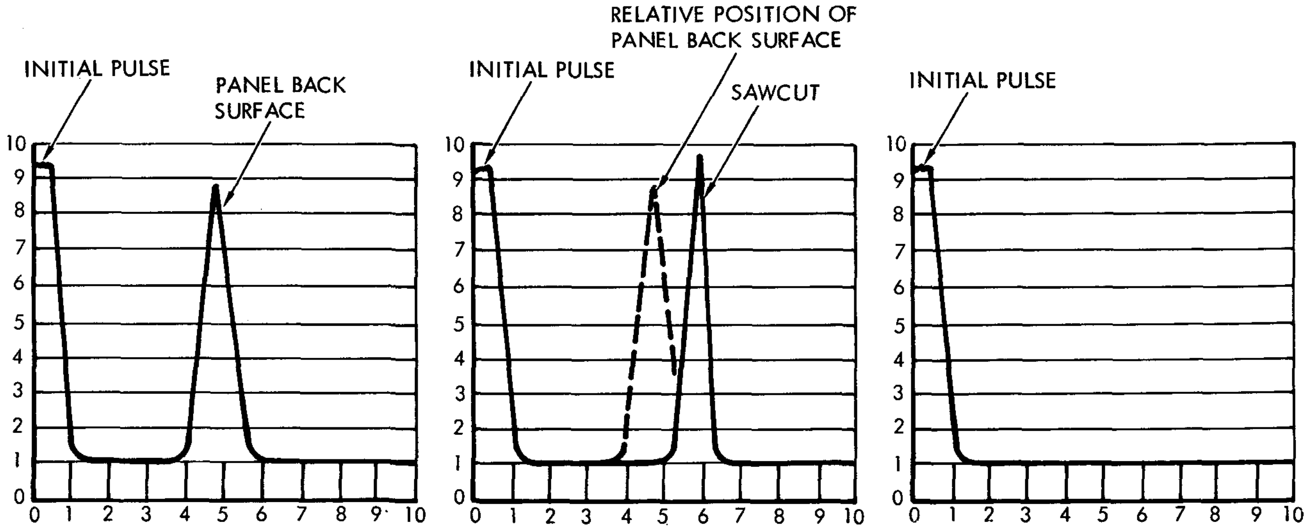
Figure 58.2-8-5/2. Wing Panel Risers (Sheet 3 of 6)



CALIBRATION SETUP
PRIMARY NDE PROCEDURE NO. 1

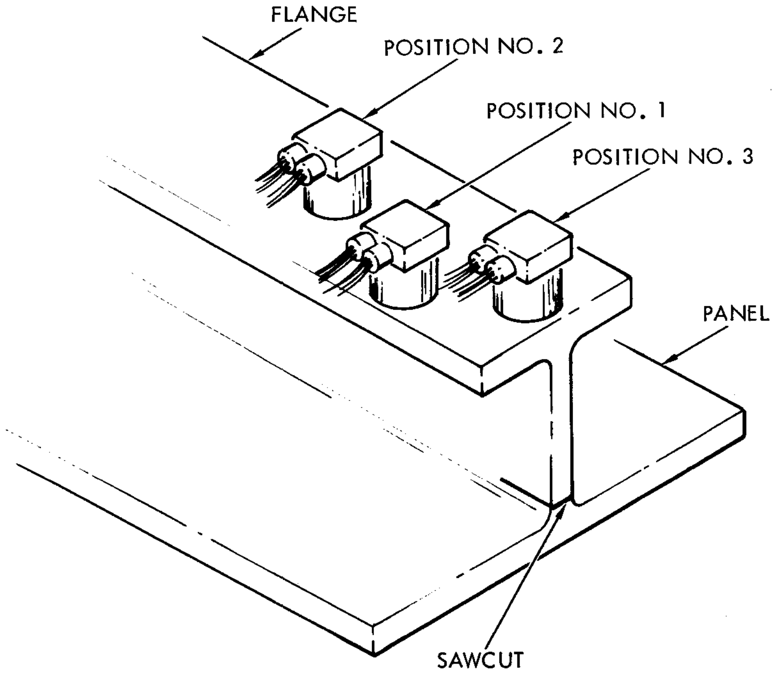


POSITION NO. 1 POSITION NO. 2 POSITION NO. 3

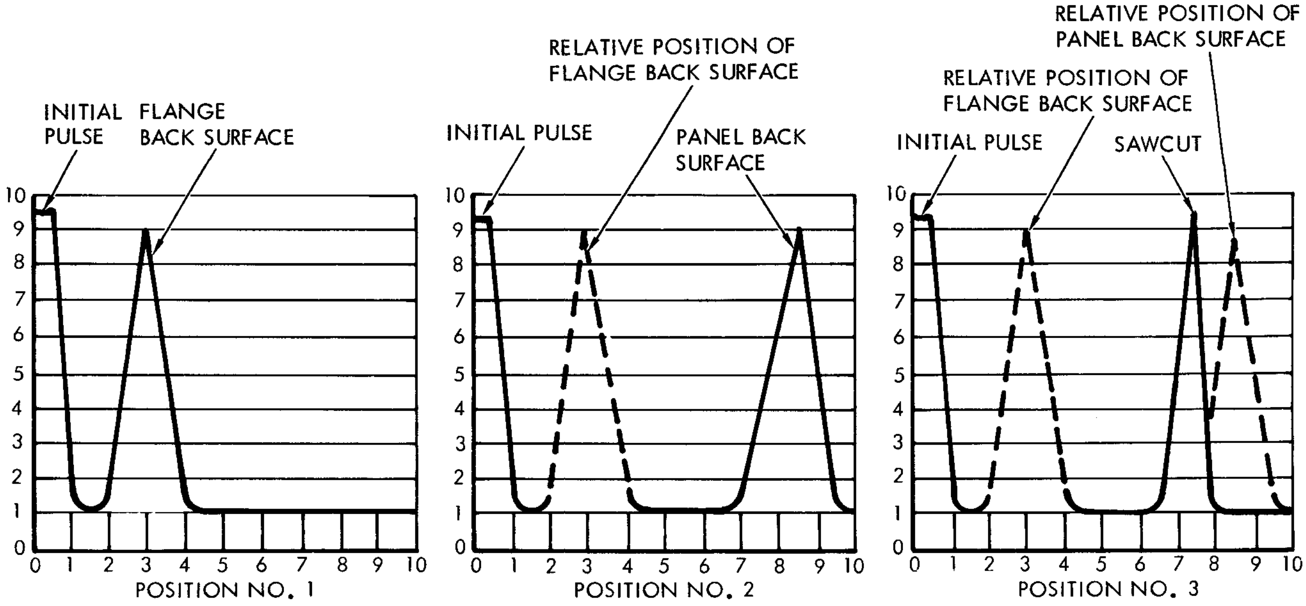
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 1

SMR151-58.2-0-008-4

Figure 58.2-8-5/2. Wing Panel Risers (Sheet 4 of 6)



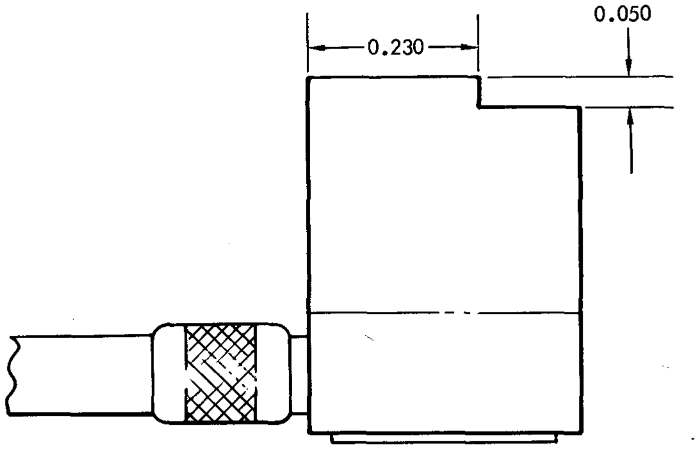
CALIBRATION SETUP
PRIMARY NDE PROCEDURE NO. 2



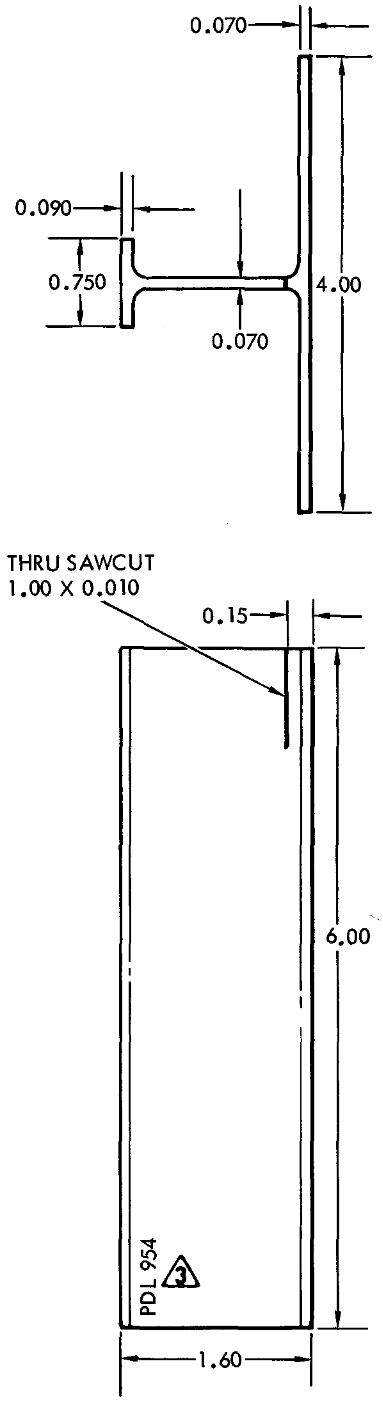
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 2

SMR151-58.2-0-008-5

Figure 58.2-8-5/2. Wing Panel Risers (Sheet 5 of 6)



MODIFIED TRANSDUCER



CALIBRATION STANDARD
MATERIAL: 7075-T6 ALUMINUM

NOTE

△3 SCRIBE ON PART

SMR151-58.2-0-008-6

Figure 58.2-8-5/2. Wing Panel Risers (Sheet 6 of 6)

58.2.10 AILERON COUNTERWEIGHT REGION.

58.2.10.1 DESCRIPTION. (See figure 58.2-9-6/4). The ailerons are aluminum alloy construction consisting of front and rear beams, formed ribs and chemically milled skins. Spot welding and flush rivets are used to assemble the structure. The aileron is attached to the upper cap of the wing rear beam. The depleted uranium brass or sintered tungsten counterweight is located at the outboard end of the aileron. The leading edge skins, ribs, and former are 2024-T4 clad, the upper and lower skins are 7075-T6 clad and the cover assembly on the outboard end is 6061-T6 clad. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.10.2 DEFECTS. Cyclic operation of the aileron may cause fatigue cracks in the fastener holes and in the bend radius of the formed ribs, fittings and channel.

58.2.10.3 PRIMARY NDE PROCEDURE — RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane: Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

c. Access: Aileron surfaces are accessible with aircraft in the parked position.

d. Preparation of part: No special preparation is required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.2.10.4 BACKUP NDE PROCEDURE — EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, General Purpose, 1/8-inch diameter, Ideal Specialty Co. P/N 6100-1/8-S-BNC or equivalent.

(3) Calibration Standard, Aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure with the following additional requirements.

(1) Remove the aileron with the suspected defect.

(2) Disassemble the aileron as required for access to the area of the suspect defect.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole or area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

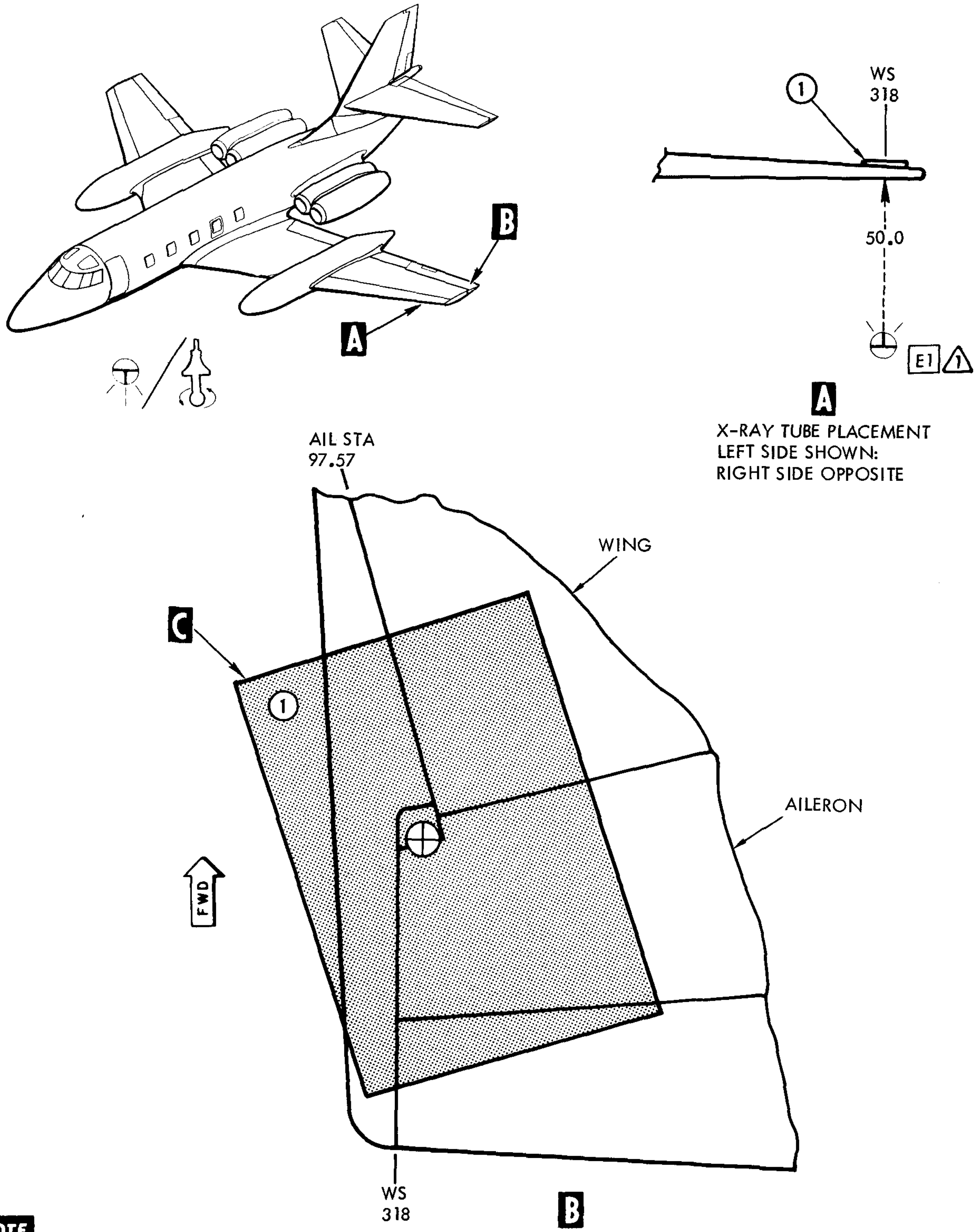
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the surface of the suspected defective area. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.2.10.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



A

X-RAY TUBE PLACEMENT
LEFT SIDE SHOWN:
RIGHT SIDE OPPOSITE

B

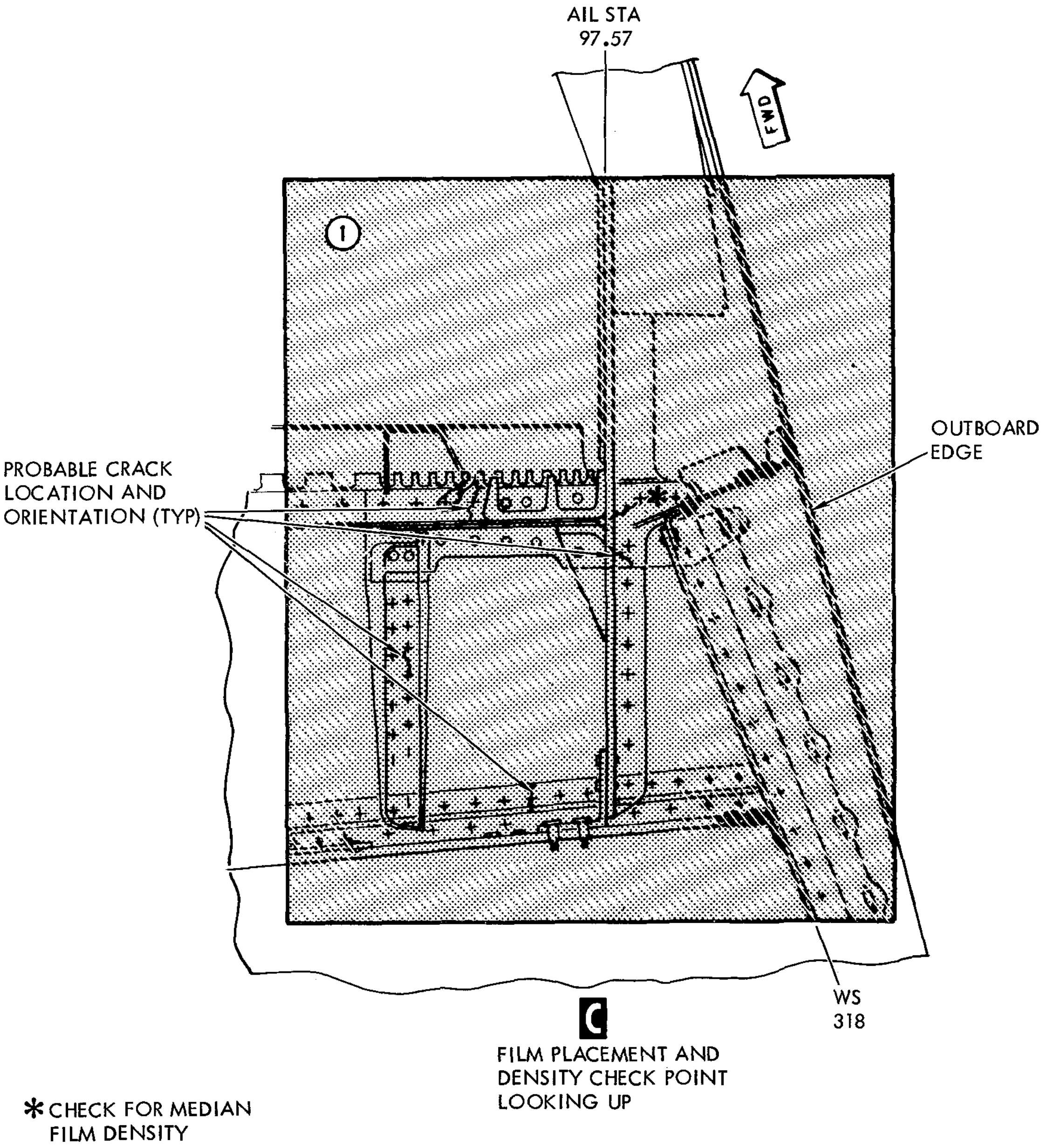
FILM PLACEMENT AND AIMING POINT
LOOKING DOWN LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

NOTE

1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND AIMED AT CENTER OF FILM.

SMR151-58.2-0-009-1

Figure 58.2-9-6/4. Aileron Counterweight Region (Sheet 1 of 2)


AIL STA
97.57
FWD
①
OUTBOARD EDGE
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
*
WS 318
C
FILM PLACEMENT AND DENSITY CHECK POINT LOOKING UP
* CHECK FOR MEDIAN FILM DENSITY

RADIOGRAPHIC INSPECTION DATA

EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	100	5	50.0	120	NDT 45	14 X 17	EXPOSE ①

SMR151-58.2-0-009-2

Figure 58.2-9-6/4. Aileron Counterweight Region (Sheet 2 of 2)

58.2.11 WING LOWER SURFACE SPANWISE SPLICES.

58.2.11.1 DESCRIPTION. (See figure 58.2-10-4/2.) The wing lower surface consists of five extruded panels joined together at spanwise splices with countersunk fasteners. The panels are attached to shear beams and ribs with flush rivets and hi-loks, and are sealed completely to form the lower boundary of the integral fuel tanks. These panels are 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162), or 7075-T76511 (replacement panels) extrusions and are machined and chemically milled to gradually reduce the cross sectional area as they extend outboard. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.11.2 DEFECTS. Fatigue cracks may occur in the lower member of the lap splice joint at the fastener holes. Cracks will initiate in the fastener hole and propagate in the chordwise direction.

58.2.11.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM100SS or equivalent.

(3) Calibration Standard, aluminum (See figure 58.1-9).

(4) Straight edge, local purchase.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

c. Access: The inspection area is accessible from underneath the wing with the external fuel tanks removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean local inspection areas on the lower surface as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Locate the straight edge parallel to the row of fasteners to permit eddy current scanning as near to the fastener as possible. Using the surface probe, scan along the forward and aft sides of the fastener holes in an inboard and outboard direction. Scan the external panel in the lower surface panel splice joint at:

Front Beam to Panel No. 1	BBS 89-148
Panel No. 1 to Panel No. 2	BBS 89-148
Panel No. 2 to Panel No. 3	BBS 70-210
Panel No. 4 to Panel No. 5	BBS 89-210

(2) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the panel.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.11.4 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

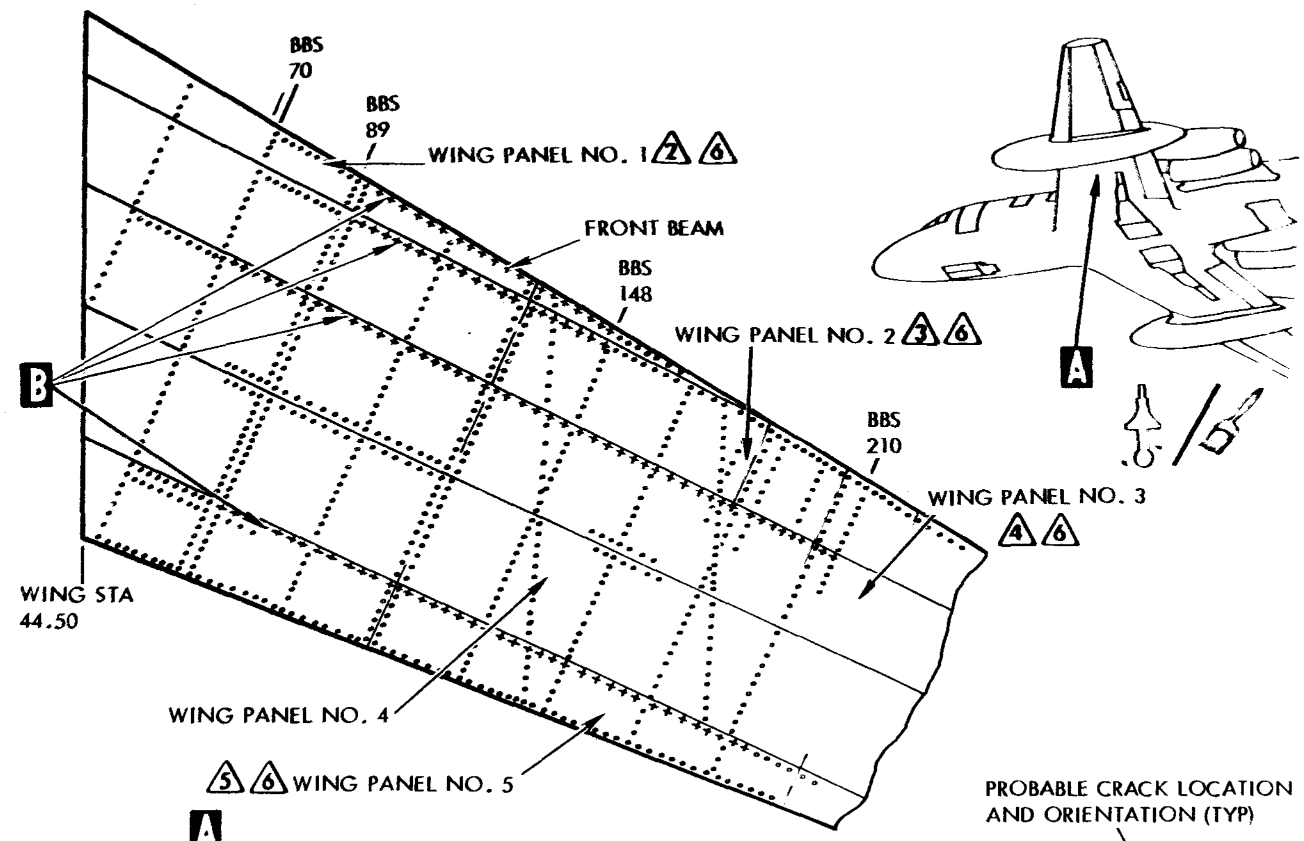
d. Preparation of part: Same as for Primary NDE Procedure except, refer to Section 58.0 of this manual and strip finish and clean suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

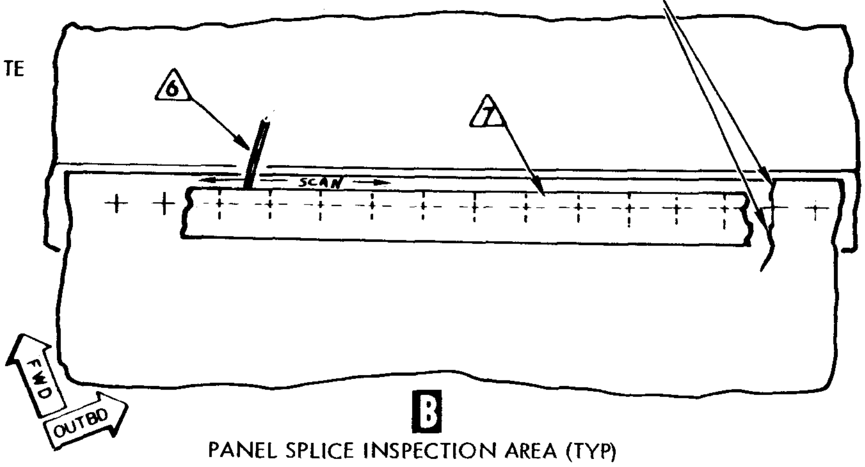
g. Mark and report indicated defects.

58.2.11.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components and perform operational checkouts as required in accordance with applicable sections of this manual.



A

LOWER SURFACE - LOOKING UP
LEFT WING SHOWN; RIGHT WING OPPOSITE



B

PANEL SPLICE INSPECTION AREA (TYP)

NOTE

1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

△2 INSPECT PANEL NO. 1 ALONG FASTENERS COMMON TO FRONT BEAM CAP FROM BBS 89 TO BBS 148.

△3 INSPECT PANEL NO. 2 ALONG FASTENERS COMMON TO PANEL NO. 1 FROM BBS 89 TO BBS 148.

△4 INSPECT PANEL NO. 3 ALONG FASTENERS COMMON TO PANEL NO. 2 FROM BBS 70 TO BBS 210.

△5 INSPECT PANEL NO. 5 ALONG FASTENERS COMMON TO PANEL NO. 4 FROM BBS 89 TO BBS 210.

△6 SHIELDED PENCIL PROBE - SCAN IN AN INBOARD AND OUTBOARD DIRECTION ON FORWARD AND AFT SIDES OF INDICATED FASTENERS.

△7 STRAIGHT EDGE SHOWN ALIGNED WITH FORWARD EDGE OF FASTENER ROW FOR SCANNING FORWARD SIDE OF FASTENERS. ALIGN STRAIGHT EDGE WITH AFT EDGE OF FASTENER ROW FOR SCANNING AFT SIDE OF FASTENERS.

SMR151-58.2-0-010

Figure 58.2-10-4/2. Wing Lower Surface Spanwise Splices

58.2.12 WING PANEL TO REAR BEAM LOWER CAP SPANWISE SPLICE, BBS 75 TO BBS 148.

58.2.12.1 DESCRIPTION. (See figure 58.2-11-4/4.) Wing panel number 5 is attached to the forward leg of the rear beam lower cap. The joint is a lap splice with the wing panel being the lower member. The panel is 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162) or 7075-T76511 (replacement panel) extrusion and is machined and chemically milled to gradually reduce the cross sectional area as it extends outboard. The rear beam lower cap is machined from 7075-T6 aluminum extrusion. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.12.2 DEFECTS. Fatigue cracks may occur in the fastener holes common to the wing panel and rear beam cap. Cracks will originate in the fastener holes and propagate in a chordwise direction.

Note

Primary NDE Procedure No. 1 is used to inspect for cracks on the exterior surface of the wing panel. Primary NDE Procedure No. 2 is used to inspect for cracks in both layers. Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.2.12.3 PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded Pencil, High Sensitivity, VM Products P/N VM100PSM or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane: Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

c. Access: Access to the inspection area can be gained while the airplane is in the normal parked condition.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the wing surface as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Locate the straight edge parallel to fasteners in a way which will permit eddy current scanning as near to the fastener as possible. Scan along the forward and aft sides of the fastener holes on the wing panel common to the rear beam from BBS 75 outboard to BBS 148.

(2) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the panel.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.2.12.4 PRIMARY NDE PROCEDURE NO. 2 RADIOGRAPHIC

WARNING

RADIATION HAZARD

Ensure compliance with all safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(3) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

WARNING

The wing must be supported in a neutral stress position prior to removal of the upper surface wing panels in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(4) Remove the integral fuel tank top access panel in accordance with Section 28.2 of this manual.

c. Access: Access to the inspection areas is gained with the external fuel tanks and integral fuel tank access panels removed.

d. Preparation of part: No special preparation is required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.12.5 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products Co. P/N VM101B-3/16 or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure No. 2.

c. Access: Same as Primary NDE Procedure No. 2.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.2.12.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

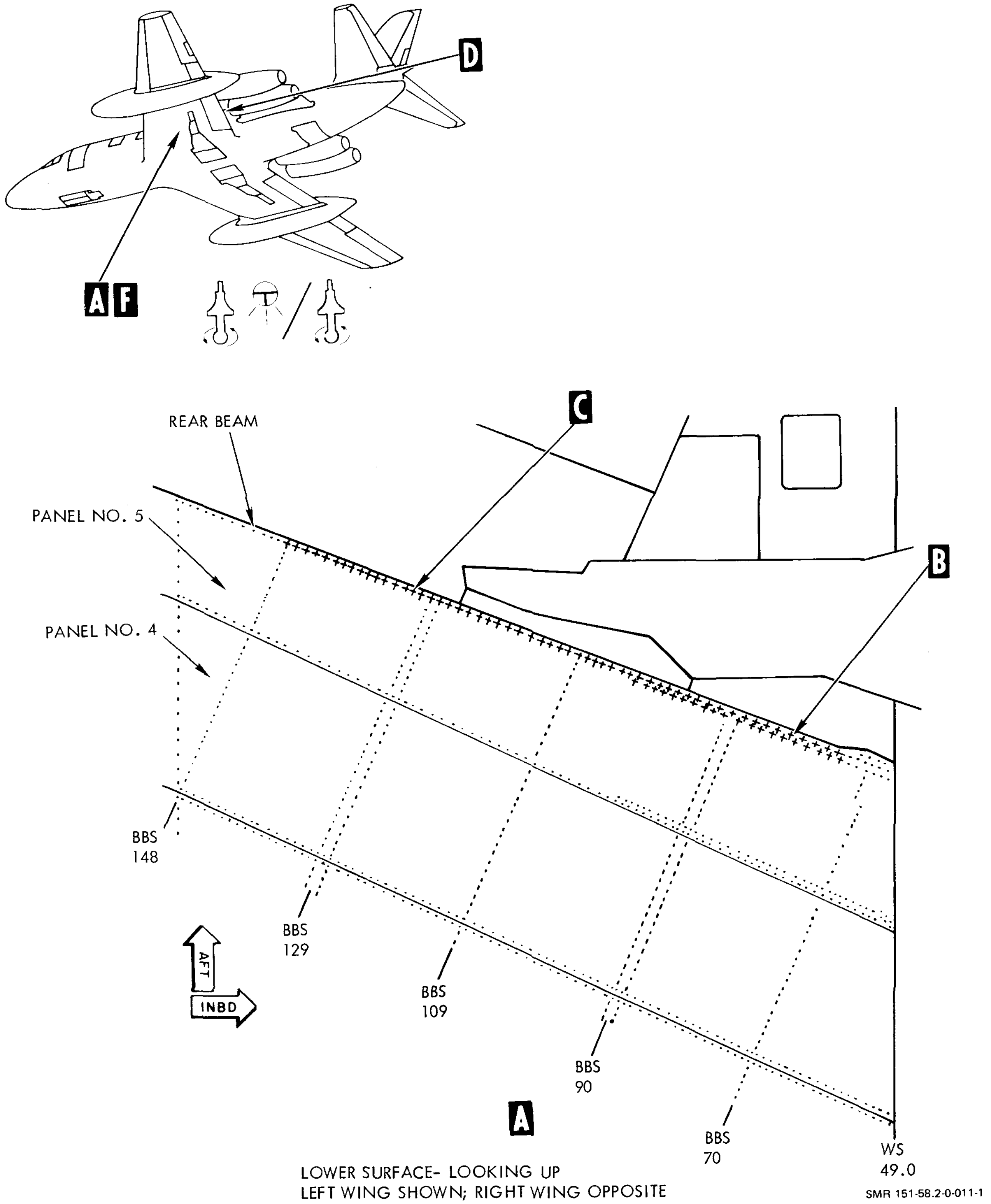
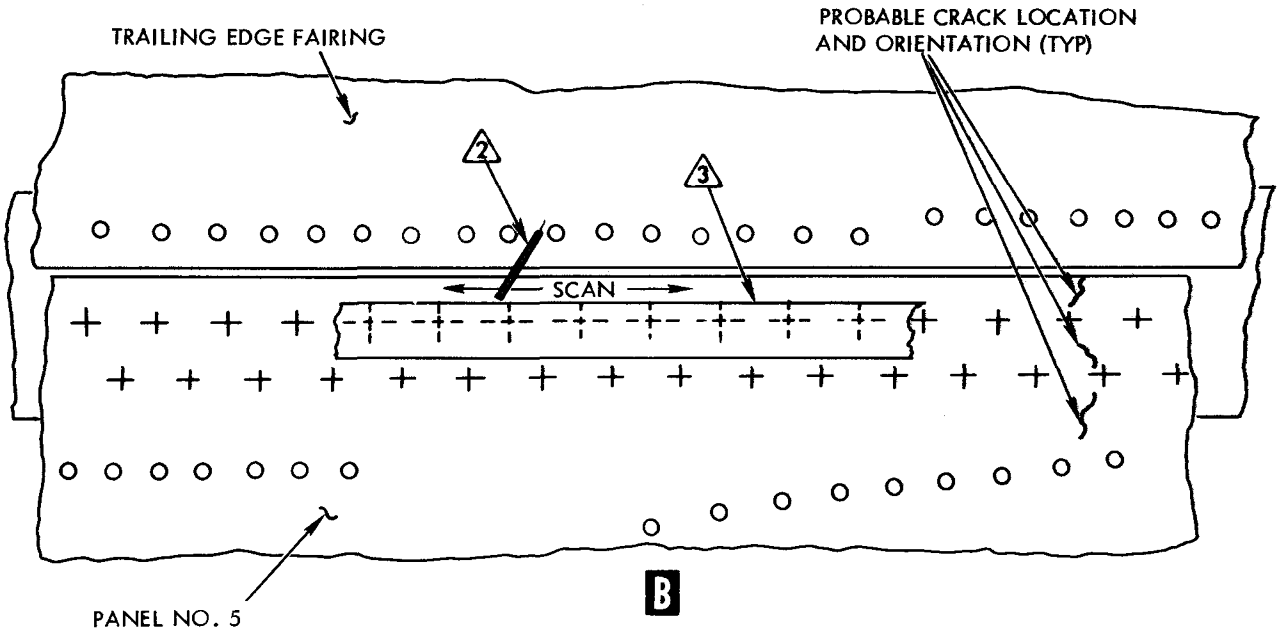
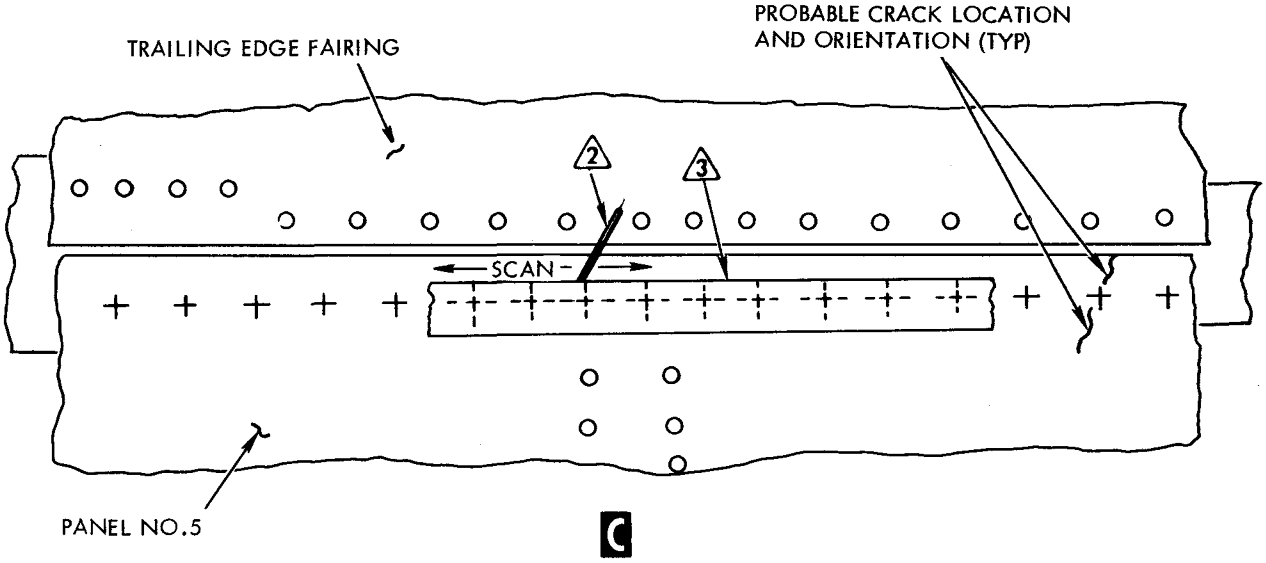


Figure 58.2-11-4/4. Wing Panel to Rear Beam Lower Cap Spanwise Splice, BBS 75 to BBS 148 (Sheet 1 of 4)



B

INSPECTION AREA
TYPICAL BBS 75 TO BBS 103



C

INSPECTION AREA
TYPICAL BBS 103 TO BBS 148

NOTE

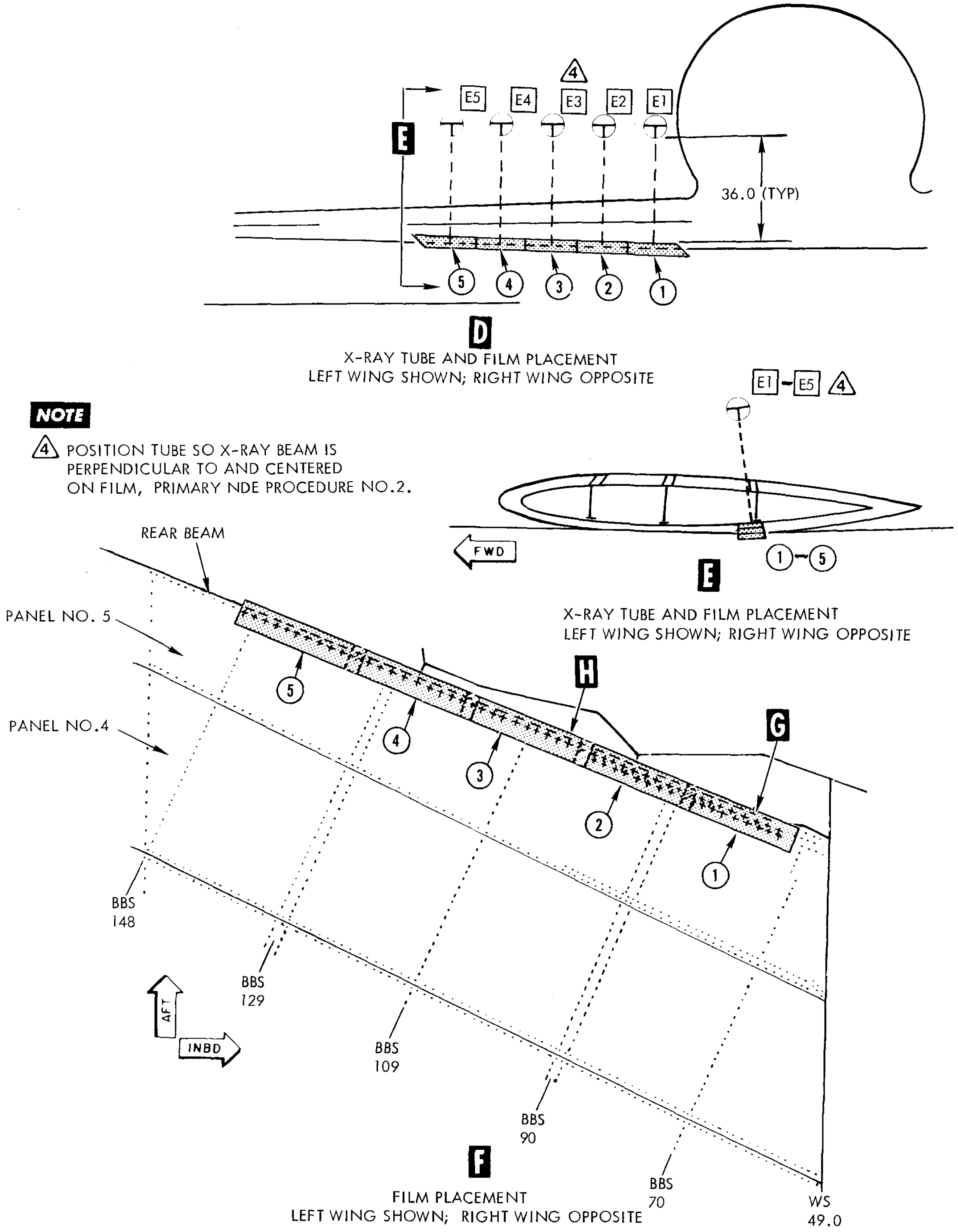
1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "O" FOR REFERENCE ONLY

△2 SHIELDED PENCIL PROBE-SCAN IN AN INBOARD AND OUTBOARD DIRECTION ON FORWARD AND AFT SIDES OF INDICATED FASTENERS, PRIMARY NDE PROCEDURE NO. 1.

△3 STRAIGHT EDGE SHOWN ALIGNED WITH AFT EDGE OF FASTENER ROW FOR SCANNING AFT SIDE OF FASTENERS. ALIGN STRAIGHT EDGE WITH FORWARD EDGE OF FASTENER ROW FOR SCANNING FORWARD SIDE OF FASTENERS.

SMR151-58.2-0-011-2

Figure 58.2-11-4/4. Wing Panel to Rear Beam Lower Cap Spanwise Splice, BBS 75 to BBS 148 (Sheet 2 of 4)



D

X-RAY TUBE AND FILM PLACEMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

NOTE

4 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND CENTERED ON FILM, PRIMARY NDE PROCEDURE NO.2.

E

X-RAY TUBE AND FILM PLACEMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

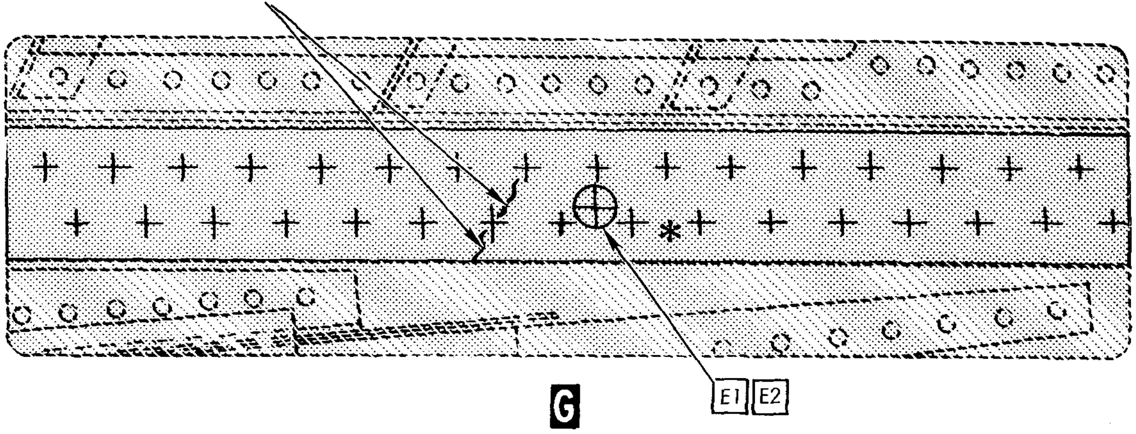
F

FILM PLACEMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

SMR 151-58.2-0-011 3

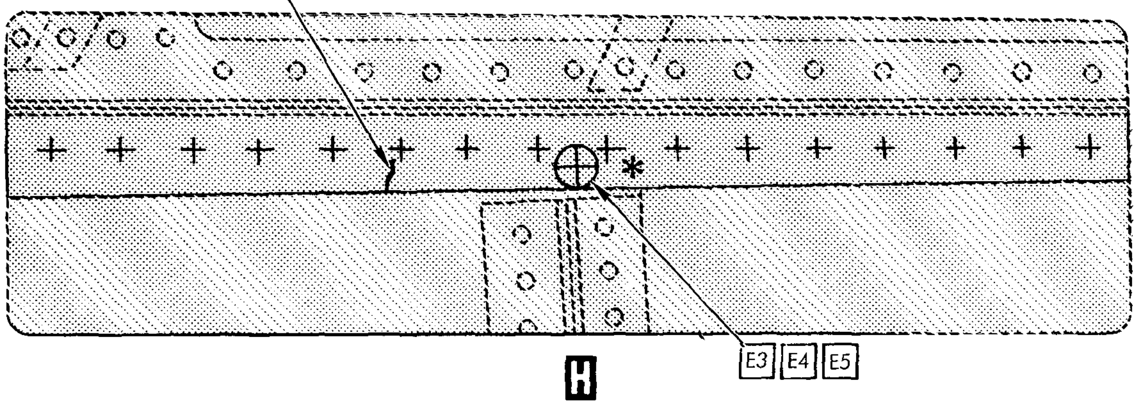
Figure 58.2-11-4/4. Wing Panel to Rear Beam Lower Cap Spanwise Splice, BBS 75 to BBS 148 (Sheet 3 of 4)

PROBABLE CRACK LOCATION AND ORIENTATION (TYP



FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINT FILM ① SHOWN; FILM ② SIMILAR

PROBABLE CRACK LOCATION AND ORIENTATION (TYP)



* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINT FILM ③ SHOWN; FILM ④ AND ⑤ SIMILAR

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	120	4	36	60	NDT45/ NDT55	4.5x17.0	FILM NUMBER SAME AS EXPOSURE NUMBER
E2							
E3	↕	↕	↕	↕	↕	↕	
E4							
E5	120	4	36	60	NDT45/ NDT55	4.5x17.0	

SMR-151-58.2-0-011-4

Figure 58.2-11-4/4. Wing Panel to Rear Beam Lower Cap Spanwise Splice, BBS 75 to BBS 148 (Sheet 4 of 4)

58.2.13 WING PANEL TO MID BEAM LOWER CAP SPANWISE SPLICE, BBS 55 TO BBS 187.

58.2.13.1 DESCRIPTION. (See figure 58.2-12-6/4.) The mid beam lower cap is attached on the interior surface of the lower wing panel No. 3 to No. 4 spanwise splice. The wing panels are machined and chemically milled from 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162), or 7075-T76511 (replacement panels) extrusions. The mid beam lower cap is machined from 7075-T6 aluminum extrusions. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.13.2 DEFECTS. Fatigue cracks may occur in the fastener holes common to the mid beam cap and the lower surface wing panels. Cracks will initiate in the fastener holes and propagate hole-to-hole or hole-to-edge of part in the chordwise direction.

58.2.13.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD
Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the upper and lower wing-to-fuselage fillet fairings from the wing front beam to the wing rear beam.

(3) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(4) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

WARNING

The wing must be supported in a neutral stress position prior to removal of the upper surface wing panels in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(5) Remove the integral fuel tank top access panels in accordance with Section 28.2 of this manual.

c. Access: Access to the inspection areas is gained with the external fuel tanks and integral fuel tank top access panels removed.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.13.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

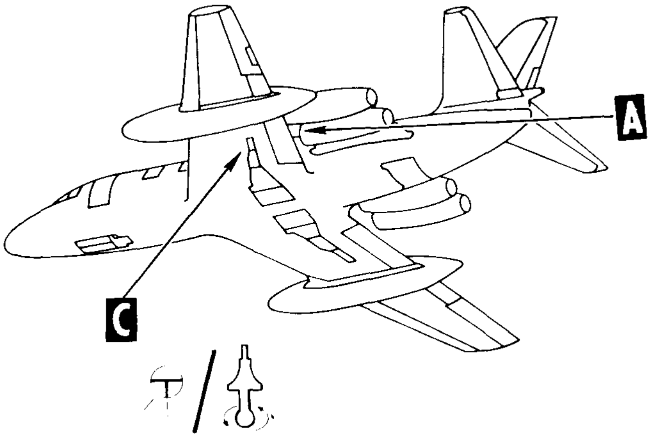
Note

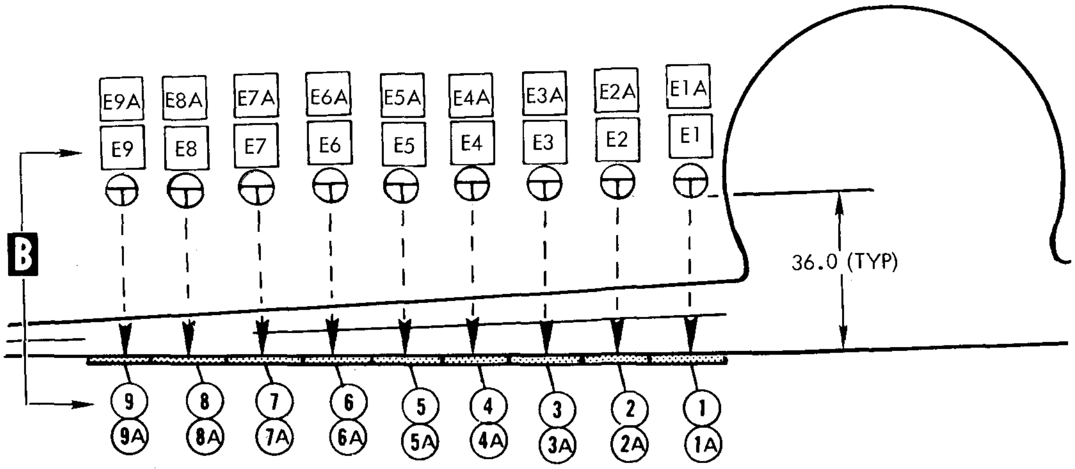
The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.2.13.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.





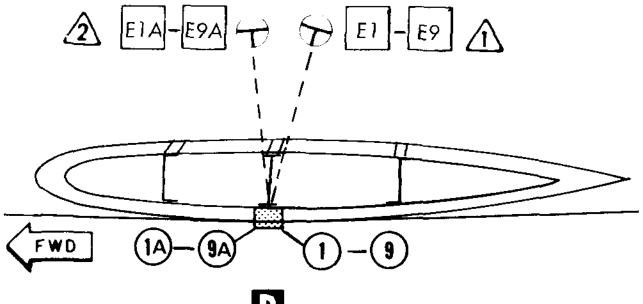
A

X-RAY TUBE AND FILM PLACEMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

NOTE

⚠1 POSITION TUBE SO X-RAY BEAM PASSES JUST AFT OF MID BEAM UPPER CAP AND PLUMBING AND IS CENTERED ON FILM.

⚠2 POSITION TUBE SO X-RAY BEAM PASSES JUST FORWARD OF MID BEAM UPPER CAP AND PLUMBING AND IS CENTERED ON FILM.

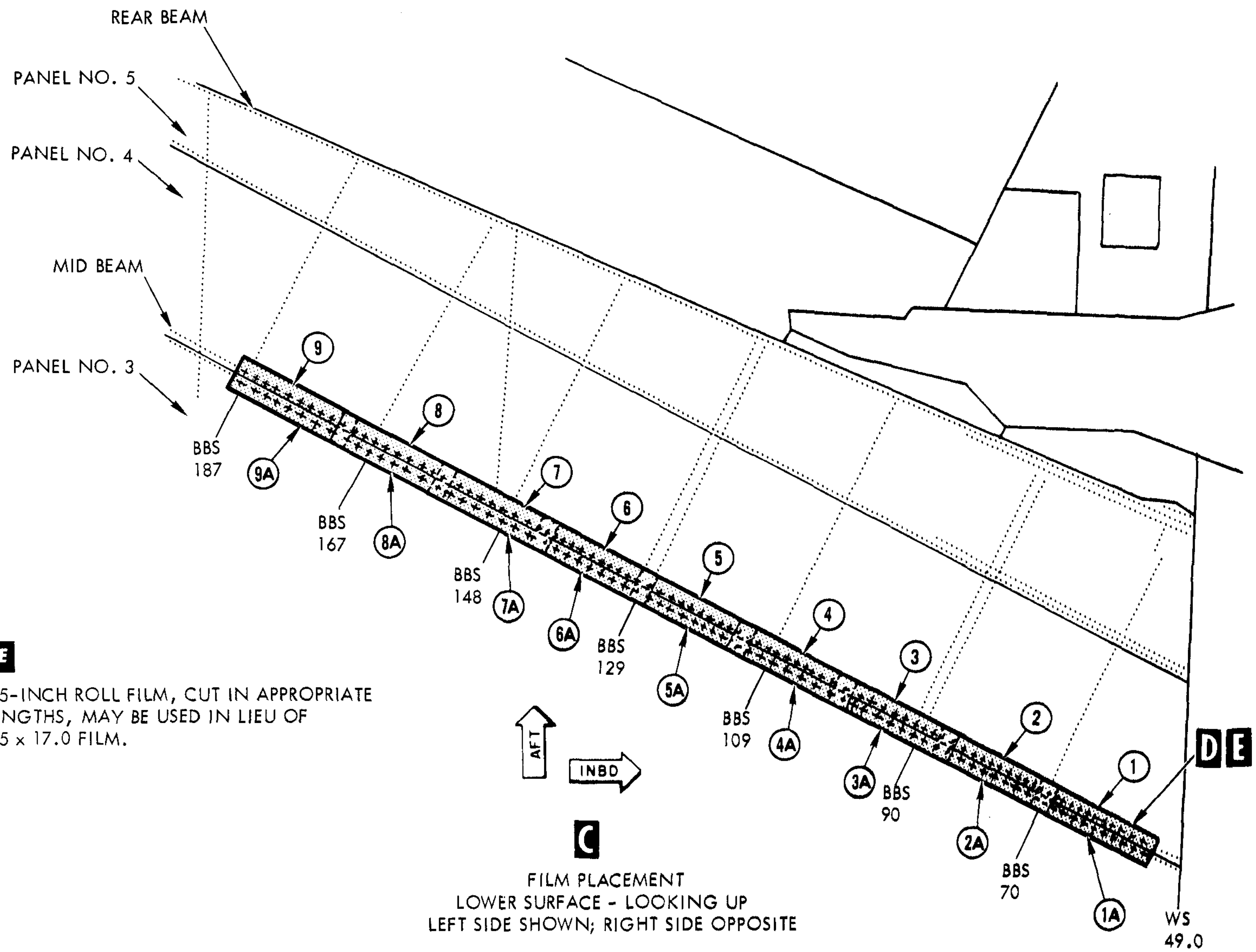


B

X-RAY TUBE AND FILM PLACEMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

SMR151-58.2-0-012-1

Figure 58.2-12-6/4. Wing Panel to Mid Beam Lower Cap Spanwise Splice, BBS 55 to BBS 187 (Sheet 1 of 3)



NOTE

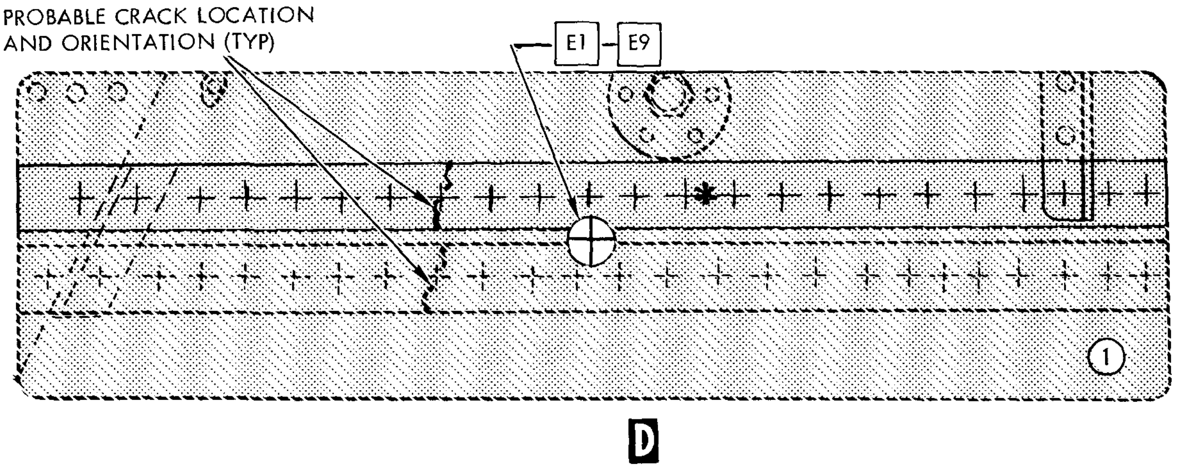
3. 4.5-INCH ROLL FILM, CUT IN APPROPRIATE LENGTHS, MAY BE USED IN LIEU OF 4.5 x 17.0 FILM.

C

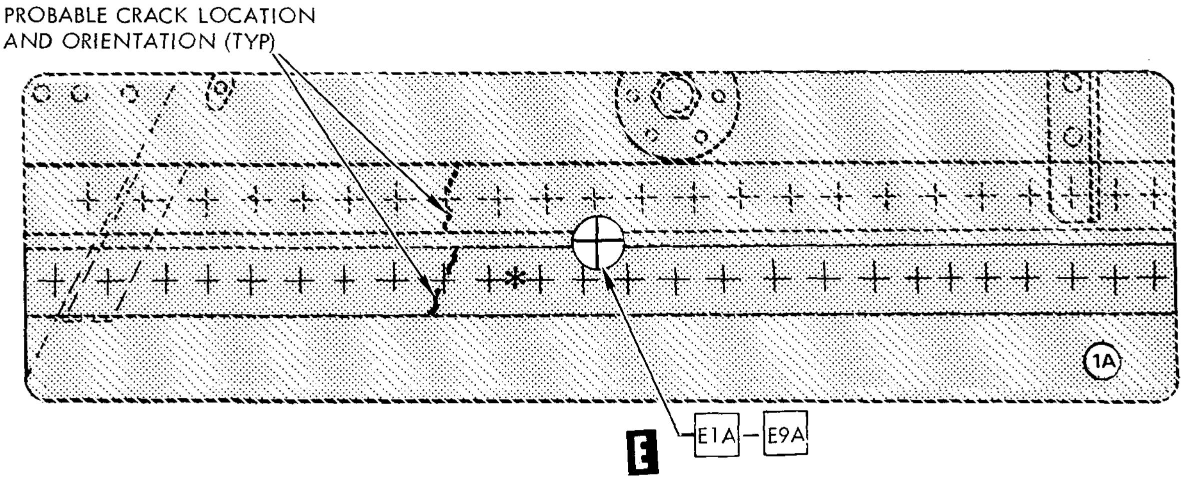
FILM PLACEMENT
LOWER SURFACE - LOOKING UP
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

SMR 151-58.2-0-012 2

Figure 58.2-12-6/4. Wing Panel to Mid Beam Lower Cap Spanwise Splice, BBS 55 to BBS 187
(Sheet 2 of 3)



FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINT - AFT SIDE OF MID BEAM LOWER CAP
FILM ① SHOWN; FILMS ② THRU ⑨ SIMILAR



FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINT - FWD SIDE OF MID BEAM LOWER CAP
FILM (1A) SHOWN; FILMS (2A) THRU (9A) SIMILAR

* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

RADIOGRAPHIC INSPECTION DATA

EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1 - E9	120	4	36	60	NDT 45/ NDT 55	4.5 x 17.0	FILM NUMBER SAME AS EXPOSURE NUMBER
E1A - E9A	120	4	36	60	NDT 45/ NDT 55	4.5 x 17.0	

SMR151-58.2-0-012-3

Figure 58.2-12-6/4. Wing Panel to Mid Beam Lower Cap Spanwise Splice, BBS 55 to BBS 187 (Sheet 3 of 3)

58.2.14 WING LOWER PANEL DRAIN HOLES, LEFT AND RIGHT.

58.2.14.1 DESCRIPTION. (See figure 58.2-13-4/2.) The extruded wing panels having integral "I" section risers form the lower surface of the wings. These panels are 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162) or 7075-T76511 (replacement panels), extrusions, and are machined and chemically milled to gradually reduce the cross sectional area as they extend outboard. The integral risers have drain holes at the base of the risers. Some of these holes have aluminum sleeves installed. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.14.2 DEFECTS. Stress corrosion and/or fatigue cracks may occur in the riser at the drain holes and propagate in any direction.

58.2.14.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM202RA or equivalent.

(3) Probe, Bolt Hole, 3/16-inch diameter VM Products P/N VM101B or equivalent.

(4) Probe, Bolt Hole, 1/4-inch diameter VM Products P/N VM101B or equivalent.

(5) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(3) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

WARNING

The wing must be supported in a neutral stress position prior to removal of the upper surface wing panels in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(4) Remove the upper surface wing panels (integral fuel tank top access panel) in accordance with Section 28.2 of this manual.

c. Access: The inspection area is accessible with the upper surface panels removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean local inspection areas around the drain holes as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for a calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using 90-degree probe, inspect the surface around the drain holes on both sides of the risers by scanning in a clockwise and counterclockwise direction. Where sleeves are installed, scan adjacent to the edge of the sleeve.

(2) Using 90-degree probe, scan the first drain hole outboard of the end of the angles on each riser and between BBS 129 and BBS 148 aft of the mid beam as illustrated. The list as illustrated is shown for reference; the drain hole locations may vary with different serialization.

(3) Using 90-degree probe, scan the first drain hole outboard and/or inboard adjacent to the end of any riser repair. When a repair covers a drain hole identified in the illustration, that drain hole will not require this inspection.

(4) Using appropriate size bolt-hole probe, scan the entire inner surface of drain holes in the front beam as illustrated.

(5) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the riser.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.14.4 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

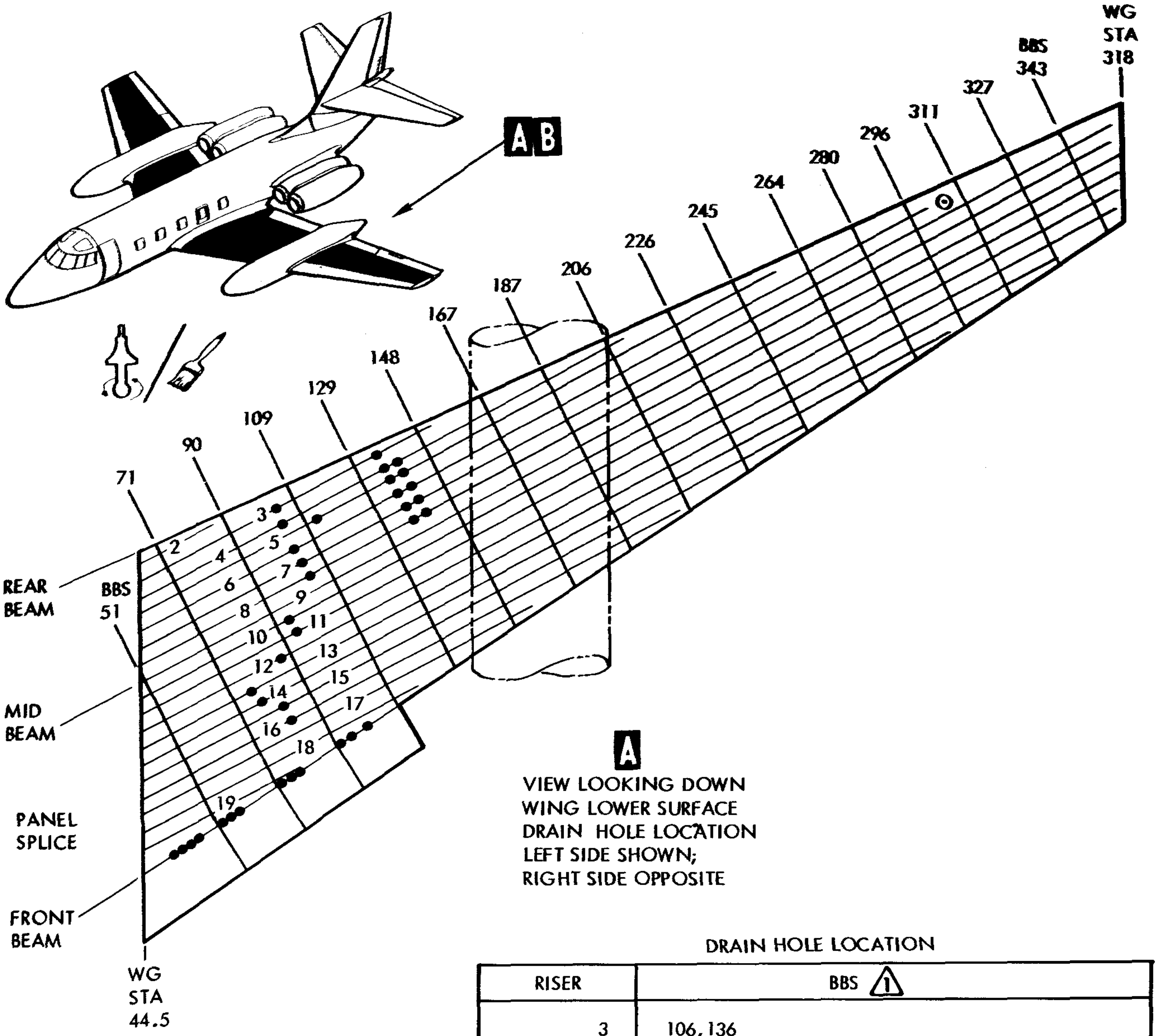
d. Preparation of part: Same as for Primary NDE Procedure except, strip finish and clean suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.2.14.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



VIEW LOOKING DOWN
WING LOWER SURFACE
DRAIN HOLE LOCATION
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

DRAIN HOLE LOCATION

RISER	BBS ⚠1
3	106, 136
4	104, 136, 140
5	113, 136, 140
6	104, 136, 140
7	105, 136, 140
8	105, 136, 140
10	94
11	94
12	88
13	77
14	79
15	83
16	83
FRONT BEAM	34, 36, 37, 40, 53, 54, 56, 72, 74, 75, 91, 93, 99

NOTE

⚠1 BBS MAY VARY SLIGHTLY DEPENDING ON AIRPLANE SERIALIZATION.

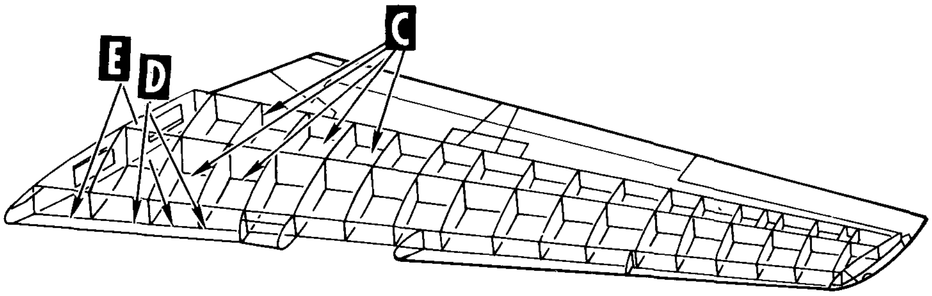
2. INSPECT RISERS AT INDICATED DRAIN HOLE LOCATIONS.

3. WHERE ANGLES ARE ATTACHED TO RISER, INSPECT AROUND FIRST DRAIN HOLE OUTBOARD OF ANGLES.

4. WHERE RISER REPAIRS ARE INSTALLED, INSPECT AROUND FIRST DRAIN HOLE OUTBOARD AND/OR INBOARD OF REPAIR.

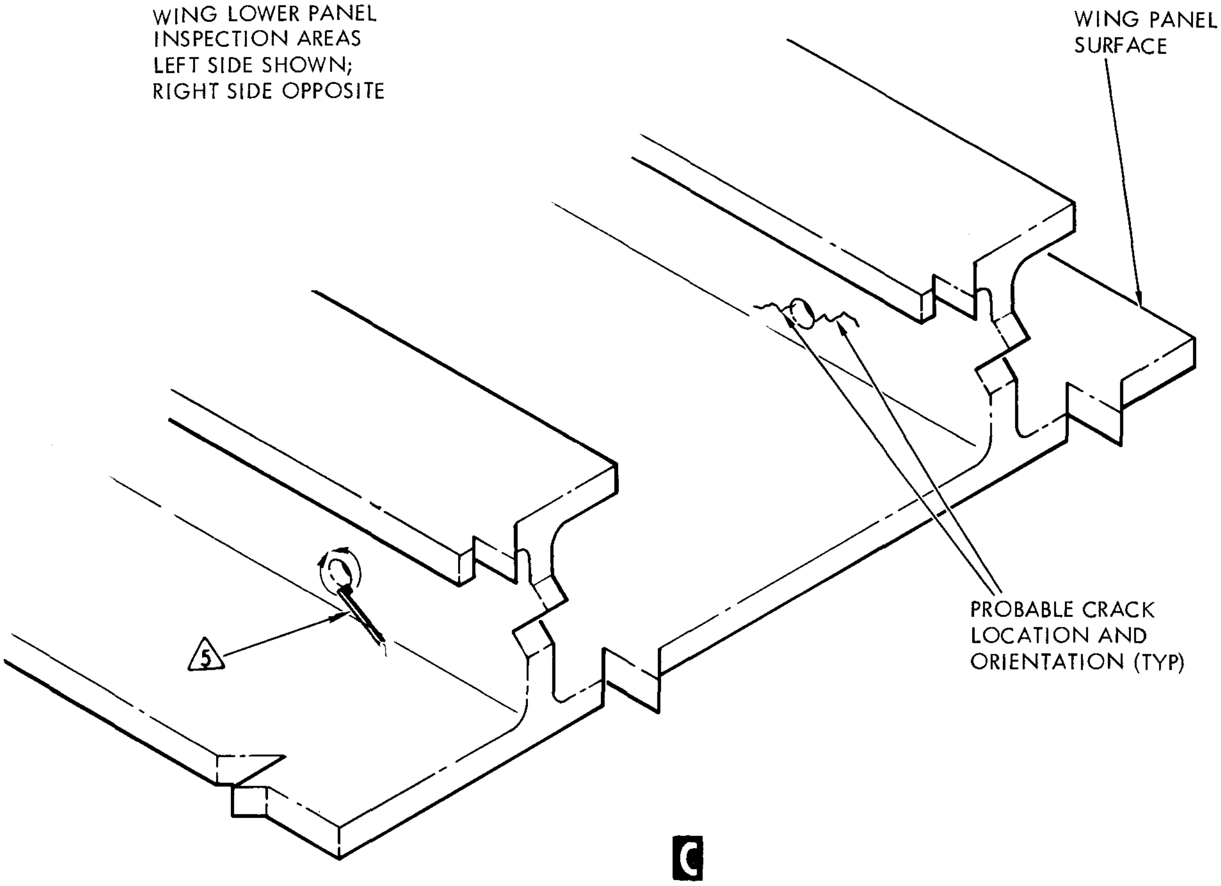
SMR151-58.2-0-013-1

Figure 58.2-13-4/2. Wing Lower Panel Drain Holes, Left and Right (Sheet 1 of 3)



B

WING LOWER PANEL
INSPECTION AREAS
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



C

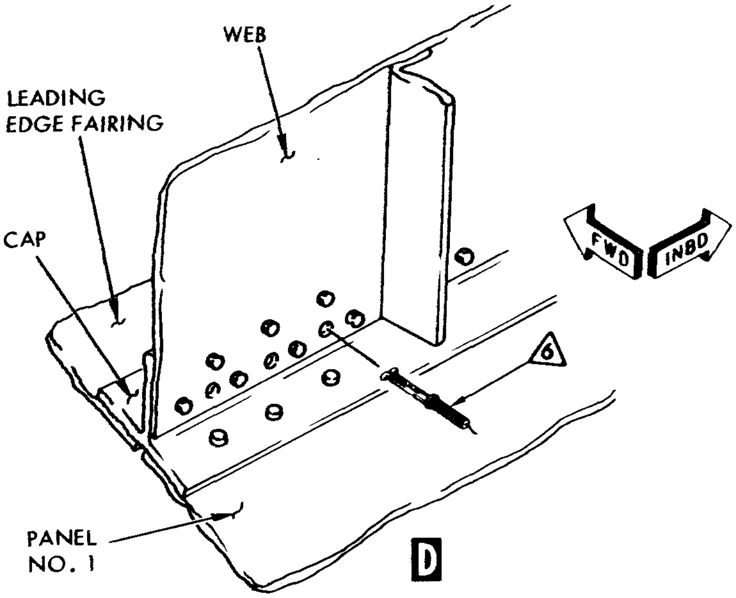
RISER DRAIN HOLE
INSPECTION (TYP)

NOTE

5 90-DEGREE PROBE - SCAN SURFACE AROUND DRAIN HOLE ON BOTH SIDES OF RISER IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION. WHERE SLEEVES ARE INSTALLED, SCAN ADJACENT TO EDGE OF SLEEVE.

SMR151-58.2-0-013-2

Figure 58.2-13-4/2. Wing Lower Panel Drain Holes, Left and Right (Sheet 2 of 3)

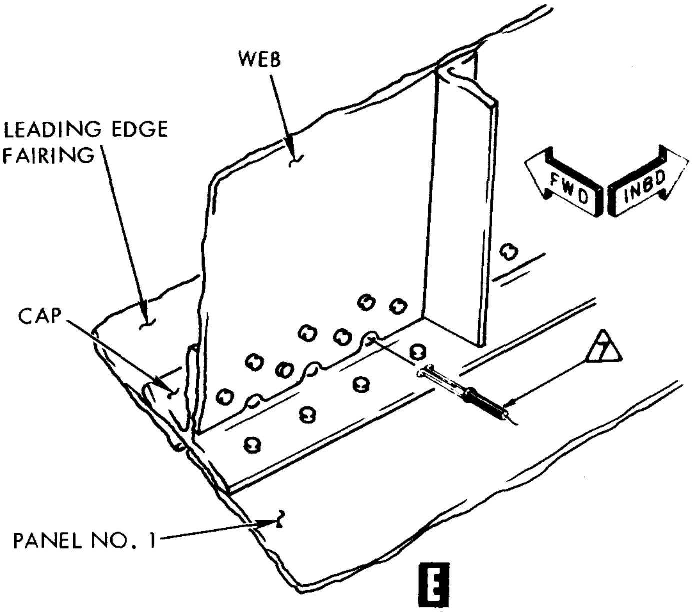


FRONT BEAM DRAIN HOLE
INSPECTION AREA
TYPICAL - BBS53, 54, 56, 91, 93, & 99

NOTE

6 3/16-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF DRAIN HOLES IN FRONT BEAM AT BBS 53, 54, 56, 91, 93, AND 99.

7 1/4-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF DRAIN HOLES IN FRONT BEAM AT BBS 34, 36, 37, 40, 72, 74, AND 75.



FRONT BEAM DRAIN HOLE
INSPECTION AREA
TYPICAL - BBS34, 36, 37, 40, 72, 74, & 75

SMR151-58.2-0-013-3

Figure 58.2-13-4/2. Wing Lower Panel Drain Holes, Left and Right (Sheet 3 of 3)

58.2.15 WING UPPER AND LOWER PANEL TO ROOT RIB ATTACHMENT.

58.2.15.1 DESCRIPTION. (See figure 58.2-14-6/4.) The wing panels are fastened together and attached to the outboard flange of the root rib assembly. The lower surface of the root rib assembly is one continuous fitting and the joint is stationary on the airplane. The upper surface of the root rib assembly is separated into the forward and aft segments which are removable for access to the wing box structure. External doublers are attached to the aft end of the upper and lower joints. The wing panels are machined and chemically milled from 7075-T6 (5001-5060, 5062-5066), 7075-T6511 (5061, 5067-5162), or 7075-T76511 (replacement panels) extrusions. The fittings and angles are machined from 7075-T6 aluminum extrusions and the doublers and straps are fabricated from 7075-T6 aluminum. See Section 51.8 for interior surface finish. The exterior surface finish will conform to customer peculiar paint scheme.

58.2.15.2 DEFECTS. Fatigue cracks may occur in the fastener holes in wing panels, fittings, angles, straps, and doublers outboard of the wing attachment locations at FS 410, 430, 450, 470 and 490. Cracks will initiate in the fastener hole and propagate hole-to-hole or hole-to-edge of part in the chordwise direction.

58.2.15.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD
Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the upper and lower wing-to-fuselage fillet fairings from the wing front beam to the wing rear beam.

(3) Remove the external fuel tanks in accordance with Section 28.2 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(4) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

WARNING

The wing must be supported in a neutral stress position prior to removal of the upper surface wing panels in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(5) Remove the upper surface wing panels (integral fuel tank top access panel) in accordance with Section 28.2 of this manual.

c. Access: Access to the inspection area is gained with the upper surface panel removed.

d. Preparation of part: Position the removed wing panels on mats or on supports for easy access to position film. No special preparation is required for the lower surface panels.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.2.15.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 1/4-inch diameter, VM Products P/N VM101B or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

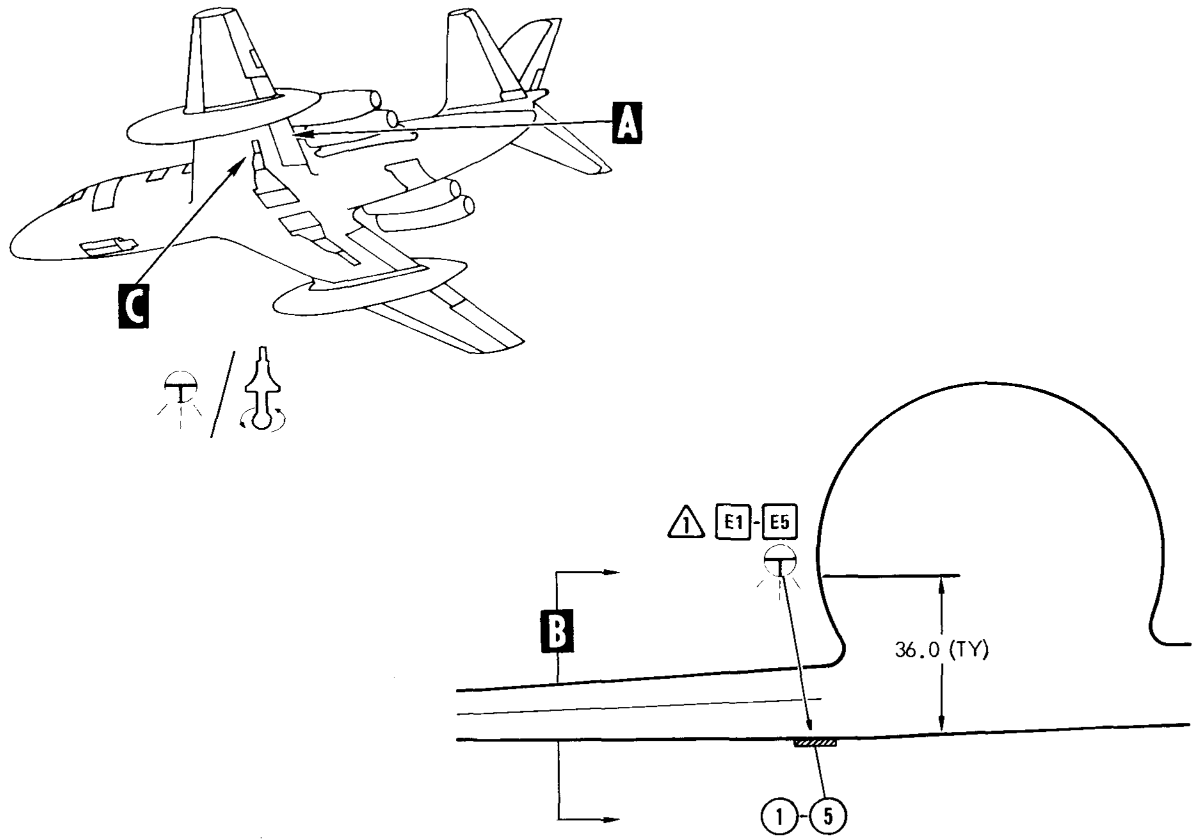
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using the appropriate size probe, scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

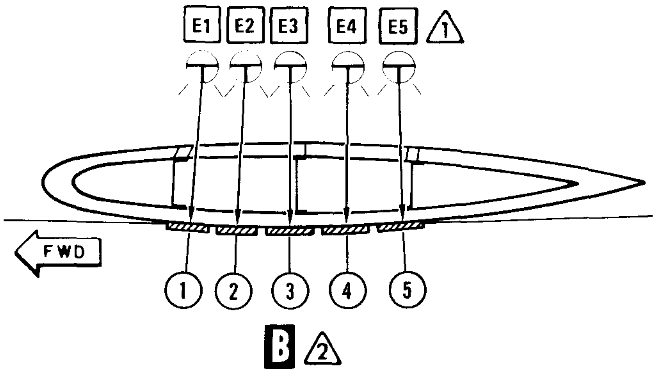
g. Mark and report indicated defects.

58.2.15.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



A △2

X-RAY TUBE AND FILM PLACEMENT
LOWER SURFACE ROOT RIB ATTACHMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE



B △2

X-RAY TUBE AND FILM PLACEMENT
LOWER SURFACE ROOT RIB ATTACHMENT
LEFT WING SHOWN; RIGHT WING OPPOSITE

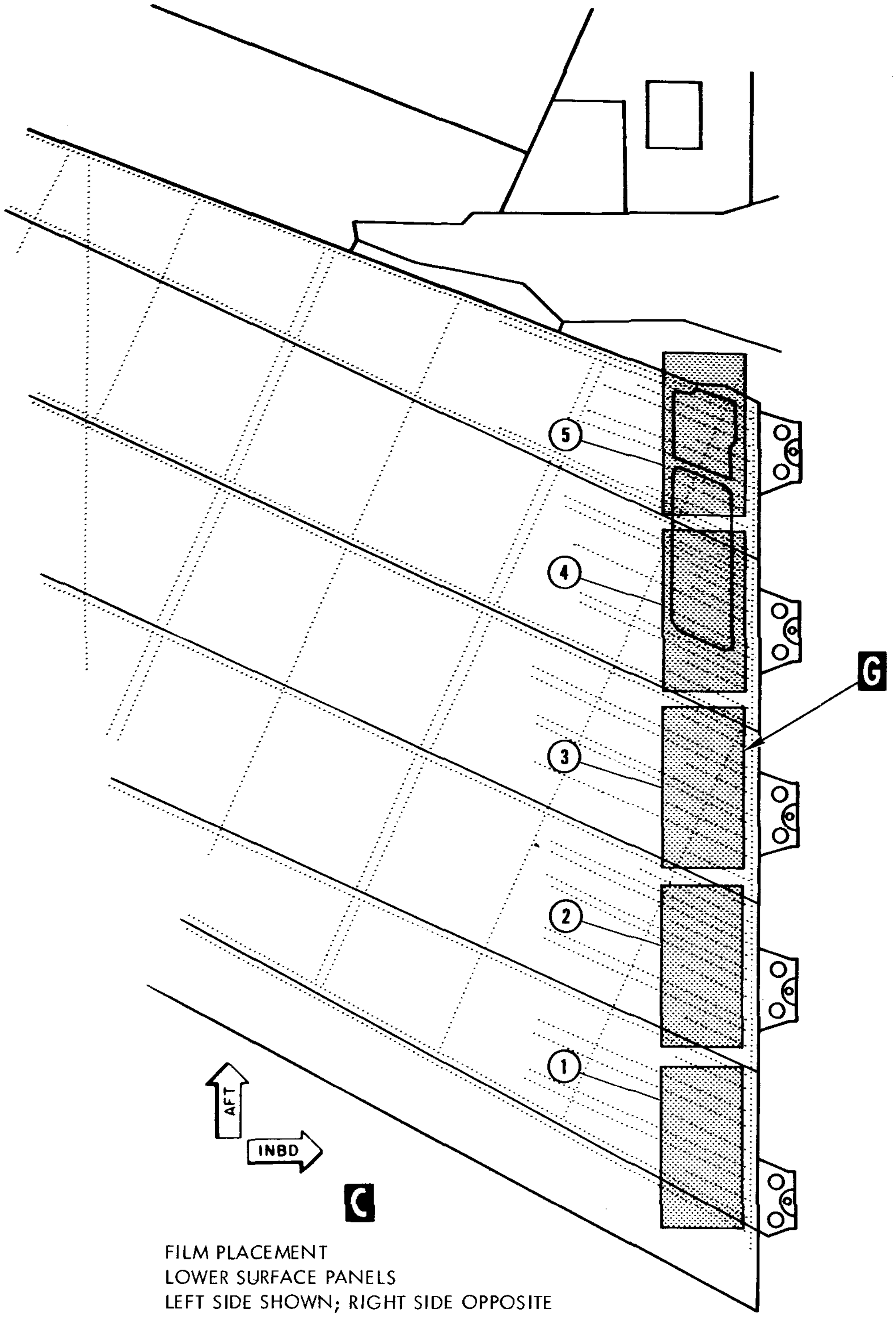
NOTE

△1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND CENTERED ON FILM.

△2 UPPER WING PANELS REMOVED; SEE VIEWS **D** AND **E**.

SMR 151-58.2-0-014-1

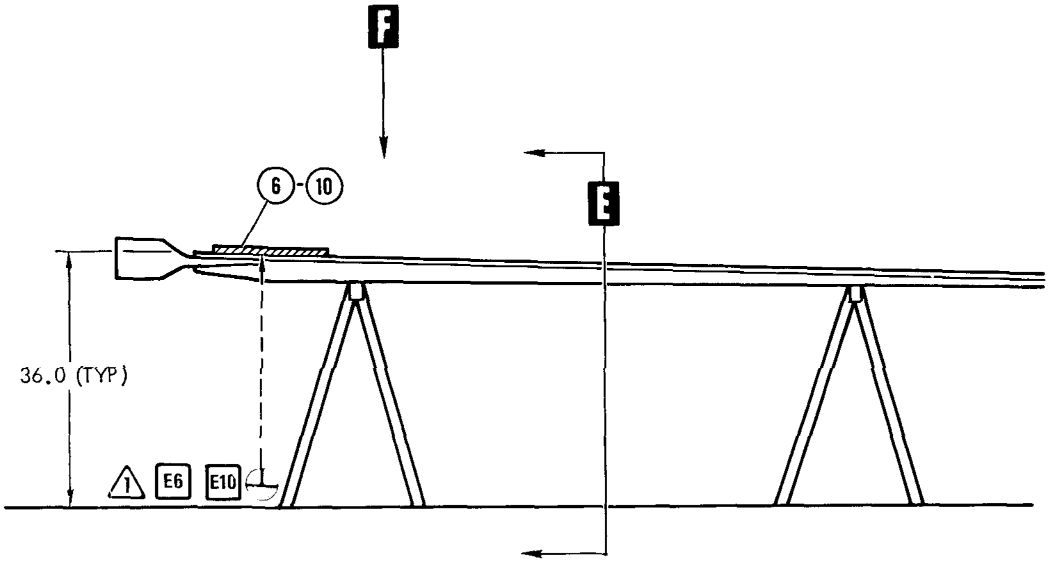
Figure 58.2-14-6/4. Wing Upper and Lower Panel to Root Rib Attachment (Sheet 1 of 5)



FILM PLACEMENT
LOWER SURFACE PANELS
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

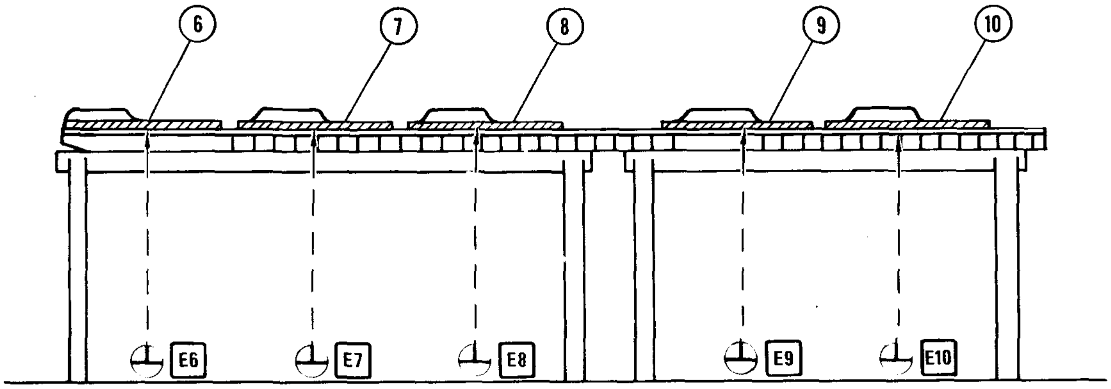
SMR 151-58.2-0-014-2

Figure 58.2-14-6/4. Wing Upper and Lower Panel to Root Rib Attachment (Sheet 2 of 5)



D

X-RAY TUBE AND FILM PLACEMENT
UPPER SURFACE ROOT RIB ATTACHMENT
UPPER PANELS ON SUPPORTS



E

X-RAY TUBE AND FILM PLACEMENT
UPPER SURFACE ROOT RIB ATTACHMENT
UPPER PANELS ON SUPPORTS

NOTE

⚠1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND CENTERED ON FILM.

SMR 151-58.2-0-014-3

Figure 58.2-14-6/4. Wing Upper and Lower Panel to Root Rib Attachment (Sheet 3 of 5)

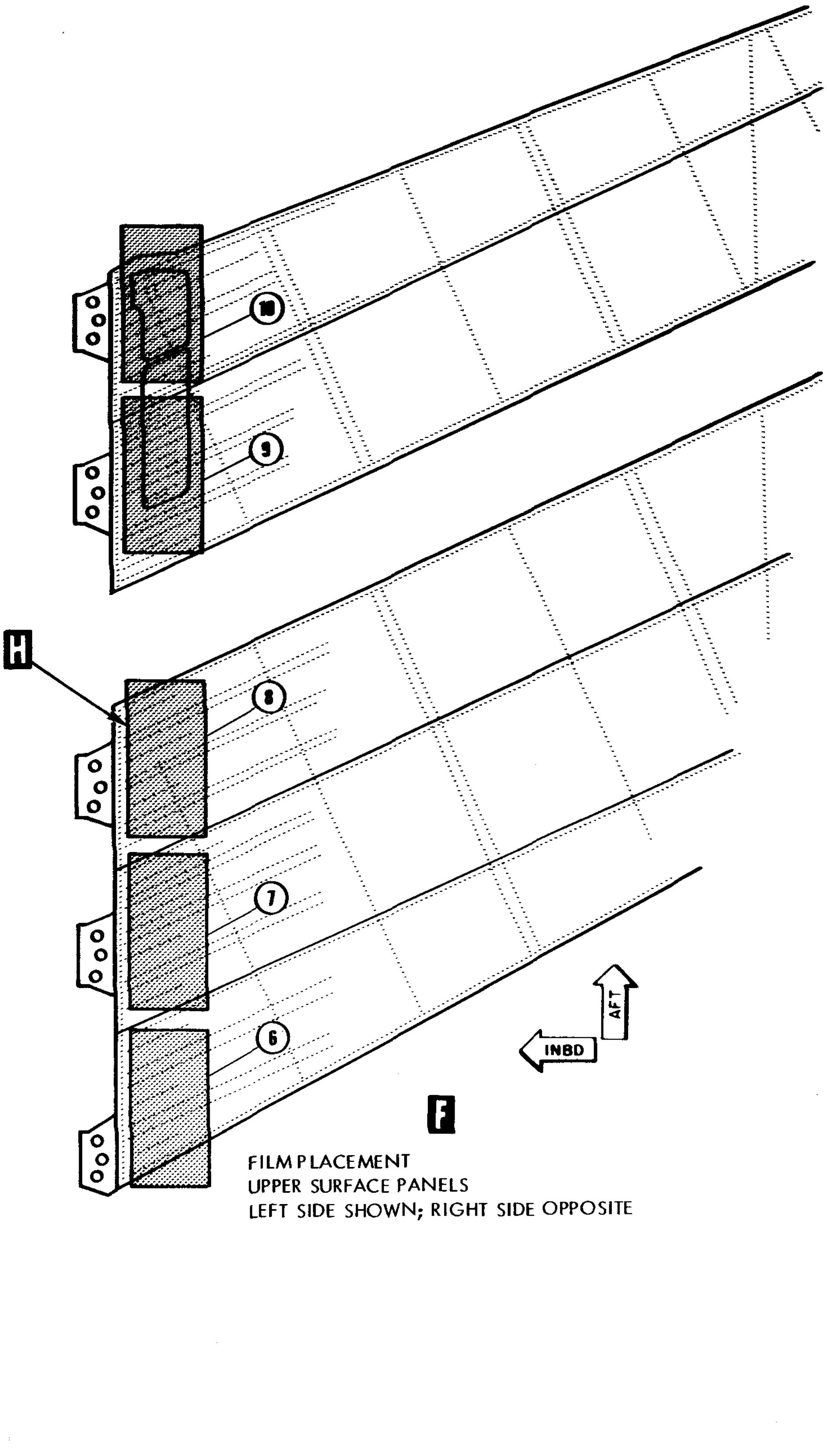
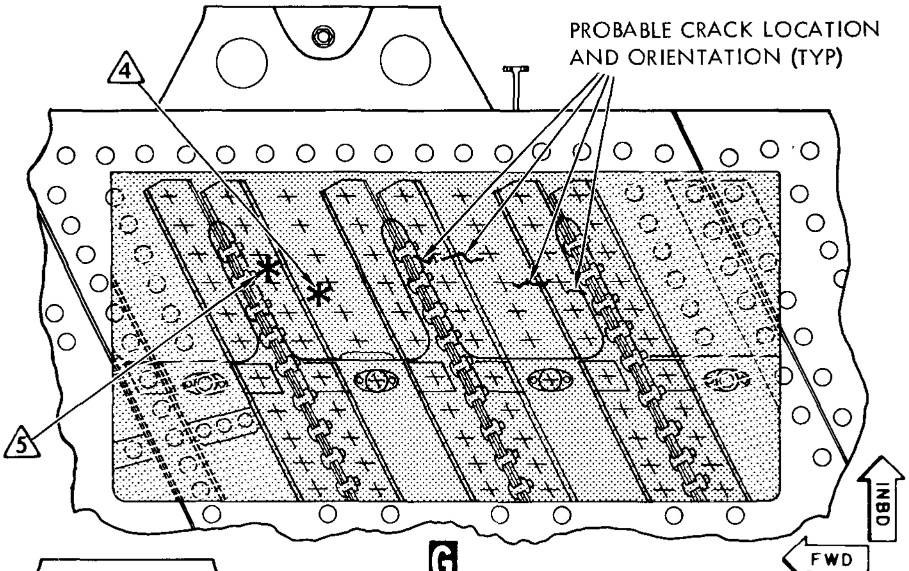
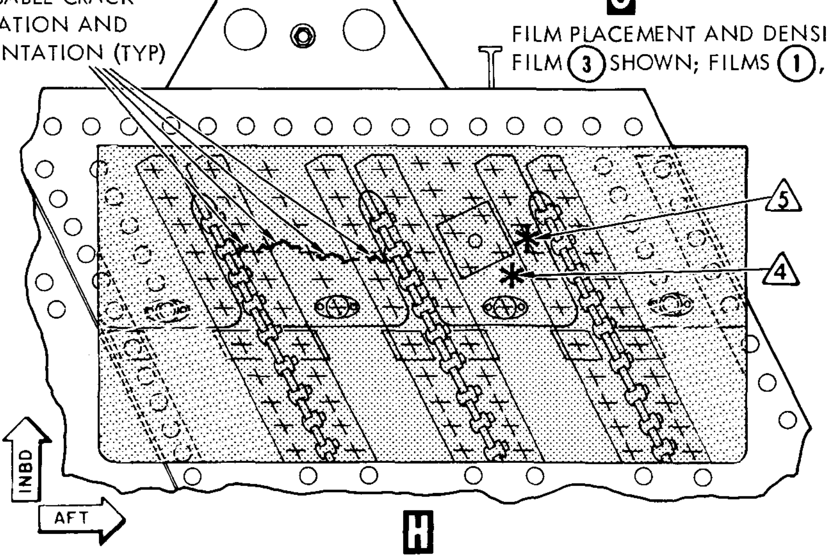


Figure 58.2-14-6/4. Wing Upper and Lower Panel to Root Rib Attachment (Sheet 4 of 5)



G

FILM PLACEMENT AND DENSITY CHECK POINTS
FILM ③ SHOWN; FILMS ①, ②, ④, AND ⑤ SIMILAR



H

FILM PLACEMENT AND DENSITY CHECK POINTS
FILM ⑧ SHOWN; FILMS ⑥, ⑦, ⑨, AND ⑩ SIMILAR

NOTE

3. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODE "0" FOR REFERENCE ONLY.

△4 CHECK DENSITY ON NDT 45 FILM.

△5 CHECK DENSITY ON NDT 55 FILM.

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1 - E5	130	4	36.0	60	NDT 45 NDT 55	8.5 x 17.0	FILM NUMBER SAME AS EXPOSURE NUMBER
E6 - E10	130	4	36.0	60	NDT 45 NDT 55	8.5 x 17.0	

SMR 151-58.2-0-014-5

Figure 58.2-14-6/4. Wing Upper and Lower Panel to Root Rib Attachment (Sheet 5 of 5)

58.2.16 WING UPPER AND LOWER ROOT RIB FITTING ATTACHMENT.

58.2.16.1 DESCRIPTION. (See figure 58.2-15-4/0.) The wing panels are attached to the fuselage by wing joint fittings secured to the box structure at the wing root rib. The root fittings are attached by tension bolts to the fuselage frames and root rib shear fittings. This joint is at butt line 41.5 with the upper and lower tension bolts at fuselage stations 410, 430, 450, 470 and 490. The root rib fittings are machined from 7075-T6 aluminum extrusions and the shear fittings are machined from 7075-T6 aluminum forgings. All fittings are finished with zinc chromate primer.

58.2.16.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the root rib fittings, root rib shear fittings and bolts. In both upper and lower joints, cracks will initiate in the attachment holes and propagate in a random direction. Cracks in the bolts will propagate in a circumferential and longitudinal direction.

Note

Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.2.16.3 PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Bolt Hole, 1/4-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Shielded, Bolt Hole, 5/16-inch diameter, VM Products P/N VM101B or equivalent.

(4) Probe, Shielded, Bolt Hole, 3/8-inch diameter, VM Products P/N VM101B or equivalent.

(5) Probe, Shielded, Bolt Hole, 7/16-inch diameter, VM Products P/N VM101B or equivalent.

(6) Probe, Shielded, Bolt Hole, 5/8-inch diameter, VM Products P/N VM101B or equivalent.

(7) Calibration Standard, aluminum (See figure 58.1-9)

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane:

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the upper and lower wing-to-fuselage fillet fairings from the wing front beam to the wing rear beam.

(3) Remove belly fairings to provide access to the lower wing to fuselage attach area at fuselage stations 410, 430, 450, 470 and 490. Refer to Section 52.3 of this manual.

WARNING

Ensure that the wing fuel tanks have been drained and purged to 1.5 percent or less of the lower explosive limit. Failure to comply may result in serious injury to personnel.

(4) Defuel, drain and purge the wing fuel tanks to 1.5 percent or less of the lower explosive limit in accordance with Section 28.2 of this manual.

CAUTION

The wing must be supported in a neutral stress position prior to removal of the fasteners in order to maintain the integrity of the wing and to preclude movement between mating surfaces.

(5) Jack and shore the airplane to provide support of the wing in the neutral position per instructions in Sections 7.0 and 7.1 of this manual.

(6) Remove the fasteners to be inspected from the inboard end of the integral fuel tank top access panels in accordance with Section 28.2 of this manual.

c. Access: The fastener holes are accessible with the fasteners removed.

d. Preparation of part:

(1) Remove the upper and lower tension bolt at fuselage stations 410, 430, 450, 470 and 490.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface areas as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

Note

The lower extremity of the fastener holes in the upper root rib fittings are curved at the barrel nut cavity. The inboard and outboard surfaces extend lower than the fore and aft surfaces. A downward deflection of the instrument needle will occur when scanning at the fore and aft quadrants of the hole. This will not indicate a defect.

f. Inspection: Using the appropriate bolt hole probe, scan the entire inner surface of each fastener hole. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

h. If no defect is indicated, proceed with Primary NDE Procedure No. 2.

58.2.16.4 PRIMARY NDE PROCEDURE NO. 2 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14A Magnaflo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The tension bolts are accessible when removed from the fittings.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the tension bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

MS 20004-12	250 amperes
MS 20005-10	350 amperes
MS 20005-14	350 amperes
MS 20006-16	400 amperes
MS 20007-14	450 amperes
MS 20010-22	650 amperes
MS 20010-24	650 amperes

(2) Longitudinal field:

MS 20004-12	1800 amperes
MS 20005-10	2400 amperes
MS 20005-14	2000 amperes
MS 20006-16	2000 amperes
MS 20007-14	2400 amperes
MS 20010-22	2500 amperes
MS 20010-24	2300 amperes

f. Inspection: Inspect the tension bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetc fields and demagnetization.

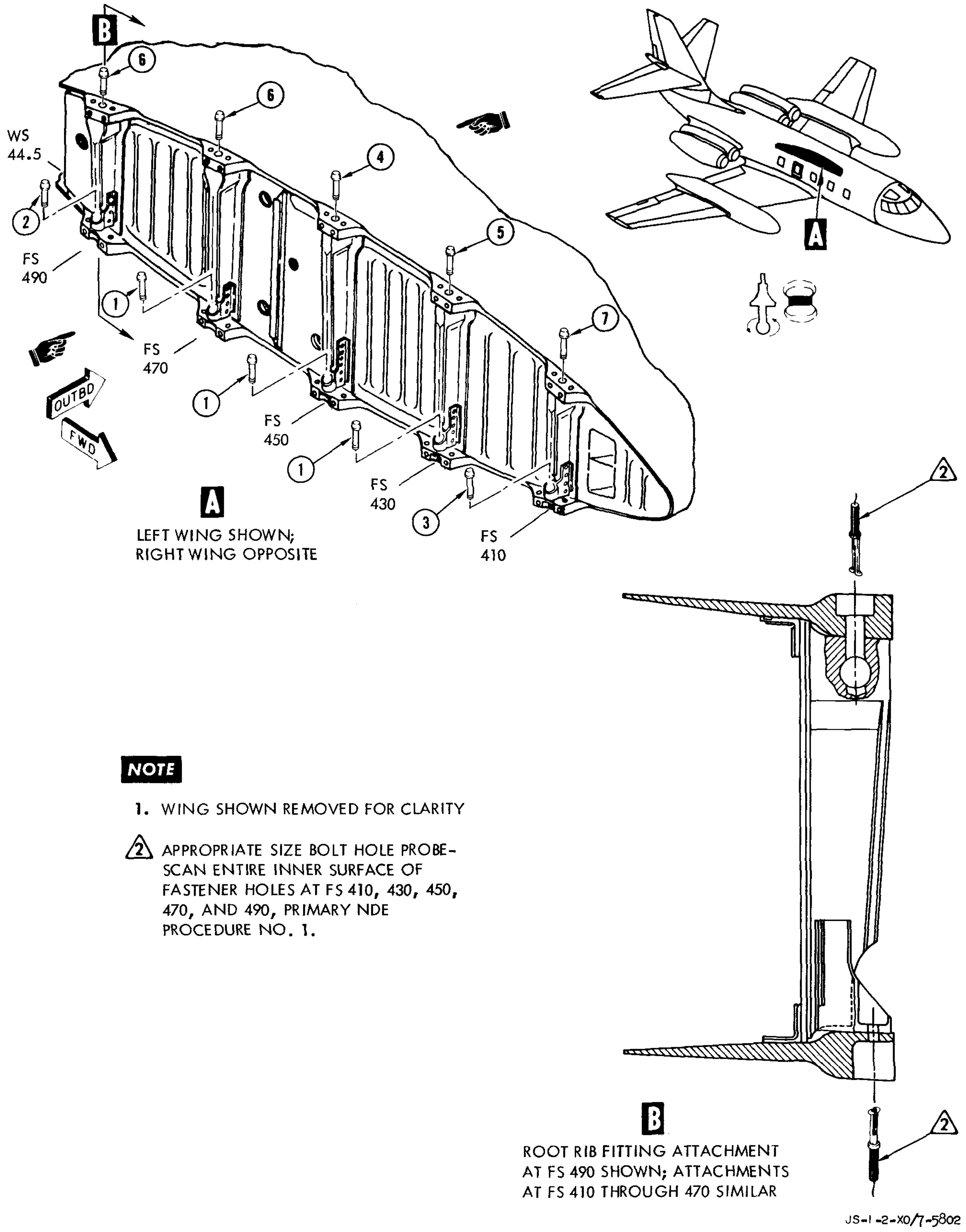
g. Mark and report indicated defects.

58.2.16.5 BACKUP NDE PROCEDURE.

None required.

58.2.16.6 SYSTEM SECURING.

Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

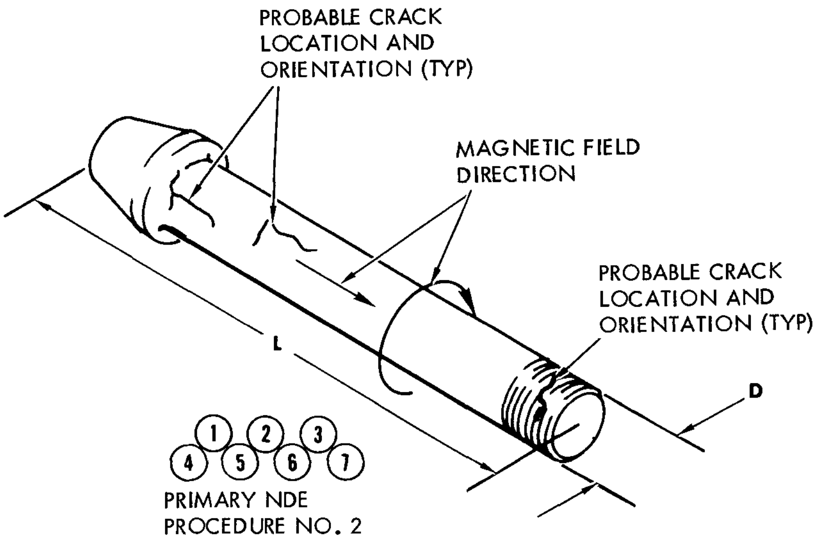


NOTE

1. WING SHOWN REMOVED FOR CLARITY

△2 APPROPRIATE SIZE BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF FASTENER HOLES AT FS 410, 430, 450, 470, AND 490, PRIMARY NDE PROCEDURE NO. 1.

Figure 58.2-15-4/0. Wing Upper and Lower Root Rib Fitting Attachment (Sheet 1 of 2)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
① ② ③ ④ ⑤ ⑥ ⑦
PRIMARY NDE PROCEDURE NO. 2

BOLT	D(IN)	L(IN)	MATERIAL	FINISH
① MS 20004-12	0.25	1.50	△3	△4
② MS 20005-10	0.31	1.50	△3	△4
③ MS 20005-14	0.31	1.75	△3	△4
④ MS 20006-16	0.37	2.06	△3	△4
⑤ MS 20007-14	0.43	2.13	△3	△4
⑥ MS 20010-22	0.63	2.94	△3	△4
⑦ MS 20010-24	0.63	3.06	△3	△4

NOTE

△3 ALLOY STEEL, INTERNAL WRENCHING

△4 CADMIUM PLATED

SMR 151-58.2-0-015-2

Figure 58.2-15-4/0. Wing Upper and Lower Root Rib Fitting Attachment (Sheet 2 of 2)

SECTION 58.3

EMPENNAGE GROUP

Contents Page

GENERAL DESCRIPTION 58-95

Inspection Procedures 58-95

58.3.1 GENERAL DESCRIPTION.

The empennage group consists of the vertical stabilizer, rudder, rudder tab, horizontal stabilizer, elevator, stabilizer tips, fairings, and fuel jettison nozzle. See Chapter 55 for complete description of the empennage structure and detail components.

58.3.1.1 INSPECTION PROCEDURES. Empennage Group areas and components to be inspected are tabulated in the Empennage Group Inspection Index (See Table 58.3-1). Each inspection procedure has been assigned an inspection index number which is cross referenced to the relevant paragraph and page number.

Table 58.3-1. Empennage Group Inspection Index

INDEX NO.	INSPECTION TITLE	PARAGRAPH NO.	PAGE NO.
E-1	Empennage Scissors Fittings, Attachment Fittings, and Bolts	58.3.2	58-96
E-2	Empennage Pitch-Trim Actuator Attachment Fittings and Bolts	58.3.3	58-108
E-3	Empennage Pivot Fittings, Bushings, and Hinge Pins	58.3.4	58-114
E-4	Horizontal Stabilizer Center Section Plates	58.3.5	58-120
E-5	Rudder Upper Counterweight Region	58.3.6	58-125
E-6	Vertical Stabilizer Actuator Attach Fitting Attachment	58.3.7	58-128
E-7	Horizontal Stabilizer Center Section Aft Lower Attachment	58.3.8	58-128B
E-8	Empennage Pivot Fitting to Vertical Stabilizer Rear Beam Attachment	58.3.9	58-128D
E-9	Empennage Pivot Fitting	58.3.10	58-128F

58.3.2 EMPENNAGE SCISSORS FITTINGS, ATTACHMENT FITTINGS, AND BOLTS.

58.3.2.1 DESCRIPTION. (See figure 58.3-1-5/2). The empennage scissors fitting provides attachment of the forward end of the vertical stabilizer to the fuselage and permits rotation of the empennage for aircraft trim. The fitting consists of the empennage scissors fitting and the fuselage scissors fitting which are linked together. The forward end of the fitting is linked to fittings attached to the fuselage and the aft end of the fitting is linked to a fitting attached to the vertical stabilizer front beam. The scissors fittings are fabricated from 7075-T6 or 7075-T73 aluminum forgings, the fitting attached to the vertical stabilizer is fabricated from 7075-T6 aluminum forging, and the fittings attached to the fuselage are fabricated from 2014-T6 aluminum forgings. All the fittings are finished with zinc chromate primer. Nitralloy 135 bushings are installed in the clevis/lug ends of each fitting and the fittings are linked together by nitralloy 135 bolts and pin. The inside surface of the bushings and grip length of the bolts and pin are treated with nitride, while the remaining surface of the bushings, bolts, and pin are cadmium plated.

58.3.2.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the scissors fittings, attach fittings, bushings, bolts, and pin. Cracks will initiate in the bushed holes and propagate in a direction perpendicular to the length of the scissors fittings and in a fore and aft direction in the attach fittings. Cracks in the bushings, bolts, and pin will propagate in a circumferential and longitudinal direction.

Note

Primary NDE Procedure No. 1, No. 2 and No. 3 must be performed to complete this inspection.

58.3.2.3 PRIMARY NDE PROCEDURE NO. 1 ULTRASONIC.

a. NDE equipment:

(1) Ultrasonic Flaw Detector, Sonic Model FTS Mark I, or equivalent.

(2) Transducer, 5.0 MHz, 0.25-inch diameter, longitudinal wave, Krautkramer-Branson P/N Z-103-ABFR or equivalent.

(3) Wedges, Lockheed P/N's PDL 949-1, PDL 949-2, PDL 950, PDL 951, PDL 952-1, and PDL 952-2 as illustrated.

(4) Calibration Standards, Lockheed P/N's PDL 935-1, PDL 935-2, PDL 937, PDL 938, and PDL 939 as illustrated.

(5) Cable, 6-foot, Microdot/BNC connector, Automation Industries P/N 57A2271 or equivalent.

(6) Couplant, light grease or oil.

b. Preparation of airplane:

WARNING

Ensure power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the vertical stabilizer lower leading edge fairing, refer to Section 52.3 of this manual for access panel locations.

(3) Hoist the empennage to a no-load condition per instructions in Section 7.0 of this manual.

(4) Remove the scissors fitting from the airplane and separate the center joint linkage.

c. Access: The lug and clevis ends are accessible with the scissors fitting removed from the airplane. For those lugs still on the airplane, use maintenance stands or appropriate AGE for access.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Remove excess sealant from the surface and clean local inspection areas as required to permit good contact between part and wedge.

e. Instrument settings/calibration:

(1) Select the appropriate wedge and calibration standard for the area to be inspected from the table contained in figure 58.0-4.

(2) Apply couplant and couple transducer wedge assembly to calibration standard in Position No. 1 as illustrated.

(3) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT display. Adjust sweep to display signal from the sawcut near the right side of the CRT display.

(4) Maximize the signal from the sawcut by adjusting the wedge position and adjust the signal amplitude to obtain 80 percent scope saturation on the CRT display.

(5) Scan to Position No. 2 as illustrated, and note the loss of sawcut signal on the CRT display.

(6) The CRT signal from the sawcut simulates a signal from a crack at the fastener hole.

58.3.2.3 (Cont).

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 of this manual.

Note

A signal from a lubrication fitting hole may appear in the same location as a crack signal on the CRT display. Care should be taken to ensure that signal is not called a crack indication.

f. Inspection:

(1) Apply couplant and couple appropriate wedge assembly to inspection area as illustrated.

(2) Scan the accessible outer surface on the clevis/lug in both a clockwise and counterclockwise direction.

(3) Any signal received similar to and at approximately the same location on the CRT display as the sawcut signals observed during calibration will indicate a probable crack in the part.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.3.2.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM102AS or equivalent.

(3) Probe, Bolt Hole, Shielded, 1/2-inch diameter, VM Products P/N VM101BS 1/2 or equivalent.

(4) Probe, Bolt Hole, Shielded, 3/4-inch diameter, VM Products P/N VM101BS 3/4 or equivalent.

(5) Calibration Standard, Aluminum (See figure 58.1-9).

(6) Calibration Standard, Steel (See figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The clevis ends and bushing are accessible with the scissors removed and disassembled.

d. Preparation of part:

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

(1) Clevis/lug ends - Clean the surface area around the bushing as required to permit good contact between part and probe. Refer to Section 58.0 for cleaning instructions.

CAUTION

The bushings have lubrication holes in the side wall which will become contaminated if cleaners or solvents are used to clean the bushing.

(2) Bushing - Clean the inner surface of the bushing using clean dry rags as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Clevis/lug ends - Using the right angle probe, scan around the clevis end adjacent to the bushing. Scan the clevis ends as illustrated in a clockwise and counterclockwise direction on both sides of the clevis.

Note

An external circumferential slot with lubrication holes is machined in the wall of each bushing and must be compensated for as the probe passes through the bushing.

(2) Bushings - Using the appropriate bolt hole probe, scan the inner surface of the bushings illustrated.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.3.2.4 (Cont).

h. No backup is required for the steel bushing inspection. To confirm suspected defects in the clevis/lug, proceed with the Backup NDE Procedure. If no defects are suspected, proceed to Primary NDE Procedure No. 3.

58.3.2.5 PRIMARY NDE PROCEDURE NO. 3 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The attachment bolts and empennage scissors pin are accessible when removed from the aircraft.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts and scissors pin of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

JE 308-1 Bolts - 750 amperes

JE 309-1 Bolt - 1000 amperes

JE 310-1 Pin - 600 amperes

(2) Longitudinal field:

JE 308-1 Bolts - 2000 amperes

JE 309-1 Bolt - 2500 amperes

JE 310-1 Pin - 1000 amperes

f. Inspection: Inspect the bolts and scissors pin by the magnetic particle wet continuous method, using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. If no defects are indicated, proceed to System Securing.

58.3.2.6 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

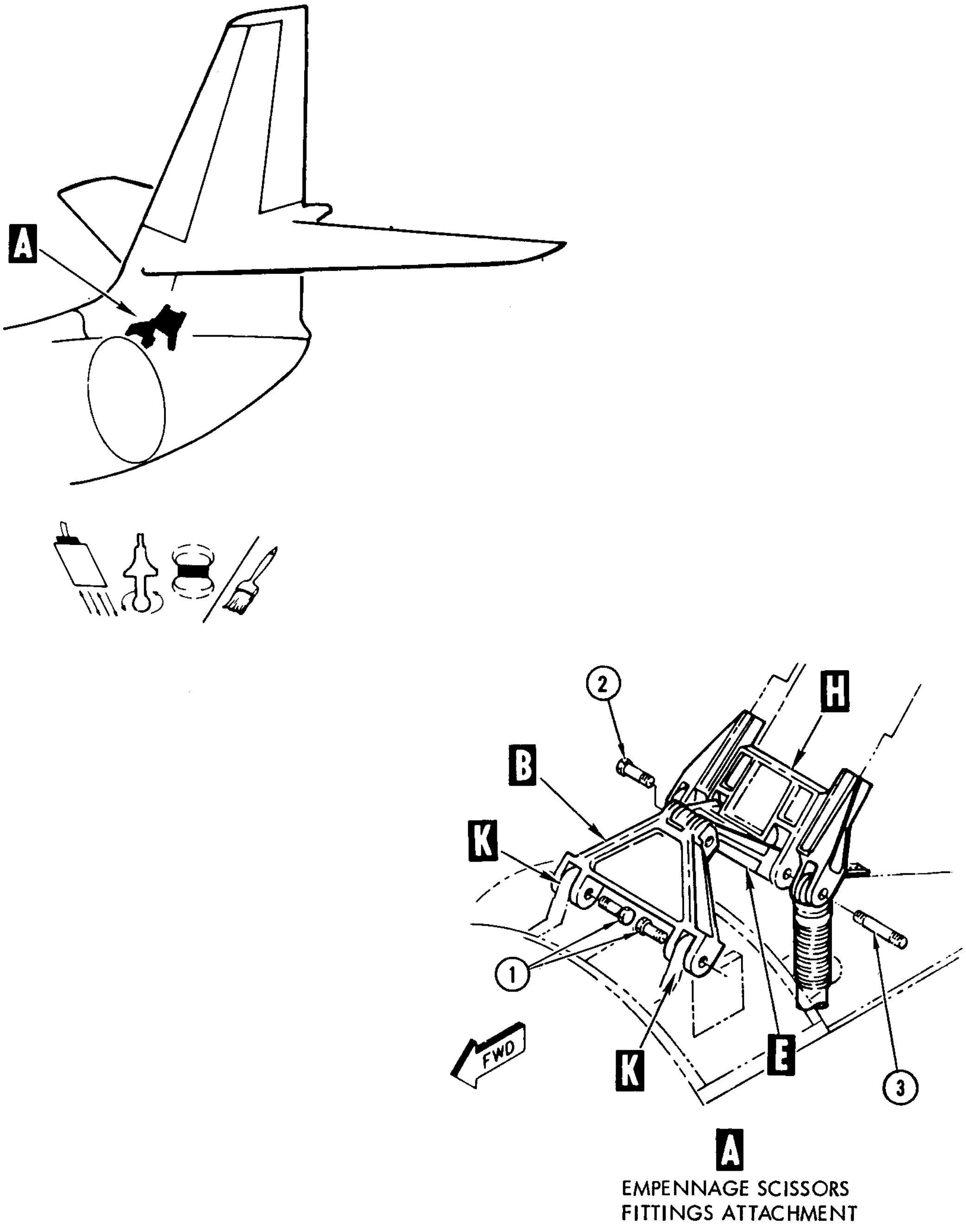
d. Preparation of part: Remove the bushing in accordance with Section 59.1 of this manual. Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

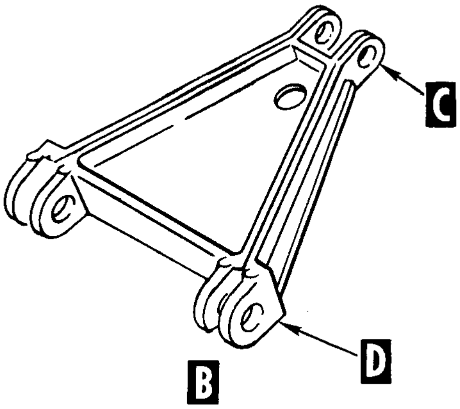
g. Mark and report indicated defects.

58.3.2.7 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

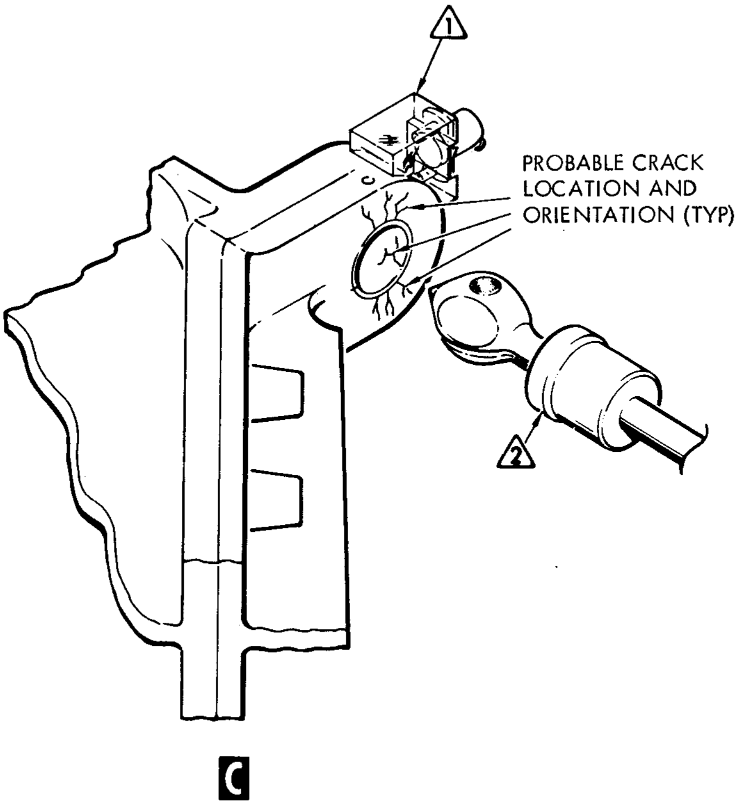


SMR151-58.3-0-001-1

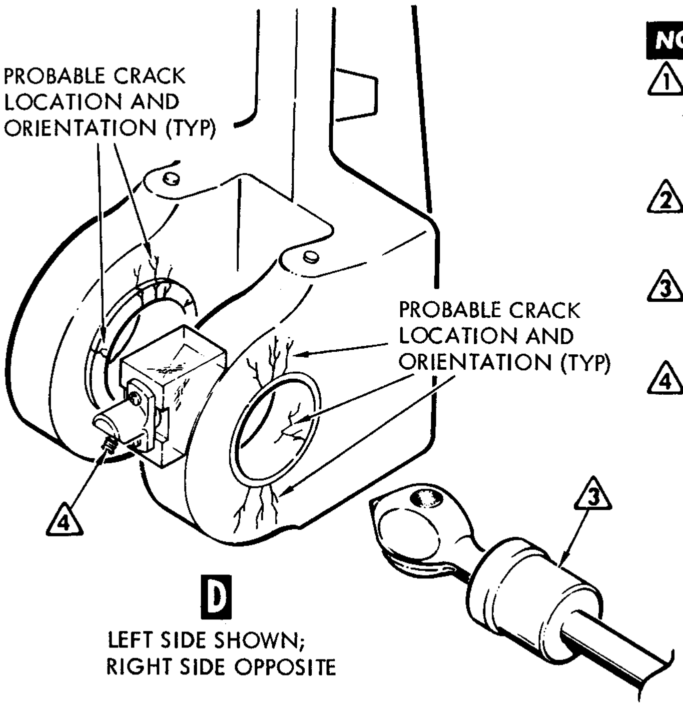
Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 1 of 9)



FUSELAGE SCISSORS FITTING



LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
4
3
D

LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

NOTE

1 WEDGE ASSEMBLY (PDL 949-2) - INSPECT BOTH SEGMENTS OF CLEVIS LUG BY SCANNING IN BOTH DIRECTIONS ON THE END OF THE FITTING, PRIMARY NDE PROCEDURE NO. 1.

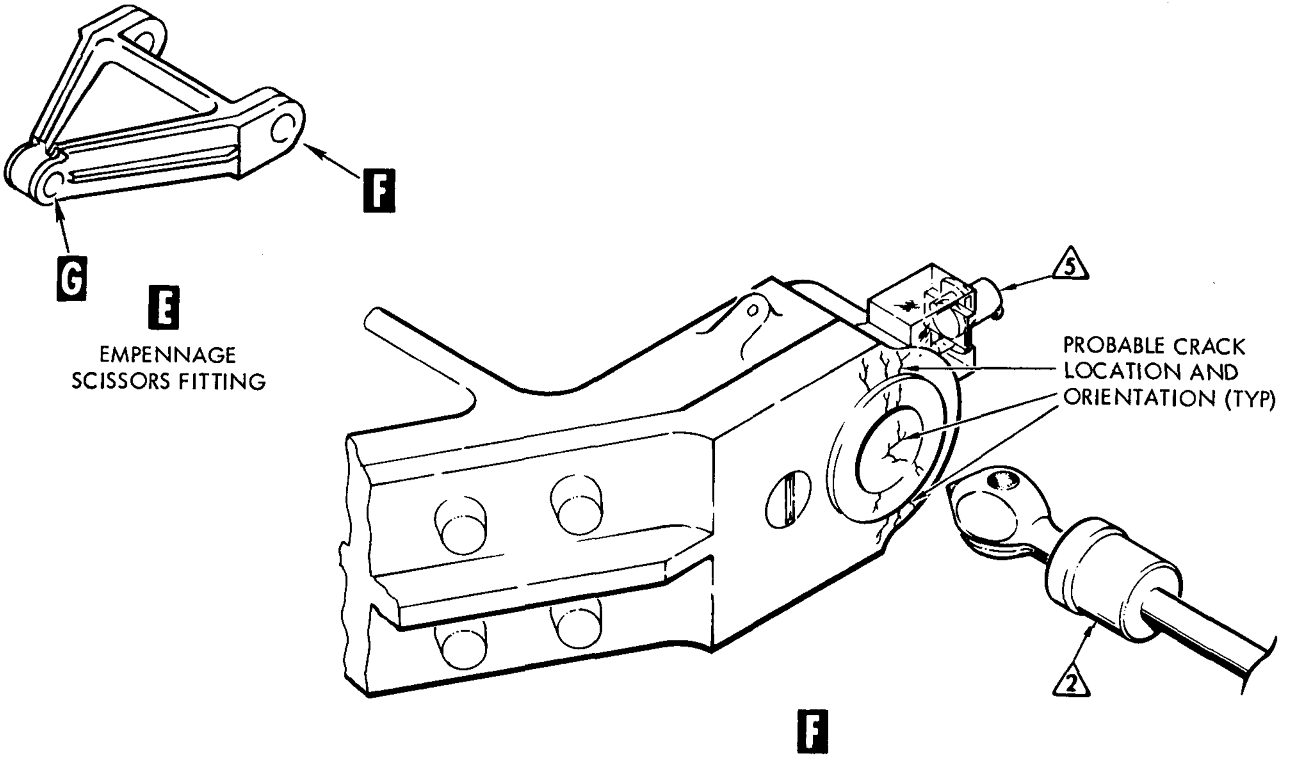
2 BOLT HOLE PROBE (1/2-INCH) - SCAN ENTIRE INNER SURFACE OF BUSHING, PRIMARY NDE PROCEDURE NO. 2.

3 BOLT HOLE PROBE (3/4-INCH) - SCAN ENTIRE INNER SURFACE OF BUSHING IN EACH LUG OF CLEVIS, PRIMARY NDE PROCEDURE NO. 2.

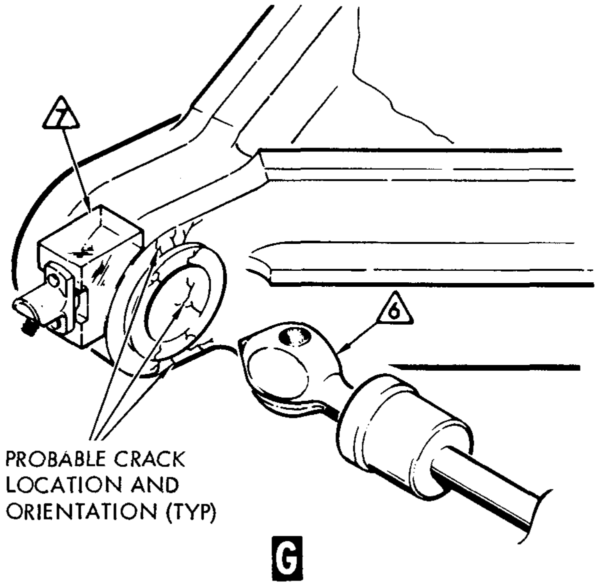
4 WEDGE ASSEMBLY (PDL 950) - INSPECT EACH LUG OF CLEVIS BY SCANNING IN BOTH DIRECTIONS ON THE END OF THE FITTING, PRIMARY NDE PROCEDURE NO. 1.

SMR151-58.3-0-001-2

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 2 of 9)



LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



NOTE

△5 WEDGE ASSEMBLY (PDL 949-1) - INSPECT BOTH SEGMENTS OF LUG BY SCANNING IN BOTH DIRECTIONS ON END OF FITTING, PRIMARY NDE PROCEDURE NO. 1.

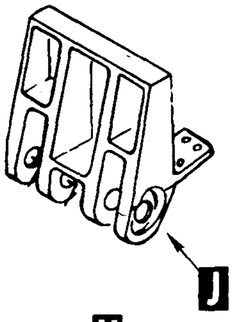
△2 BOLT HOLE PROBE (1/2-INCH) - SCAN ENTIRE INNER SURFACE OF BUSHING, PRIMARY NDE PROCEDURE NO. 2.

△6 BOLT HOLE PROBE (1/2-INCH) - SCAN ENTIRE INNER SURFACE OF BOTH BUSHINGS IN LUG, PRIMARY NDE PROCEDURE NO. 2.

△7 WEDGE ASSEMBLY (PDL 949-2) - INSPECT ALL THREE SEGMENTS OF LUG BY SCANNING IN BOTH DIRECTIONS ON END OF FITTING, PRIMARY NDE PROCEDURE NO. 1.

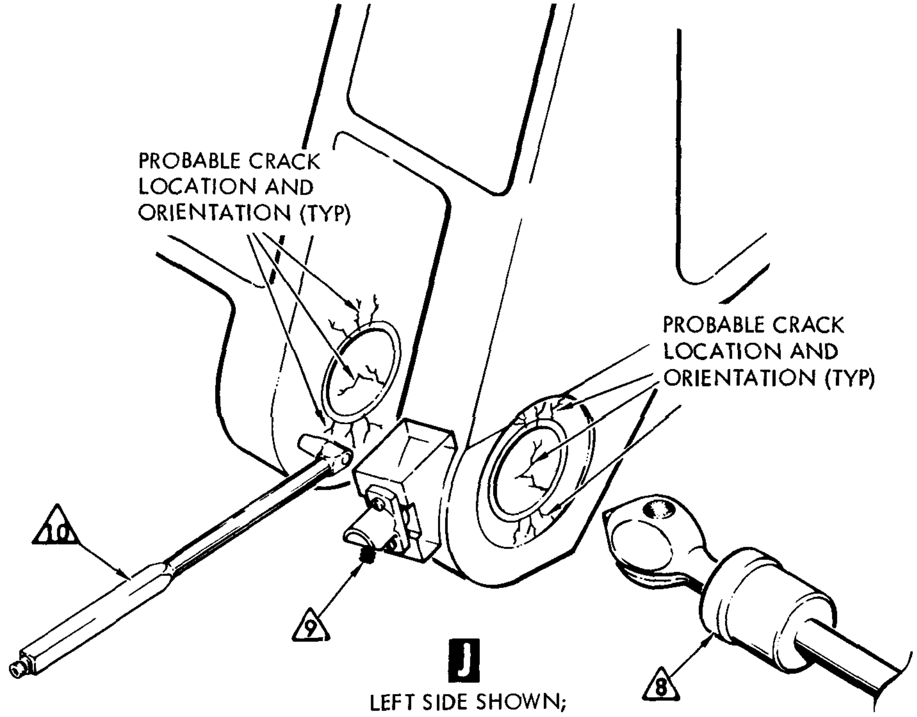
SMR151-58.3-0-001-3

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 3 of 9)



H

EMPENNAGE
ATTACH FITTING



J

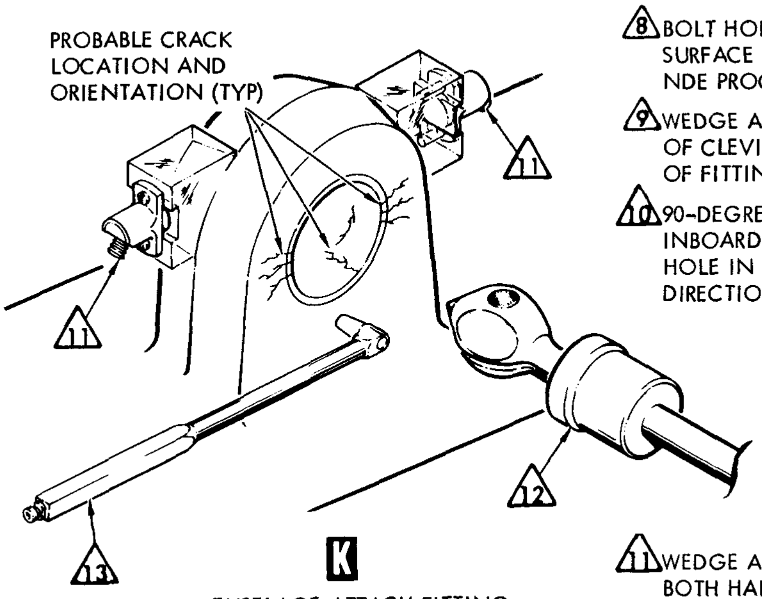
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

NOTE

△8 BOLT HOLE PROBE (1/2-INCH) - SCAN ENTIRE INNER SURFACE OF BUSHING IN EACH LUG OF CLEVIS, PRIMARY NDE PROCEDURE NO. 2.

△9 WEDGE ASSEMBLY (PDL 951) - INSPECT OUTBOARD LUG OF CLEVIS BY SCANNING IN BOTH DIRECTIONS ON END OF FITTING, PRIMARY NDE PROCEDURE NO. 1.

△10 90-DEGREE PROBE-SCAN INNER AND OUTER SURFACE OF INBOARD LUG OF CLEVIS COMPLETELY AROUND BUSHED HOLE IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION, PRIMARY NDE PROCEDURE NO. 2.



K

FUSELAGE ATTACH FITTING
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

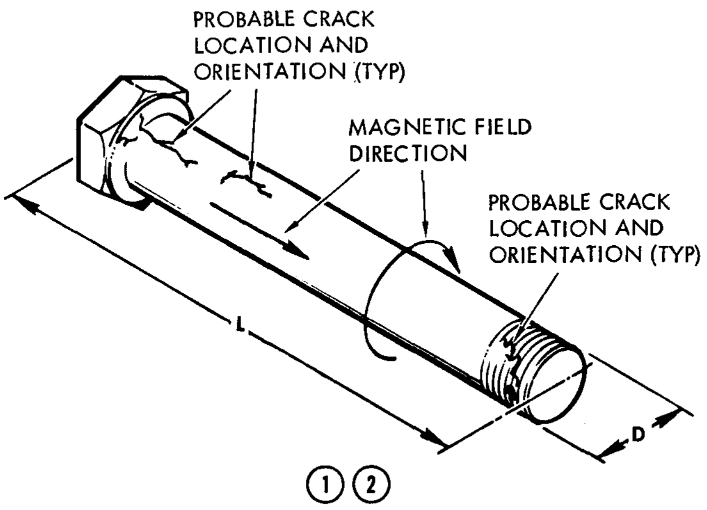
△11 WEDGE ASSEMBLIES (PDL 952-1 AND 952-2) - INSPECT BOTH HALVES OF LUG BY SCANNING IN BOTH DIRECTIONS ON END OF FITTING, PRIMARY NDE PROCEDURE NO. 1.

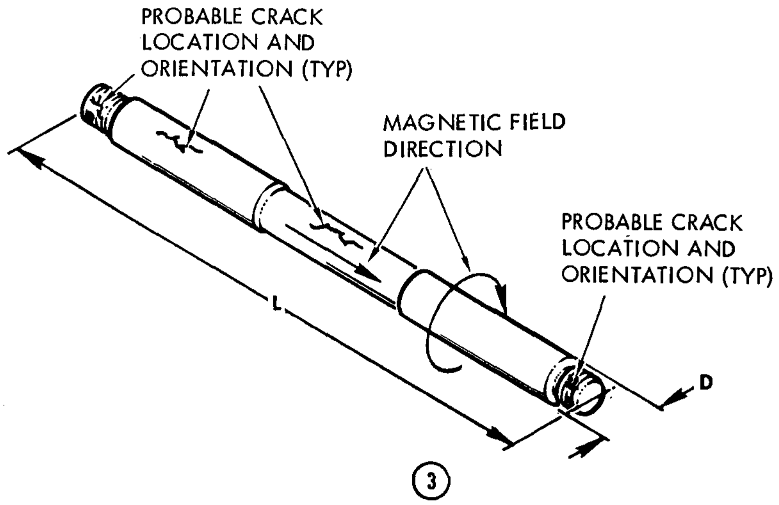
△12 BOLT HOLE PROBE (3/4-INCH) - SCAN ENTIRE INNER SURFACE OF BUSHING, PRIMARY NDE PROCEDURE NO. 2.

△13 90-DEGREE PROBE - SCAN OUTBOARD SURFACE OF LUG COMPLETELY AROUND BUSHED HOLE IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION, PRIMARY NDE PROCEDURE NO. 2.

SMR151-58.3-0-001-4

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 4 of 9)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
① ②


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
③

BOLT/PIN INSPECTION
PRIMARY NDE PROCEDURE NO. 3

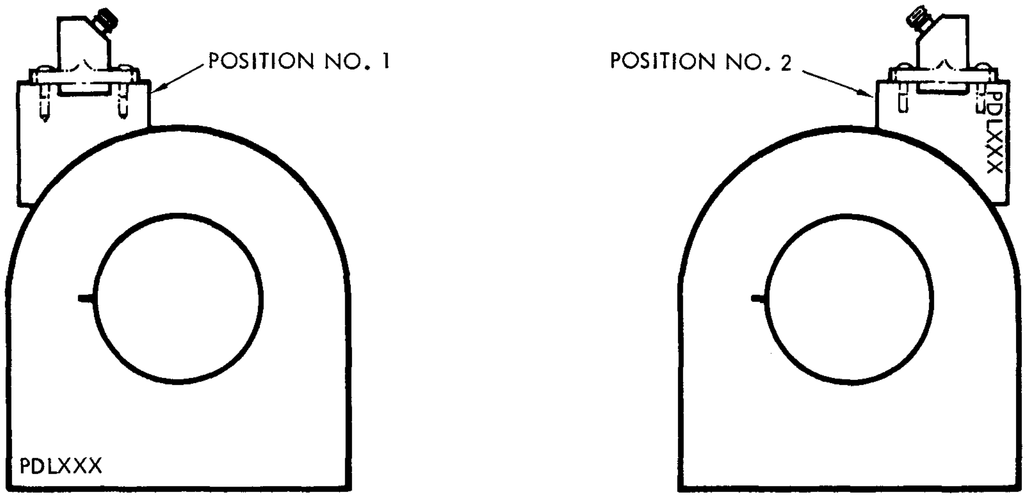
	BOLT/PIN	D (IN)	L (IN)	MATERIAL	FINISH
①	JE308-1	0.75	3.35	△14	△16
②	JE309-1	0.50	3.61	△14	△16
③	JE310-1	0.56	8.35	△15	△16

NOTE

△14 NITRALLOY 135, HEX HEAD BOLT
△15 NITRALLOY 135, PIN
△16 GRIP LENGTH-NITRIDED;
REMAINDER-CADMIUM PLATED

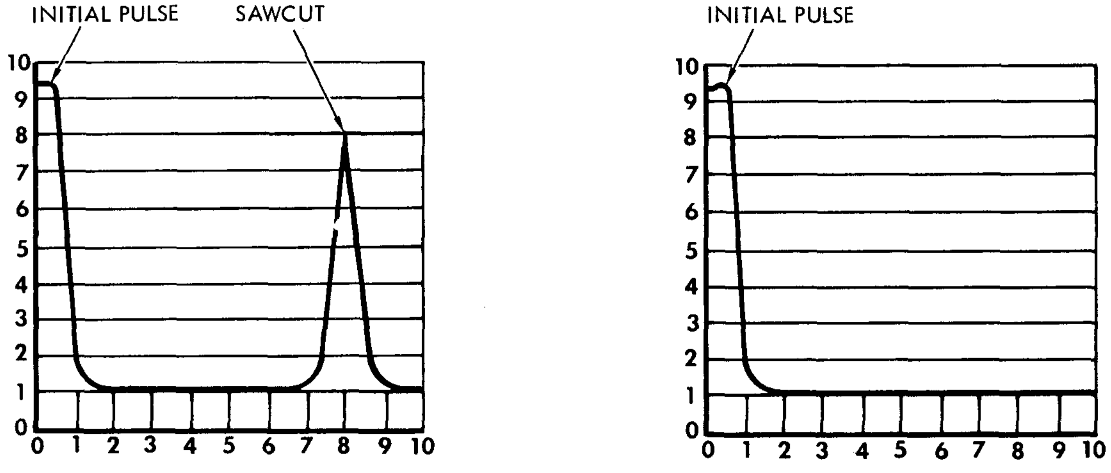
SMR151-58.3-0-001-5

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 5 of 9)



CALIBRATION SETUP
PRIMARY NDE PROCEDURE NO. 1

WEDGE ASSEMBLY	CALIBRATION STANDARD
PDL 949-1	PDL 935-1
PDL 949-2	PDL 935-2
PDL 950	PDL 937
PDL 951	PDL 938
PDL 952-1	PDL 939
PDL 952-2	PDL 939



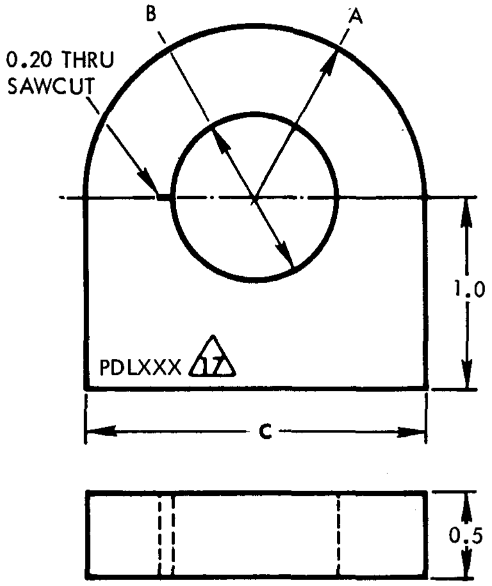
POSITION NO. 1

POSITION NO. 2

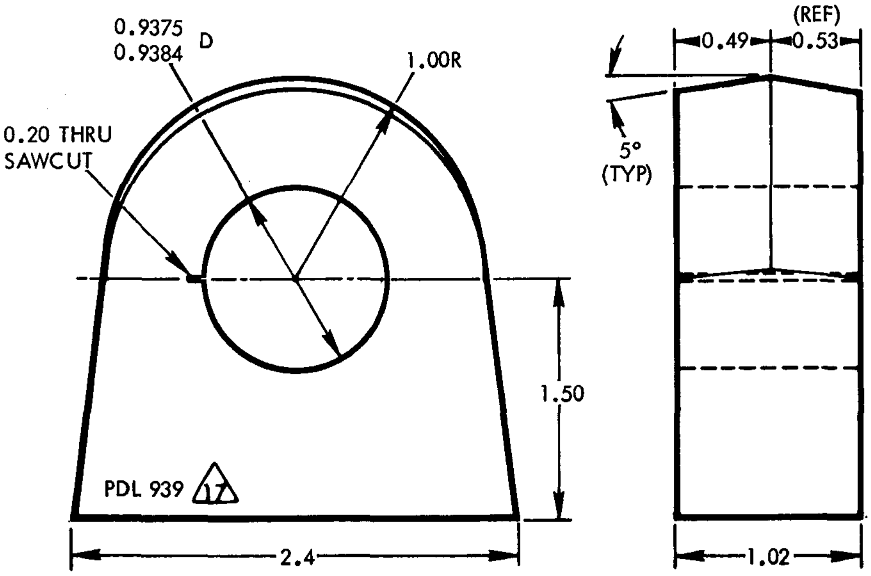
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 1

SMR151-58.3-0-001-6

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 6 of 9)



CALIBRATION STANDARD NUMBER	DIMENSIONS (INCHES)		
	A	B	C
PDL 935-1	0.63R	0.6879 0.6884 D	1.26
PDL 935-2	0.75R	0.7504 0.7509 D	1.50
PDL 937	1.00R	0.9379 0.9384 D	2.00
PDL 938	0.93R	0.9379 0.9384 D	1.86



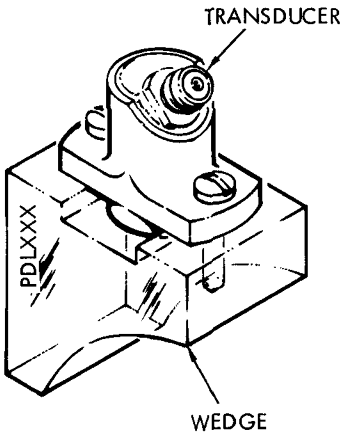
CALIBRATION STANDARDS
(7075-T6 ALUMINUM OR EQUIVALENT)

NOTE

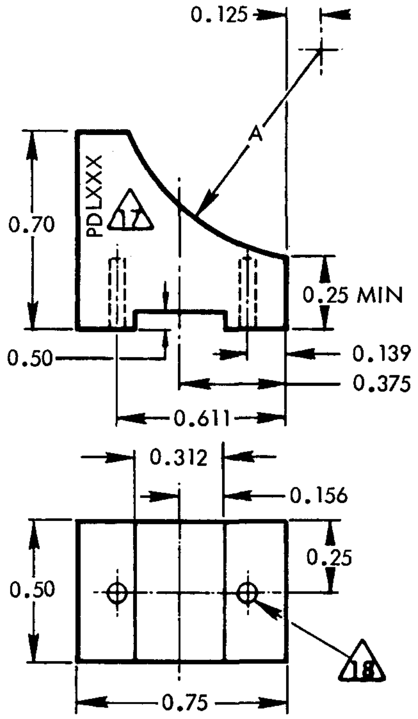
17 SCRIBE ON PART

SMR151-58.3-0-001-7

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 7 of 9)



WEDGE ASSEMBLY
(TYPICAL)



WEDGE NUMBER	DIMENSION A (INCHES)
PDL 949-1	0.63R
PDL 949-2	0.75R
PDL 950	1.00R
PDL 951	0.93R

WEDGE
(G-GRADE PLEXIGLASS OR EQUIVALENT)

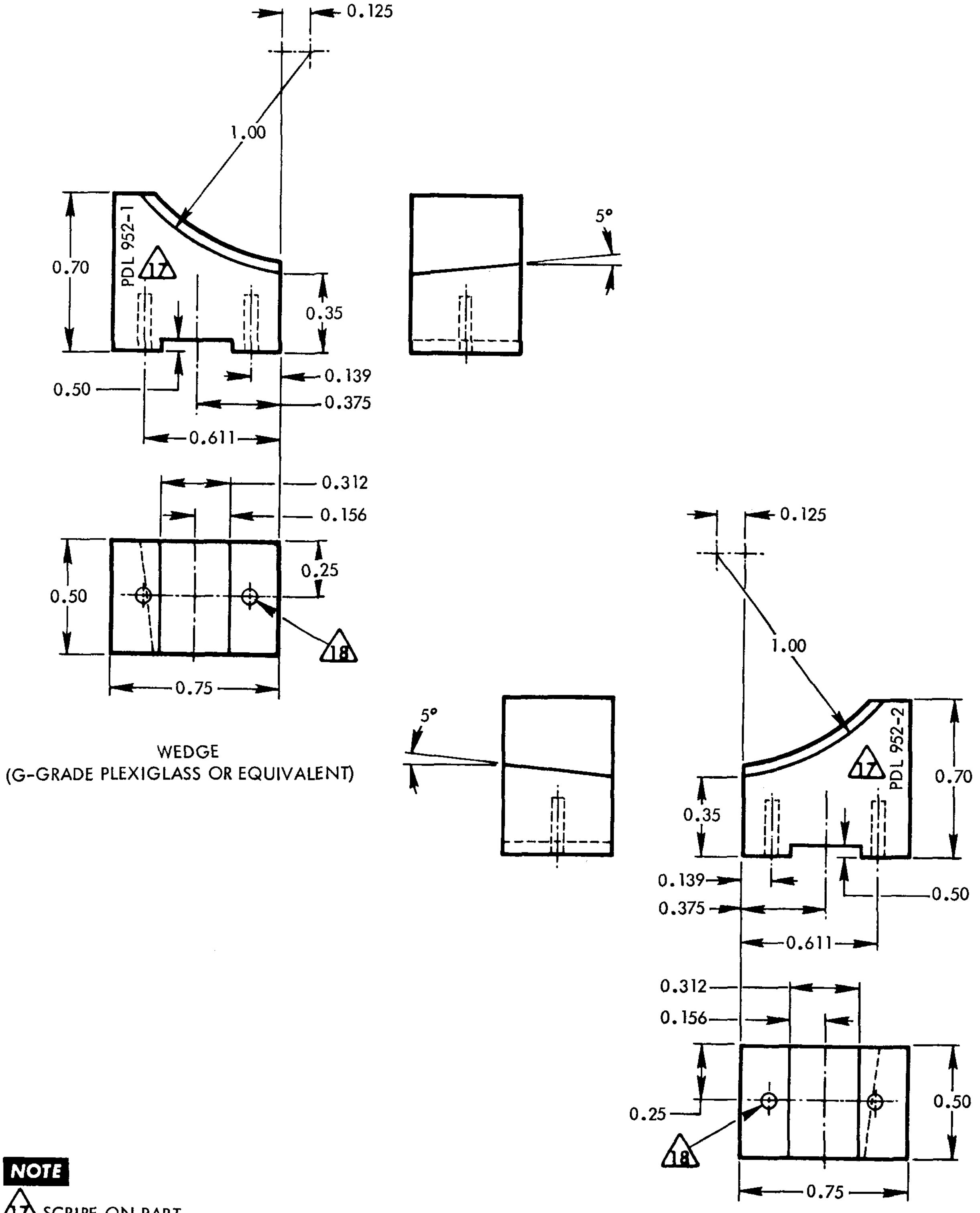
NOTE

17 SCRIBE ON PART

18 DRILL AND TAP FOR 2-56 SCREW 0.250 DEEP (2 PLACES)

SMR151-58.3-0-001-8

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 8 of 9)



WEDGE
(G-GRADE PLEXIGLASS OR EQUIVALENT)

WEDGE
(G-GRADE PLEXIGLASS OR EQUIVALENT)

NOTE

17 SCRIBE ON PART

18 DRILL AND TAP FOR 2-56 SCREW 0.250 DEEP (2 PLACES)

SMR151-58.3-0-001-9

Figure 58.3-1-5/2. Empennage Scissors Fittings, Attachment Fittings, and Bolts (Sheet 9 of 9)

58.3.3 EMPENNAGE PITCH-TRIM ACTUATOR ATTACHMENT FITTINGS AND BOLTS.

58.3.3.1 DESCRIPTION. (See figure 58.3-2-3/2.) The pitch trim system includes a dual actuator assembly that pivots the entire empennage to provide nose-up or nose-down trim of the airplane. The upper end of the actuator assembly is attached to fittings on the lower corners of the vertical stabilizer front beam. The lower end of the actuator assembly is attached to fittings on the aft side of fuselage station 682 bulkhead. The upper attach fittings are fabricated from 8630 steel forgings and are cadmium plated. The lower attach fittings are fabricated from 7075-T6 aluminum forgings and are finished with zinc chromate primer. Nitralloy 135 bushings are installed in each fitting and the actuator assembly is secured to the fittings with nitralloy 135 bolts. The inside surface of the bushings and the grip length of the bolts are treated with nitride, while the remaining surface of the bushings and bolts are cadmium plated.

58.3.3.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the attach fittings, bushings, and bolts. In both upper and lower attach fittings, cracks will initiate in the bushed holes and propagate in a random direction. In the lower attach fittings, cracks will also initiate in the aft two fastener holes of the upper and lower horizontal flanges and propagate in an inboard-outboard direction, and in the radii of the horizontal flanges and propagate in a longitudinal and transverse direction. Cracks in the bushings and bolts will propagate in a circumferential and longitudinal direction.

Note

Primary NDE Procedure No. 1, No. 2 and No. 3 must be performed to complete this inspection.

58.3.3.3 PRIMARY NDE PROCEDURE NO. 1 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Portable Hand Probe Unit, Parker Research P/N DA-200 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 AM Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with the appropriate sections of this manual.

(2) Remove the pitch trim actuator assembly in accordance with Section 27.3 of this manual.

c. Access: The clevis area of the upper attach fitting is accessible with the pitch trim actuator removed.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the clevis of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current selector to the full clockwise position.

f. Inspection: Inspect the upper attach fittings and bushings by the magnetic particle method using the portable hand probe and the magnetic particle spray. Apply the magnetic field and inspect each clevis in at least two orientations approximately 90 degrees apart. Demagnetize the clevis upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. If no defects are indicated, proceed to Primary NDE Procedure No. 2.

58.3.3.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM100SS, or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM102AS or equivalent.

(4) Probe, Bolt Hole, 3/4-inch diameter, VM Products P/N VM101B-3/4 or equivalent.

(5) Calibration Standard, Aluminum (See figures 58.1-9).

(6) Calibration Standard, Steel (See figures 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The clevis area of the lower attach fitting is accessible on the aft side of the Fuselage Station 682 bulkhead with the pitch trim actuator removed. The horizontal flange is accessible on the forward side of the bulkhead.

58.3.3.4 (Cont).

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Clean the surface area of the bushing, around fastener head, and radius as required to permit good contact between part and probe. Refer to Section 58.0 for cleaning instructions.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using the 90-degree probe, scan outer and inner surfaces of each clevis lug completely around the bushed hole in a clockwise and counterclockwise direction.

(2) Using the 90-degree and/or pencil probe, scan around the two most aft fasteners in the upper and lower horizontal flanges of both fittings.

(3) Using the 90-degree and/or pencil probe, scan the upper and lower forward radii between the vertical and forward horizontal flanges of both fittings.

(4) Using the bolt hole probe, scan the entire inner surface of bushing in each clevis lug.

(5) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

h. To confirm suspected defects, proceed with the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 3.

58.3.3.5 PRIMARY NDE PROCEDURE NO. 3 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The attachment bolts are accessible when removed from the airplane.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

JC 336-1 Bolts - 750 amperes

JC 337-1 Bolts - 900 amperes

(2) Longitudinal field:

JC 336-1 Bolts - 2800 amperes

JC 337-1 Bolts - 2700 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. If no defects are indicated, proceed to System Securing.

58.3.3.6 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

58.3.3.6 (Cont).

c. Access: Same as Primary NDE Procedure No. 1.

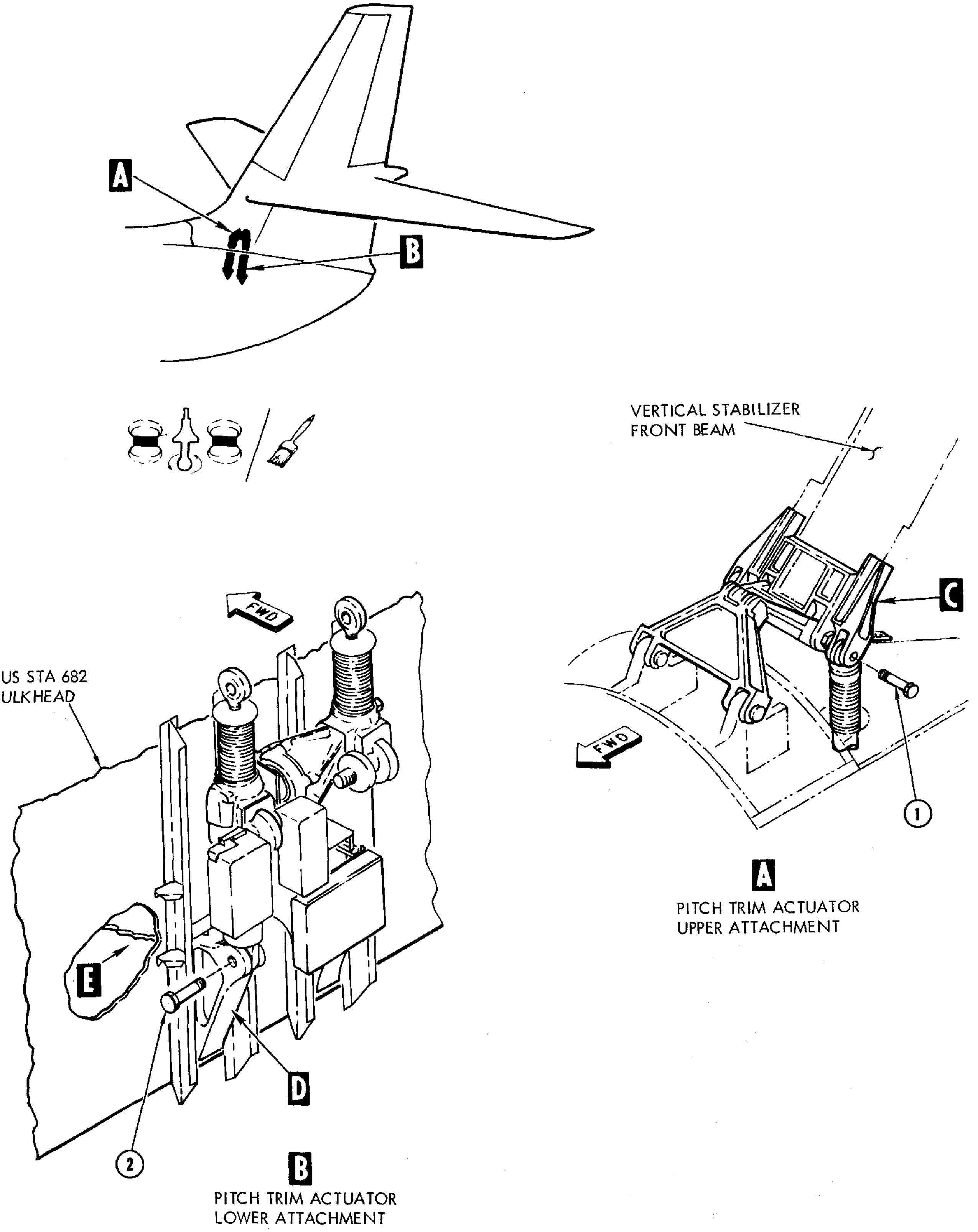
d. Preparation of part: If installed, remove the bushing from suspected defective clevis lug in accordance with Section 59.1 of this manual. Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective area.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective area. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.3.3.7 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

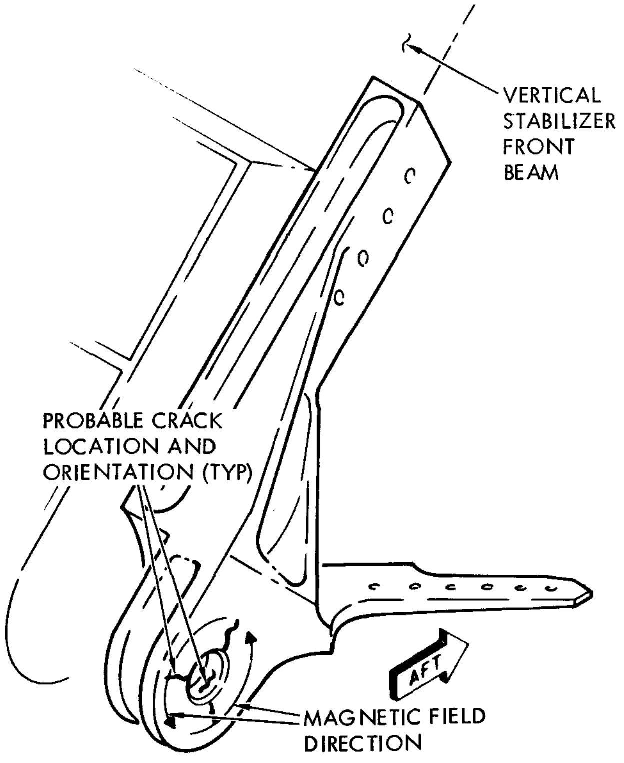


A
PITCH TRIM ACTUATOR UPPER ATTACHMENT

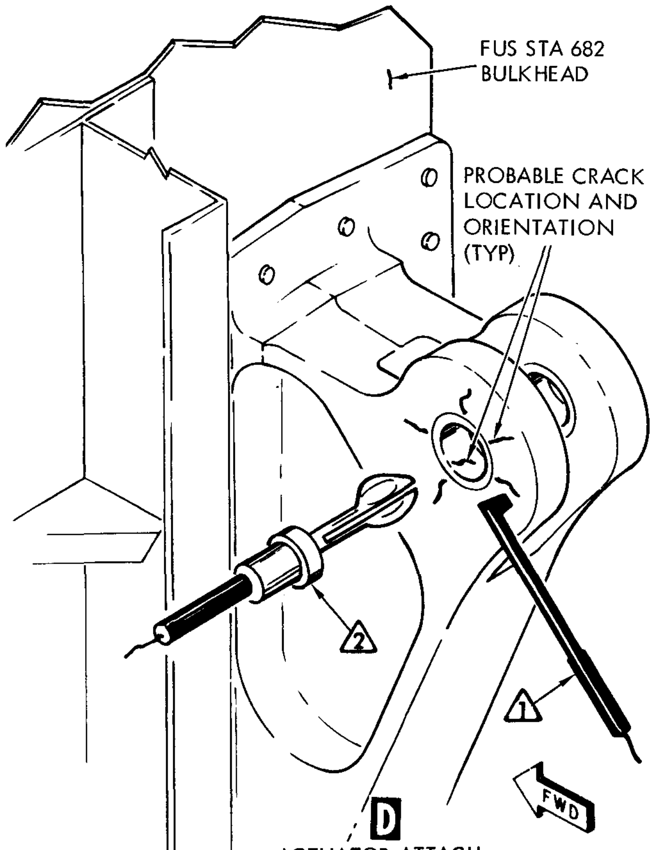
B
PITCH TRIM ACTUATOR LOWER ATTACHMENT

SMR151-58.3-0-002-1

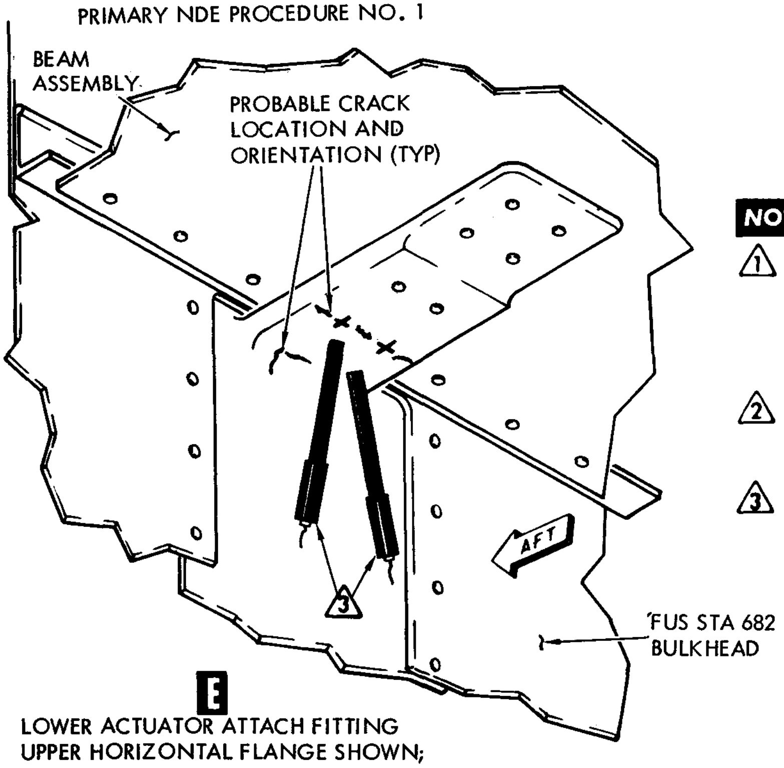
Figure 58.3-2-3/2. Empennage Pitch Trim Actuator Attachment Fittings and Bolts (Sheet 1 of 3)



ACTUATOR ATTACH FITTING, UPPER
LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE
PRIMARY NDE PROCEDURE NO. 1



ACTUATOR ATTACH
FITTING, LOWER
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



LOWER ACTUATOR ATTACH FITTING
UPPER HORIZONTAL FLANGE SHOWN;
LOWER HORIZONTAL FLANGE SIMILAR;
LEFT SIDE SHOWN; RIGHT SIDE SIMILAR

NOTE

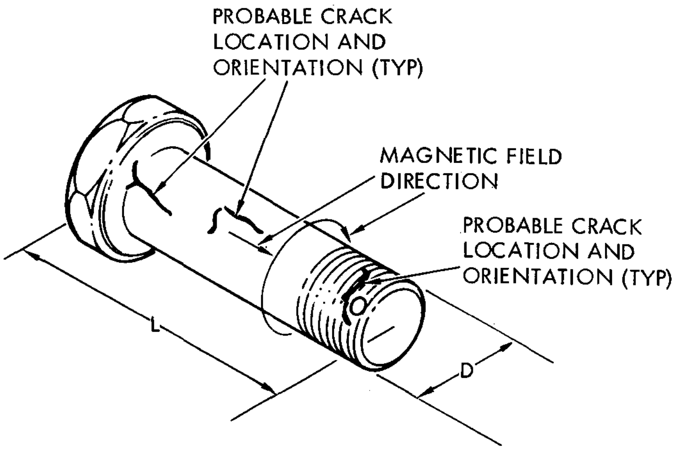
△1 90-DEGREE PROBE-SCAN OUTER AND INNER SURFACES OF EACH LUG COMPLETELY AROUND THE BUSHED HOLE IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION, PRIMARY NDE PROCEDURE NO. 2.

△2 BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF BUSHING IN EACH LUG, PRIMARY NDE PROCEDURE NO. 2.

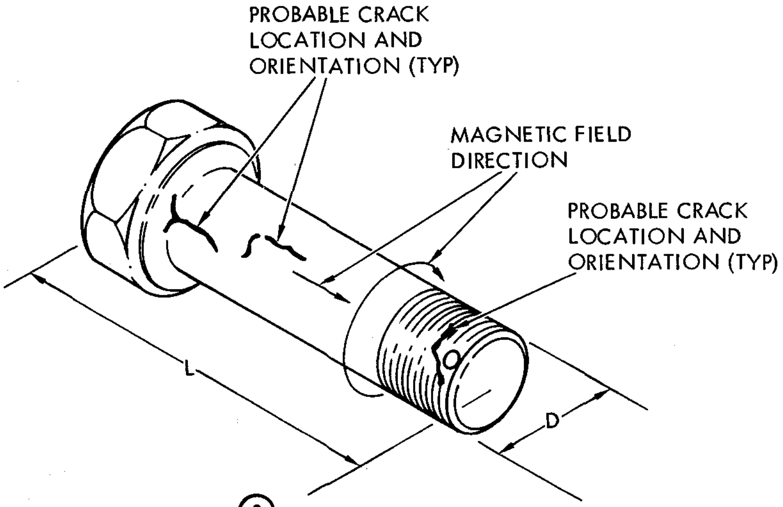
△3 90-DEGREE OR PENCIL PROBE-SCAN AROUND THE TWO INDICATED (+) FASTENERS AND IN THE RADIUS, PRIMARY NDE PROCEDURE NO. 2.

SMR151-58.3-0-002-2

Figure 58.3-2-3/2. Empennage Pitch Trim Actuator Attachment Fittings and Bolts (Sheet 2 of 3)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D

①
PRIMARY NDE
PROCEDURE
NO. 3


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D

②
PRIMARY NDE
PROCEDURE
NO. 3

BOLT	D (IN)	L (IN)	MATERIAL	FINISH
① JC336-1	0.75	2.40	△5	△6
② JC337-1	0.87	2.98	△5	△6

NOTE

△5 NITRALLOY 135, HEX HEAD

△6 GRIP LENGTH-NITRIDED; REMAINDER-CADMIUM PLATED.

SMR151-58.3-0-002-3

Figure 58.3-2-3/2. Empennage Pitch Trim Actuator Attachment Fittings and Bolts (Sheet 3 of 3)

58.3.4 EMPENNAGE PIVOT FITTINGS, BUSHINGS, AND HINGE PINS.

58.3.4.1 DESCRIPTION. (See figure 58.3-3-5/2.) The vertical stabilizer aft attachment to the fuselage provides a stationary pivot point for the empennage. The attachment includes the empennage pivot fitting which is attached to the vertical stabilizer rear beam and the aft empennage attach fittings which are attached to the fuselage structure. Both fittings are fabricated from 2014-T6 aluminum forgings and are finished with zinc chromate primer. Nitralloy 135 bushings are installed in each fitting and the fittings are linked by nitralloy 135 hinge pins. The inside surface of the bushings and outside surface of the pins are treated with nitride, while the remaining surface of the bushings and pins are cadmium plated.

58.3.4.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the fittings, bushings, and hinge pins. Cracks will initiate in the bushed holes of the fittings and propagate in a fore and aft direction. Cracks in the bushings and hinge pins will propagate in a circumferential and longitudinal direction.

Note

Primary NDE Procedure No. 1, No. 2, and No. 3 must be performed to complete this inspection.

58.3.4.3 PRIMARY NDE PROCEDURE NO. 1 - ULTRASONIC.

a. NDE equipment:

(1) Ultrasonic Flaw Detector, Sonic Model FTS Mark I, or equivalent.

(2) Transducer, 5.0 MHz, 0.25-inch diameter, longitudinal wave, Krautkramer-Branson P/N Z-103-ABFR or equivalent.

(3) Wedge, Lockheed-Georgia Company P/N PDL 954, as illustrated.

(4) Calibration Standard, Lockheed-Georgia Company P/N PDL 953, as illustrated.

(5) Cable, 6-foot, Microdot/BNC connector, Automation Industries P/N 57A2271 or equivalent.

(6) Couplant, light grease or oil.

b. Preparation of airplane:

WARNING

Ensure power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the vertical stabilizer lower fairing. Refer to Section 52.3 of this manual for access panel locations.

(3) Hoist the empennage per Section 7.0 of this manual.

c. Access: The pivot fittings are accessible with the vertical stabilizer lower fairing removed and with use of maintenance stands or appropriate AGE.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Remove excess sealant from the surface and clean local inspection areas as required to permit good contact between part and wedge.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer wedge assembly to calibration standard in Position No. 1 as illustrated.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT display. Adjust sweep to display signal from the sawcut near the right side of the CRT display.

(3) Maximize the signal from the sawcut by adjusting the wedge position and adjust the signal amplitude to obtain 80 percent scope saturation on the CRT display.

(4) Scan to Position No. 2 as illustrated, and note the loss of sawcut signal on the CRT display.

(5) Scan to Position No. 3 as illustrated and note the other sawcut signal on the CRT display.

(6) The CRT signal from the sawcut simulates a signal from a crack at the fastener hole.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 of this manual.

f. Inspection:

(1) Apply couplant and couple wedge to inspection area as illustrated.

(2) Scan the accessible outer surface on the clevis of the empennage and fuselage pivot fittings in both a clockwise and counterclockwise direction.

(3) Any signal received similar to and at approximately the same location on the CRT display as the sawcut signals observed during calibration will indicate a probable crack in the part.

g. Mark and report suspected defects.

58.3.4.3 (Cont).

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.3.4.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Bolt Hole, 1 1/2-inch diameter, VM Products P/N VM101B-1 1/2 or equivalent.

(3) Calibration Standard, Steel (See figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure No. 1 with the additional requirement to support the empennage to a no-load condition.

c. Access: The bushings are accessible with the hinge pins removed.

d. Preparation of part:

CAUTION

Do not remove more than one hinge pin at any time unless the empennage is being completely removed from the aircraft. Failure to comply could result in difficulty in reinstalling hinge pins and/or damage to structure.

(1) Remove one hinge pin from the aircraft. After the inspection of the bushings and pin, reinstall or replace that pin prior to removal of the other hinge pin.

CAUTION

The bushings have lubrication holes in the side wall which will become contaminated if cleaners or solvents are used to clean the bushing.

(2) Clean the inner surface of the bushing using clean, dry rags as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

Note

An external circumferential slot with lubrication holes is machined in the wall of some bushings and must be compensated for as the probe passes through the bushing.

f. Inspection: Using the bolt hole probe, scan the entire inner surface of the bushing in each lug of the empennage pivot and empennage attach fittings. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Inspect the hinge pins removed in step d (1) in accordance with Primary NDE Procedure No. 3.

h. Mark and report indicated defects. If no defect is indicated, reinstall or replace the hinge pins and proceed to System Securing.

58.3.4.5 PRIMARY NDE PROCEDURE NO. 3 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaflo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 2.

c. Access: The hinge pins are accessible when removed from the fittings.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the hinge pin of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field: 1500 amperes

(2) Longitudinal field: 3000 amperes

58.3.4.5 (Cont).

f. Inspection: Inspect the hinge pins by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects.

58.3.4.6 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of aircraft: Same as Primary NDE Procedure No. 2.

c. Access: Same as Primary NDE Procedure No. 1.

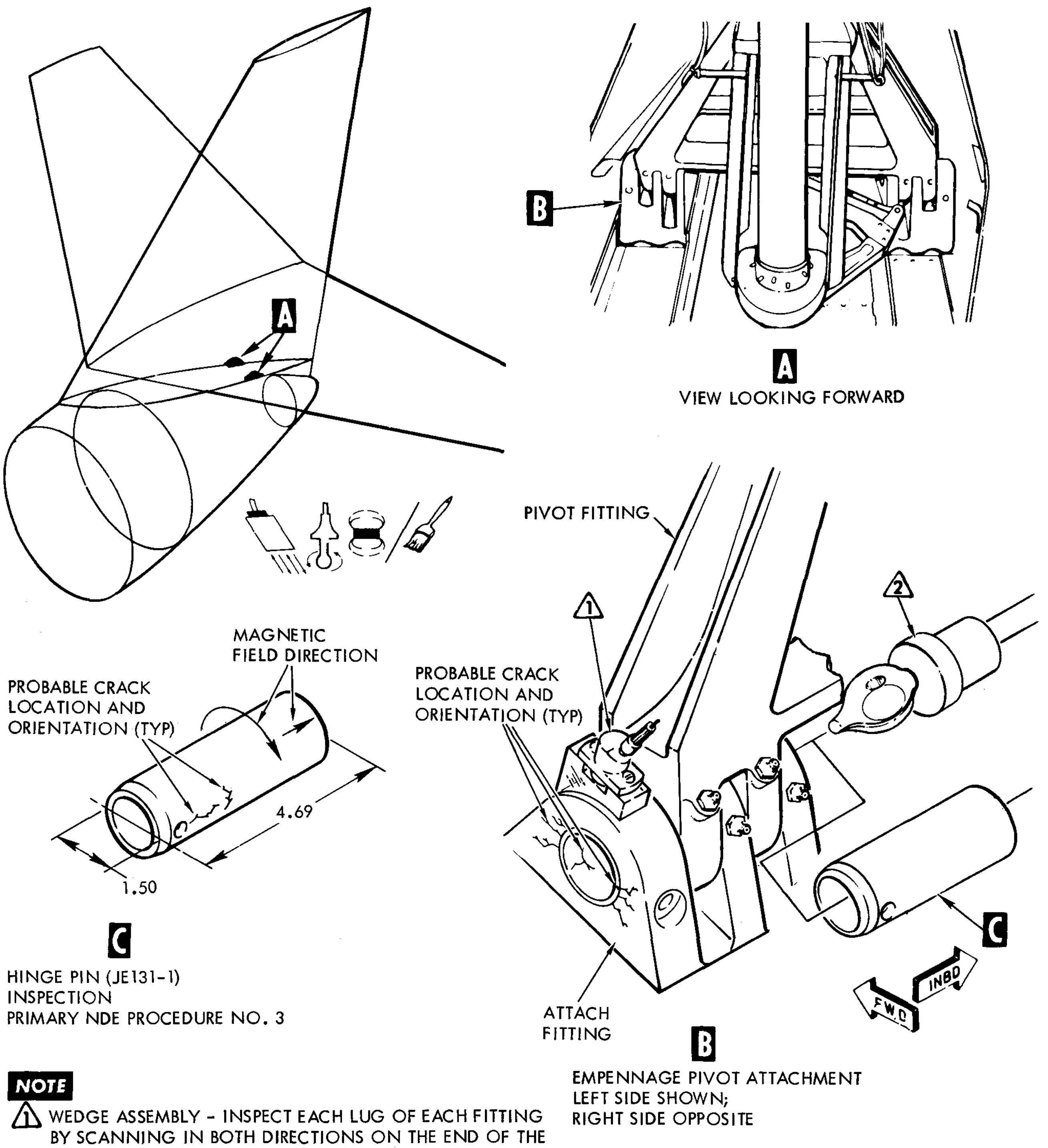
d. Preparation of part: Remove the bushing in accordance with Section 59.1 of this manual. Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.3.4.7 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



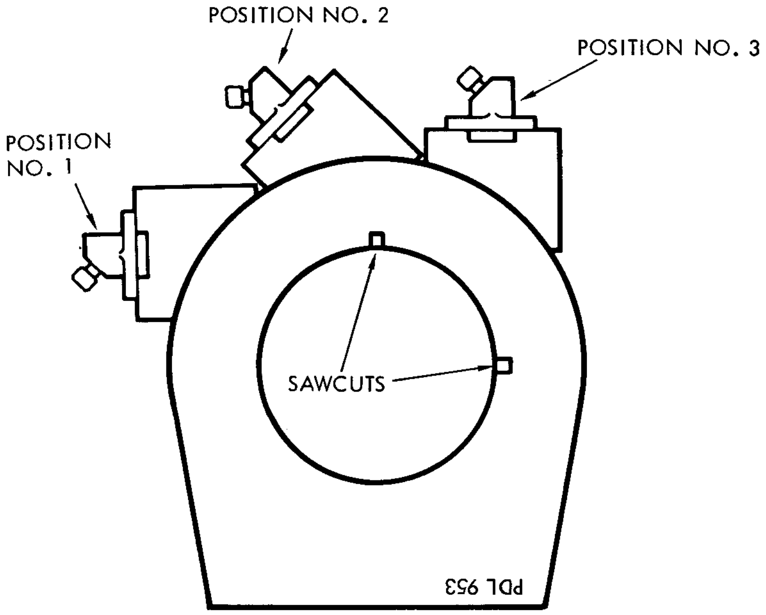
NOTE

1. WEDGE ASSEMBLY - INSPECT EACH LUG OF EACH FITTING BY SCANNING IN BOTH DIRECTIONS ON THE END OF THE FITTING, PRIMARY NDE PROCEDURE NO. 1.

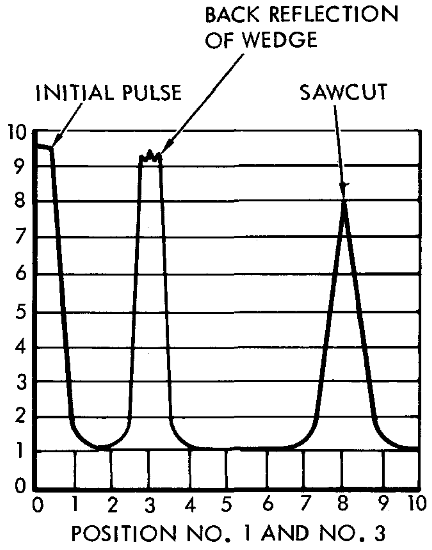
2. BOLT HOLE PROBE - SCAN ENTIRE INNER SURFACE OF BUSHING(S) IN EACH LUG OF EACH FITTING, PRIMARY NDE PROCEDURE NO. 2.

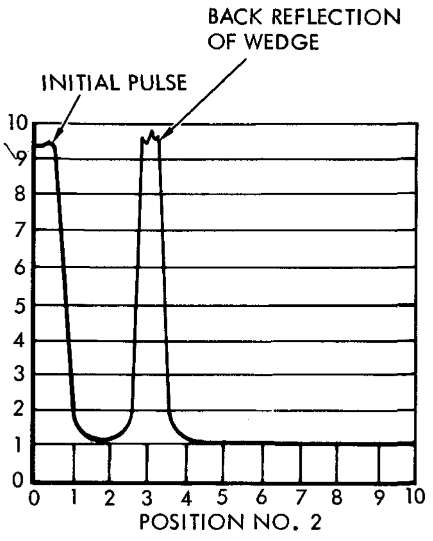
SMR151-58.3-0-003-1

Figure 58.3-3-5/2. Empennage Pivot Fittings, Bushings, and Hinge Pins (Sheet 1 of 3)



CALIBRATION SETUP
PRIMARY NDE PROCEDURE NO. 1

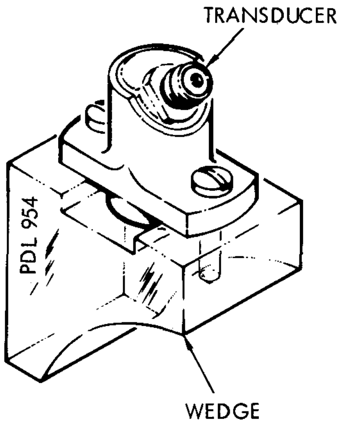




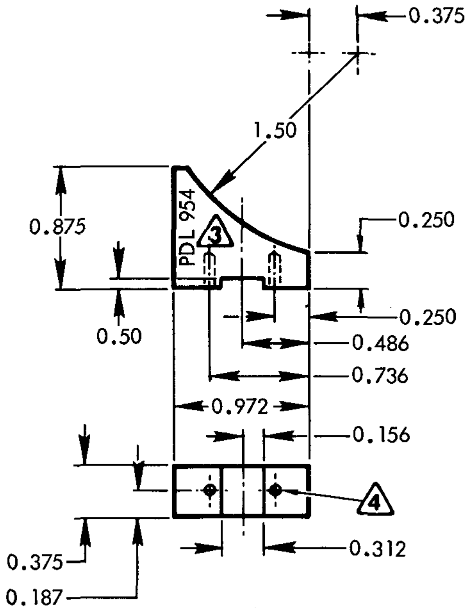
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 1

SMR151-58.3-0-003-2

Figure 58.3-3-5/2. Empennage Pivot Fittings, Bushings, and Hinge Pins (Sheet 2 of 3)



WEDGE ASSEMBLY
(TYP)



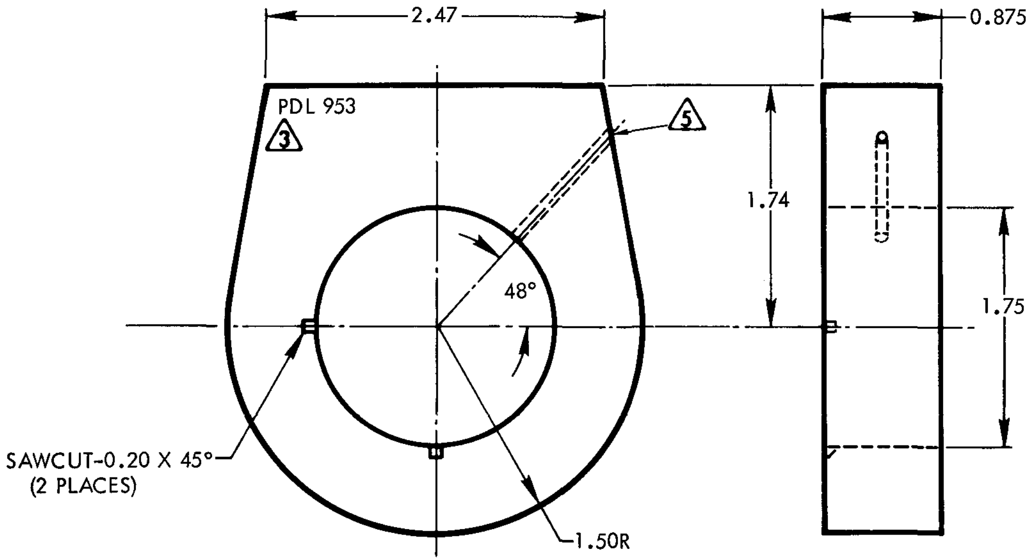
WEDGE
(G-GRADE PLEXIGLASS OR EQUIVALENT)

NOTE

3 SCRIBE ON PART

4 DRILL AND TAP FOR 2-56 SCREW 0.250 DEEP (2 PLACES)

5 NO. 54 DRILL, THRU



CALIBRATION STANDARD
(7075-T6 ALUMINUM OR EQUIVALENT)

SMR151-58.3-0-003-3

Figure 58.3-3-5/2. Empennage Pivot Fittings, Bushings, and Hinge Pins (Sheet 3 of 3)

58.3.5 HORIZONTAL STABILIZER CENTER SECTION PLATES.

58.3.5.1 DESCRIPTION. (See figure 58.3-4-6/4.) The horizontal stabilizer center section is a box structure consisting of ribs, webs, and heavy, chemically milled plates. The left and right horizontal stabilizer box sections are attached to the left and right sides of the box. The front beams of the horizontal stabilizer mate with the most forward scallop of the upper and lower center section plates and the aft beams mate with the most aft. The lower vertical stabilizer box section is attached to the lower surface and the upper vertical stabilizer is attached to the upper surface. The center section plates are machined and chemically milled from 7075-T6 aluminum. The upper and lower front and rear beam caps are machined from 7075-T6 extrusions. All parts are finished with zinc chromate primer.

58.3.5.2 DEFECTS. Inflight loads transmitted to the horizontal stabilizer center section from the horizontal stabilizers may cause fatigue cracks in the upper and lower attachment areas.

58.3.5.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the upper and lower horizontal-to-vertical stabilizer fairing skin and the trailing edge fairing installation. Refer to Chapter 55 of this manual for fairing locations.

(3) Remove the vertical stabilizer lower leading edge fairing, refer to Section 52.3 of this manual for access panel locations.

(4) Remove the trailing edge fairing, refer to Section 52.3 of this manual for fairing location.

c. Access: The inspection area is accessible on the forward and aft, upper and lower surfaces of the horizontal stabilizer center section with the use of maintenance stands.

d. Preparation of part: No special preparation is required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.3.5.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, Ideal Specialty Co. P/N 6200-3/16-BH-BNC or equivalent.

(3) Probe, Bolt Hole, 1/4-inch diameter, Ideal Specialty Co. P/N 6200-1/4-BH-BNC or equivalent.

(4) Calibration Standard, Aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

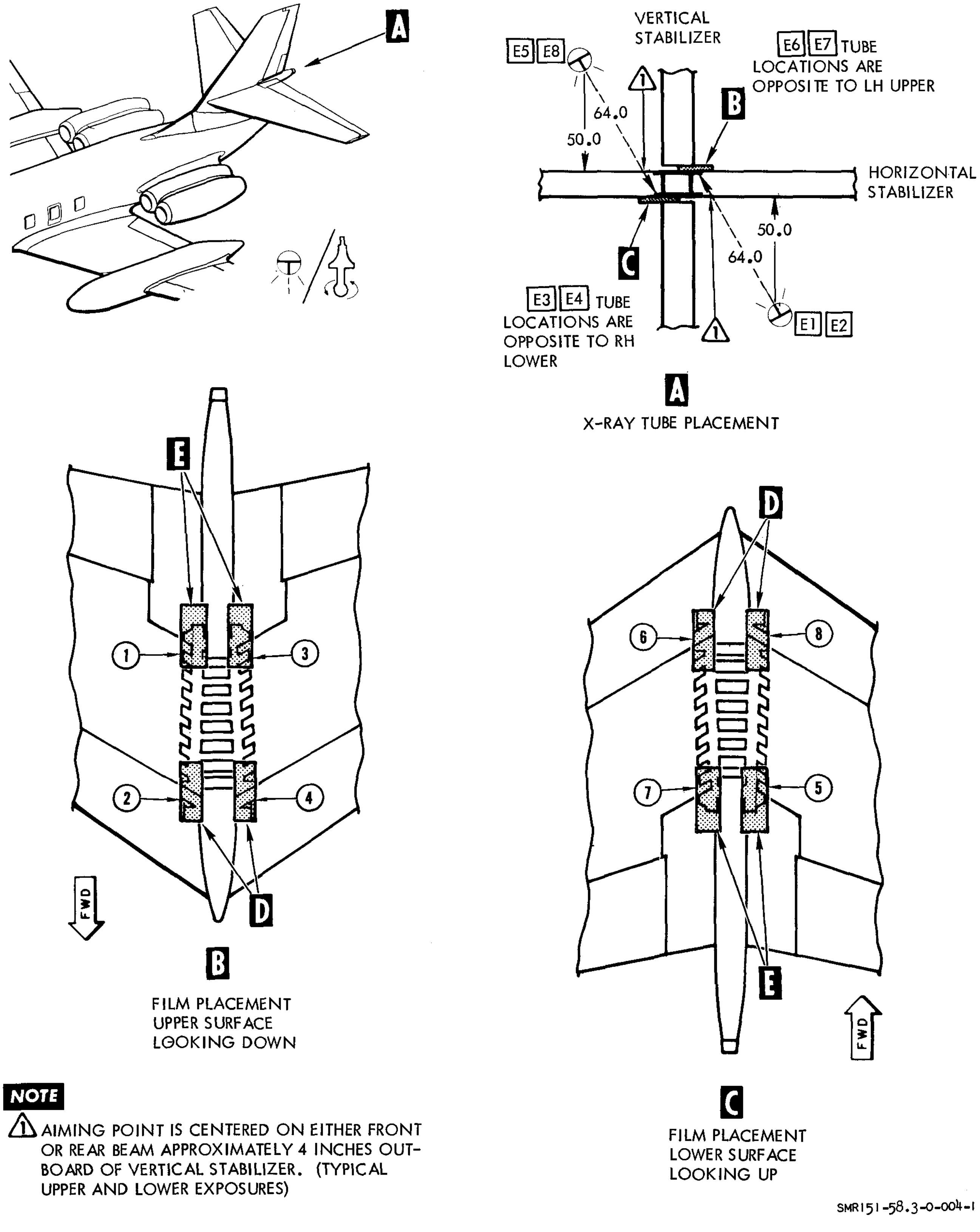
The eddy current instrument must be recalibrated when inspection probes are changed.

58.3.5.4 (Cont).

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

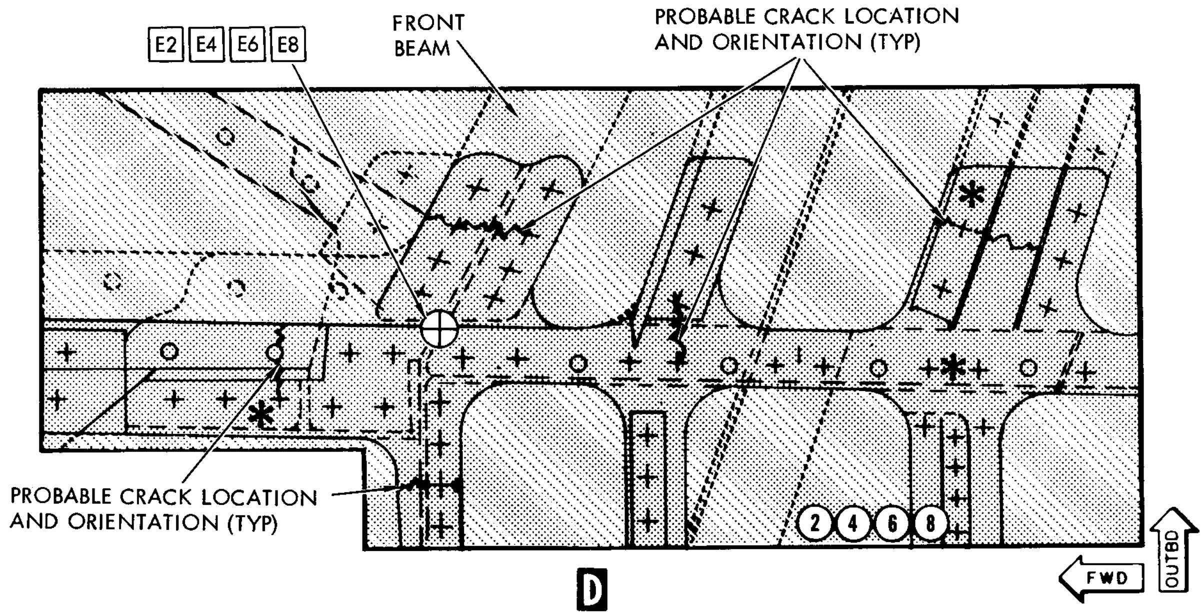
58.3.5.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


A
VERTICAL STABILIZER
E5 E8
E6 E7 TUBE LOCATIONS ARE OPPOSITE TO LH UPPER
64.0
50.0
1
B
HORIZONTAL STABILIZER
50.0
64.0
C
E3 E4 TUBE LOCATIONS ARE OPPOSITE TO RH LOWER
1
E1 E2
A
X-RAY TUBE PLACEMENT
E
①
③
②
④
D
FWD
B
FILM PLACEMENT UPPER SURFACE LOOKING DOWN
D
⑥
⑧
⑦
⑤
E
FWD
C
FILM PLACEMENT LOWER SURFACE LOOKING UP
SMR151-58.3-0-004-1

NOTE

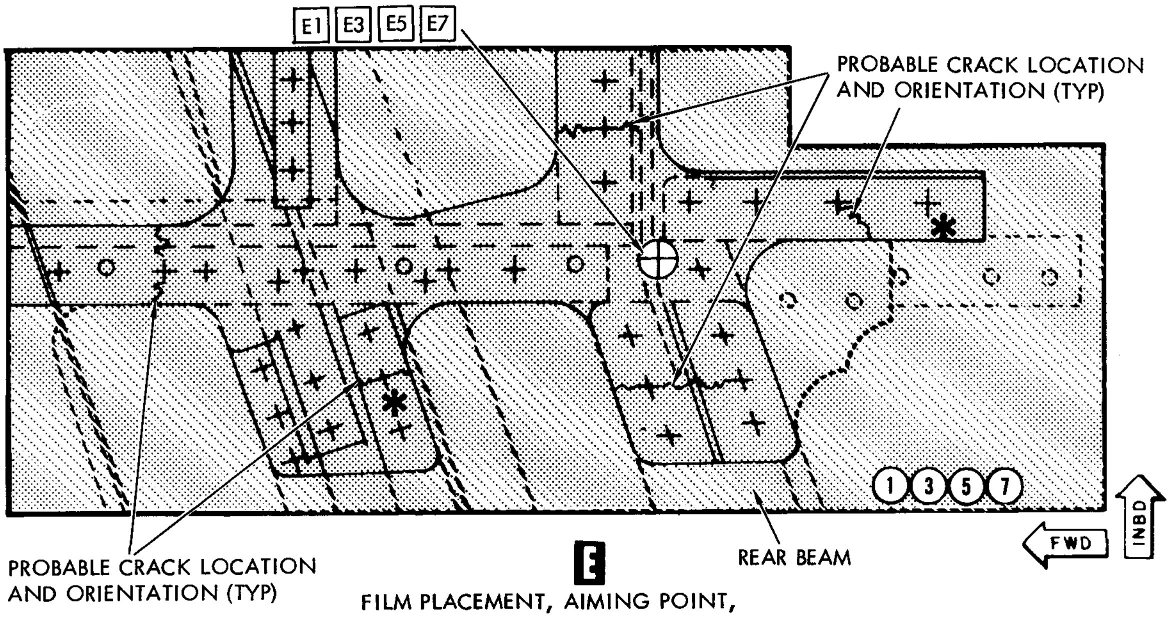
△1 AIMING POINT IS CENTERED ON EITHER FRONT OR REAR BEAM APPROXIMATELY 4 INCHES OUTBOARD OF VERTICAL STABILIZER. (TYPICAL UPPER AND LOWER EXPOSURES)

Figure 58.3-4-6/4. Horizontal Stabilizer Center Section Plates (Sheet 1 of 3)


E2 E4 E6 E8
FRONT BEAM
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
② ④ ⑥ ⑧
FWD
OUTBD

D

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS


E1 E3 E5 E7
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
① ③ ⑤ ⑦
FWD
INBD
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
REAR BEAM

E

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

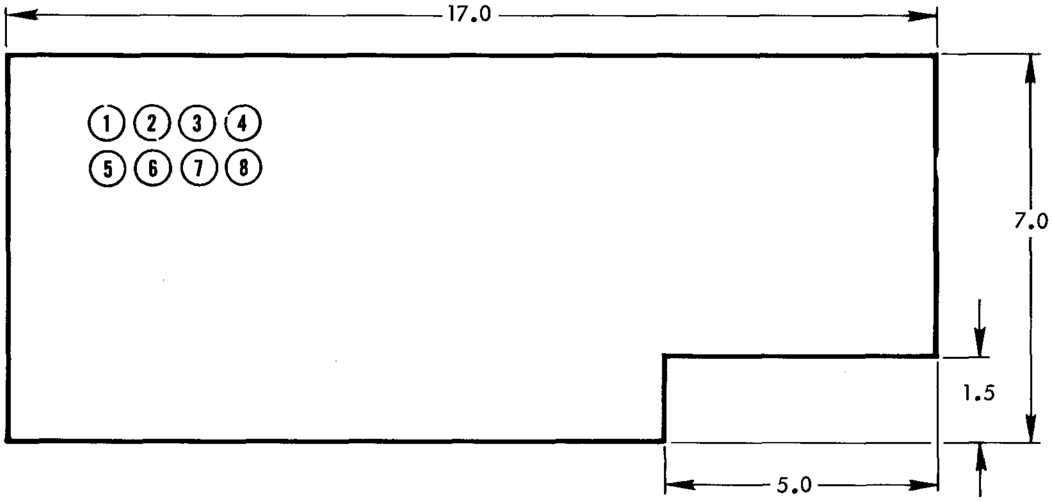
* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

NOTE

2. CRACKS MAY ORIGINATE AT FASTENER HOLES AND PROPAGATE TO ADJACENT FASTENER HOLES.

SMR151-58.3-0-004-2

Figure 58.3-4-6/4. Horizontal Stabilizer Center Section Plates (Sheet 2 of 3)



FILM CUT

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	110	3.5	64.0	250	NDT 45/ NDT 55	SEE CUT	FILM NUMBER SAME AS EXPOSURE NUMBER
E2	↑	↑	↑	↑	↑	↑	
E3							
E4							
E5							
E6							
E7	↓	↓	↓	↓	↓	↓	
E8	110	3.5	64.0	250	NDT 45/ NDT 55	SEE CUT	

SMR151-58.3-0-004-3

Figure 58.3-4-6/4. Horizontal Stabilizer Center Section Plates (Sheet 3 of 3)

58.3.6 RUDDER UPPER COUNTERWEIGHT REGION.

58.3.6.1 DESCRIPTION. (See Figure 58.3-5-6/4.) The rudder is an aluminum alloy structure consisting of front and rear beam, ribs and unstiffened skins. The rudder is attached to the rear beam of the vertical stabilizer. The depleted uranium counterweight is located at the upper end of the rudder. The rudder skins and front beam are made from 7075-T6 aluminum. The formed ribs that the depleted uranium is attached to are fabricated from 2014-T6 aluminum forgings. The exterior surface finish will conform to customer peculiar paint scheme.

58.3.6.2 DEFECTS. Cyclic operation of the rudder may cause fatigue cracks in the fastener holes and in the bend radius of the formed ribs.

58.3.6.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

WARNING

Ensure that power is isolated from all systems in the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane: Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

c. Access: The inspection area is accessible with the use of appropriate maintenance stands.

d. Preparation of part: No special preparation is required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.3.6.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

A. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, General Purpose, 1/8-inch diameter, Ideal Specialty Co. P/N 6100-1/8-S-BNC or equivalent.

(3) Calibration Standard, Aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure with the following additional requirements.

(1) Remove the rudder.

(2) Disassemble the rudder as required for access to the area of the suspect defect.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

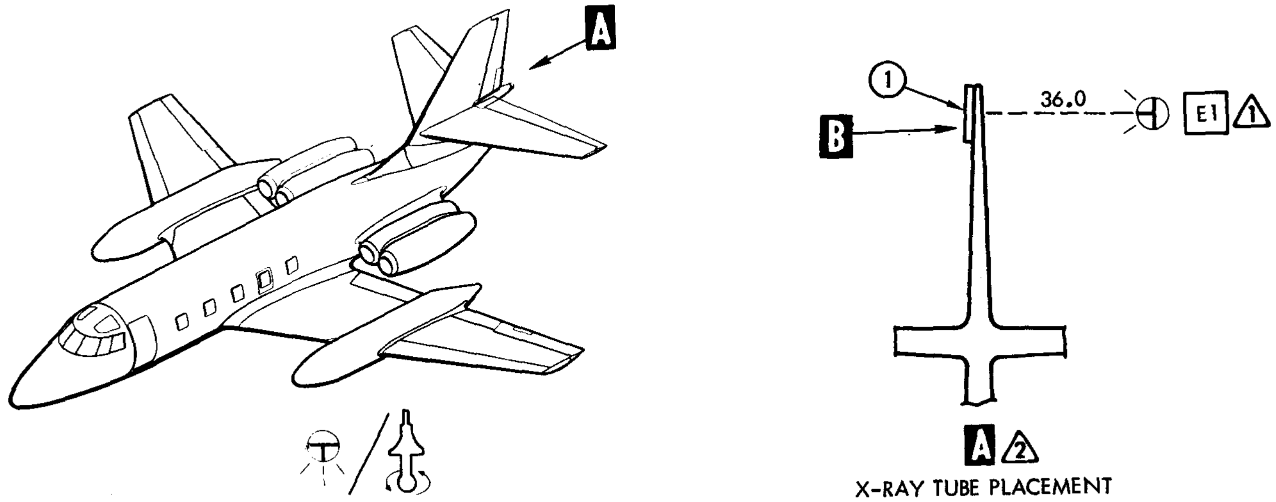
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the surface of the suspected defective area. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

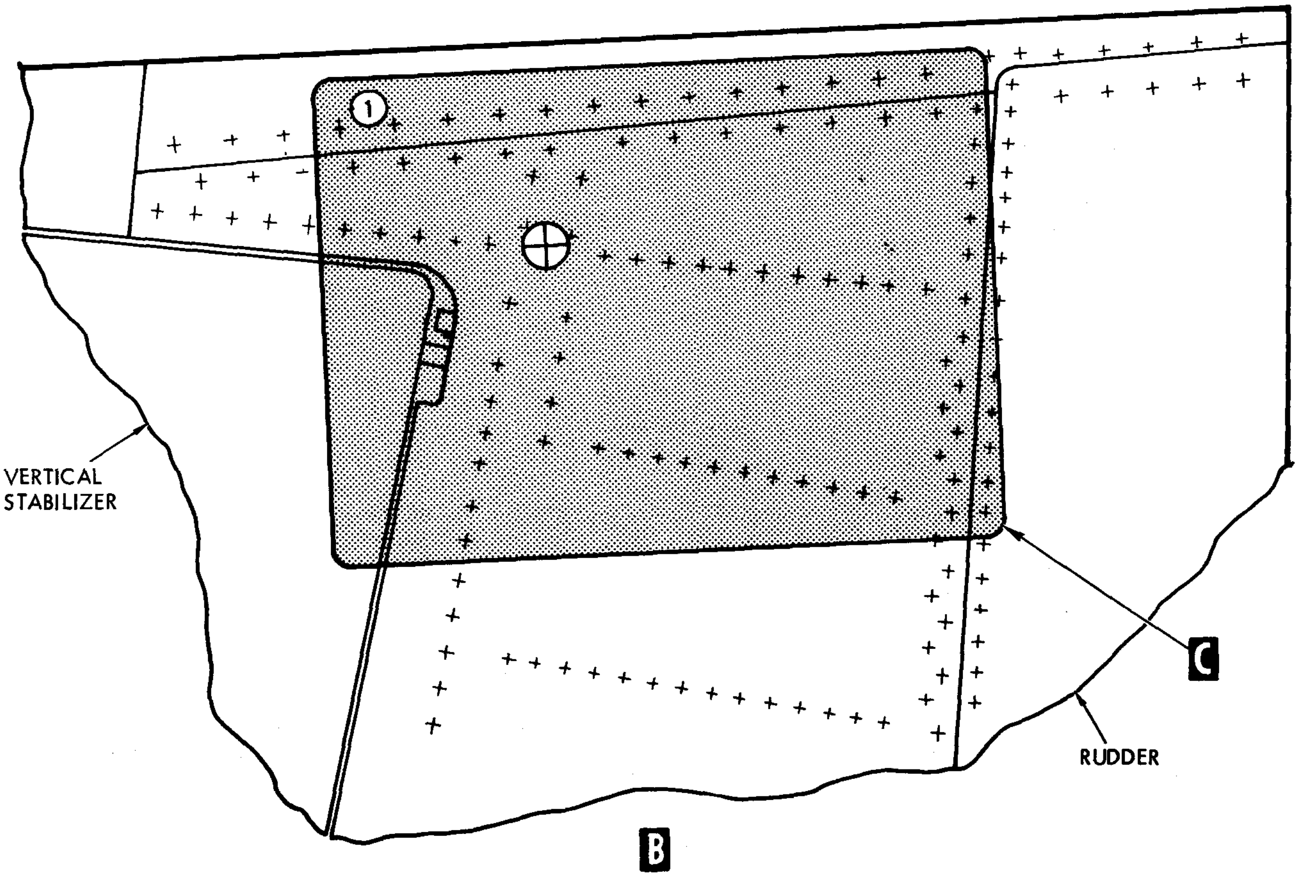
58.3.6.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



X-RAY TUBE PLACEMENT

NOTE

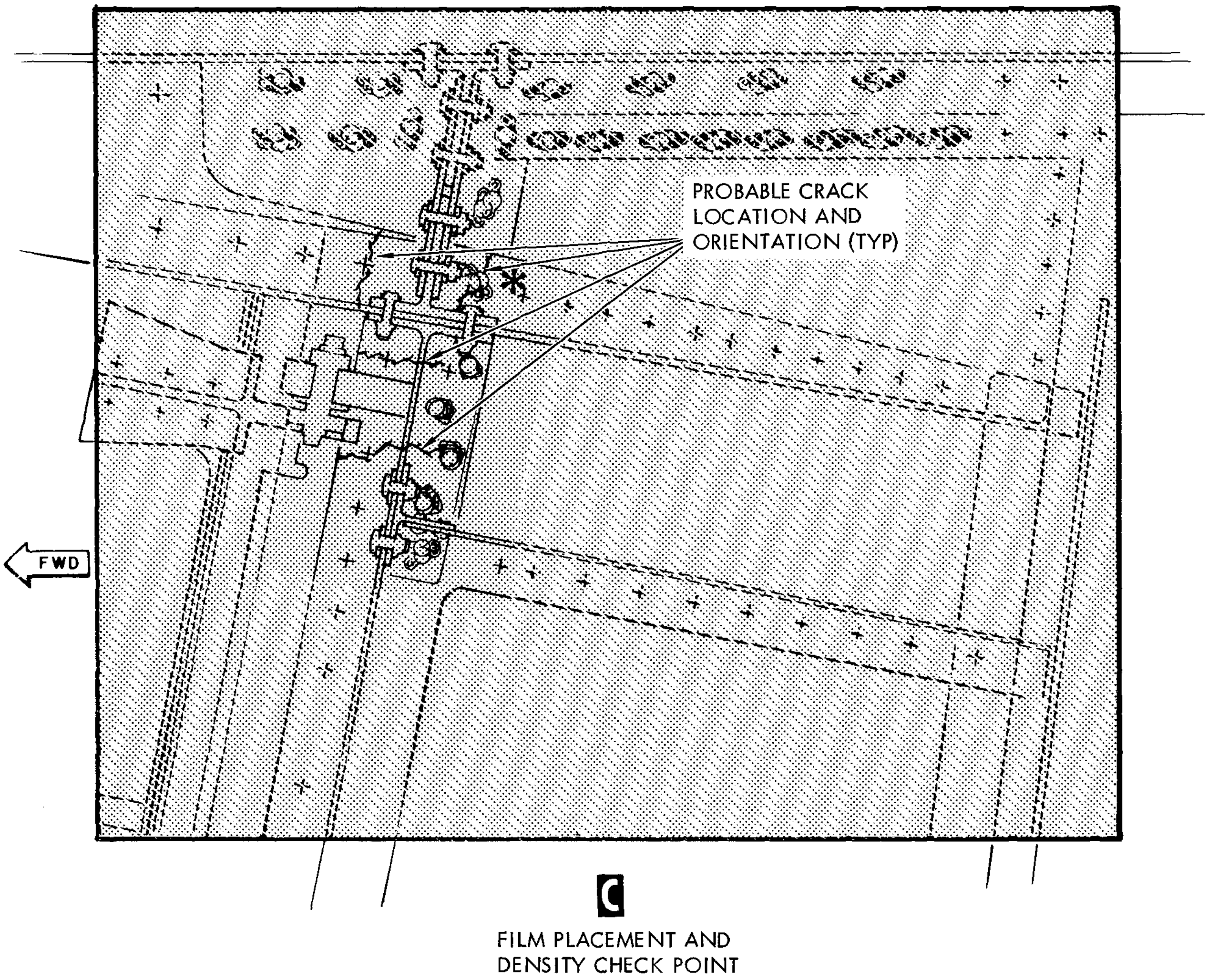
1. POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO FILM.
2. X-RAY TUBE AND FILM CAN BE SHIFTED TO OPPOSITE SIDES OF RUDDER IF DESIRED.



FILM PLACEMENT AND AIMING POINT

SMR151-58.3-0-005-1

Figure 58.3-5-6/4. Rudder Upper Counterweight Region (Sheet 1 of 2)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
FWD
C
FILM PLACEMENT AND DENSITY CHECK POINT

* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	75	5	36.0	120	NDT 45 NDT 55	14 X 17	EXPOSE ①

SMR151-58.3-0-005-2

Figure 58.3-5-6/4. Rudder Upper Counterweight Region (Sheet 2 of 2)

58.3.7 VERTICAL STABILIZER ACTUATOR ATTACH FITTING ATTACHMENT.

58.3.7.1 DESCRIPTION. (See figure 58.3-6-4/0.) The pitch trim system includes a dual actuator assembly that pivots the entire empennage to provide nose-up or nose-down trim of the airplane. The upper end of the actuator assembly is attached to fittings on the lower corners of the vertical stabilizer front beam. The upper attach fittings are fabricated from 8630 steel forgings and are cadmium plated. The front corner has the beam cap, doubler, and skin which are fabricated from 7075-T6.

58.3.7.2 DEFECTS. Fatigue cracks may occur in the actuator attach fitting, beam cap, doubler and skin. Cracks will initiate in the fastener hole and propagate outward in random direction.

58.3.7.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1/4-inch diameter, VM Products P/N VM101B, or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

(4) Calibration Standard, steel. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with the appropriate sections in this manual.

(2) Remove the vertical stabilizer lower leading edge fairing, refer to Section 52.3 of this manual for access panel locations.

c. Access: The attachment fasteners are accessible with the vertical stabilizer lower leading edge fairing removed.

d. Preparation of part:

(1) Remove the two attachment fasteners left and right, common to the upper end of the actuator attach fitting.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using the bolt hole probe and the instrument calibrated for steel, scan the entire inner surface of the two bolt holes in the actuator attach fitting, left and right, from the inner edge of the attach fitting to the first interface only.

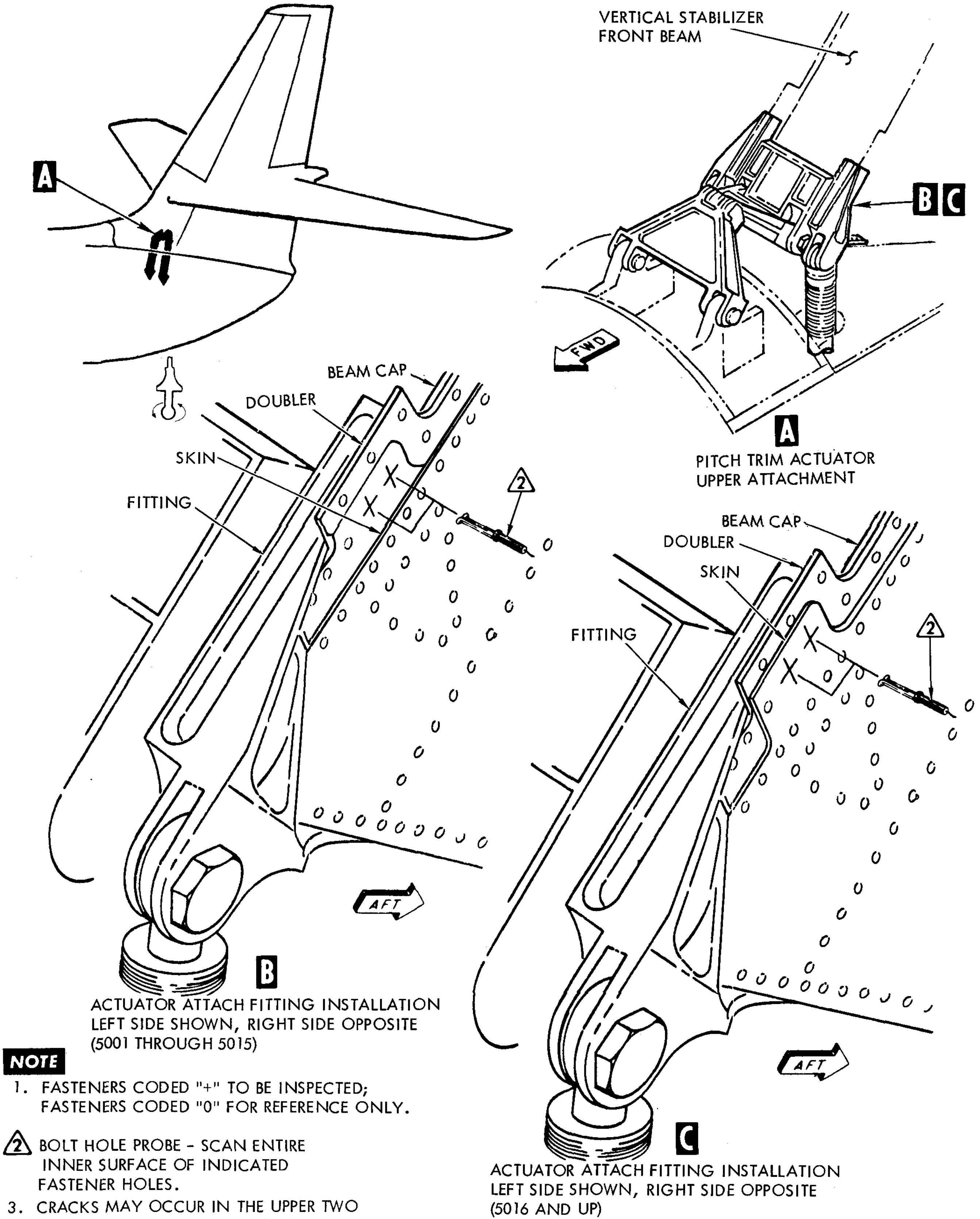
(2) Recalibrate the instrument for aluminum and scan the entire inner surface of the two bolt holes from the first interface to the outboard end of the holes.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

58.3.7.4 BACKUP NDE PROCEDURE. None required.

58.3.7.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



ACTUATOR ATTACH FITTING INSTALLATION
LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE
(5001 THROUGH 5015)

ACTUATOR ATTACH FITTING INSTALLATION
LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE
(5016 AND UP)

NOTE

1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.
2. BOLT HOLE PROBE - SCAN ENTIRE INNER SURFACE OF INDICATED FASTENER HOLES.
3. CRACKS MAY OCCUR IN THE UPPER TWO FASTENER HOLES AND PROPAGATE IN A RANDOM DIRECTION.

SMR151-58.3-0-006

Figure 58.3-6-4/0. Vertical Stabilizer Actuator Attach Fitting Attachment

58.3.8 HORIZONTAL STABILIZER CENTER SECTION AFT LOWER ATTACHMENT.

58.3.8.1 DESCRIPTION. (See figure 58.3-7-4/0.) The horizontal stabilizer center section is a box structure which is attached to the lower vertical stabilizer. The horizontal stabilizer aft end fitting, tapered shim and shear tie angle are attached to the doubler and rear beam cap extending up from the lower vertical stabilizer. These parts are fabricated from 7075-T6 and finished with zinc chromate primer.

58.3.8.2] DEFECTS. Inflight loads transmitted between the horizontal stabilizer and the lower vertical stabilizer may cause fatigue cracks in the attachment holes. Crack will propagate in a fore and aft direction.

58.3.8.3] PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE Equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 5/16-inch diameter. V.M. Products P/N VM101B, or equivalent.

(3) Probe, Shielded, V.M. Products P/N VM202-1/8 or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with the appropriate sections in this manual.

(2) Remove the empennage trailing edge fairing installation. Refer to Section 52.3 of this manual for access panel locations.

c. Access: The attachment fasteners are accessible with the fairing removed and with use of maintenance stands or appropriate AGE.

d. Preparation of part:

(1) Remove the two attachment fasteners left and right, common to the lower end of the horizontal stabilizer and the lower vertical stabilizer.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using the bolt hole probe, scan the entire inner surface of the two bolt holes in the lower aft corner of the horizontal stabilizer, left and right.

(2) Using the surface probe, scan the lower rear beam vertical stabilizer around the forward bolt common to the rear beam carry-through structure, left and right side.

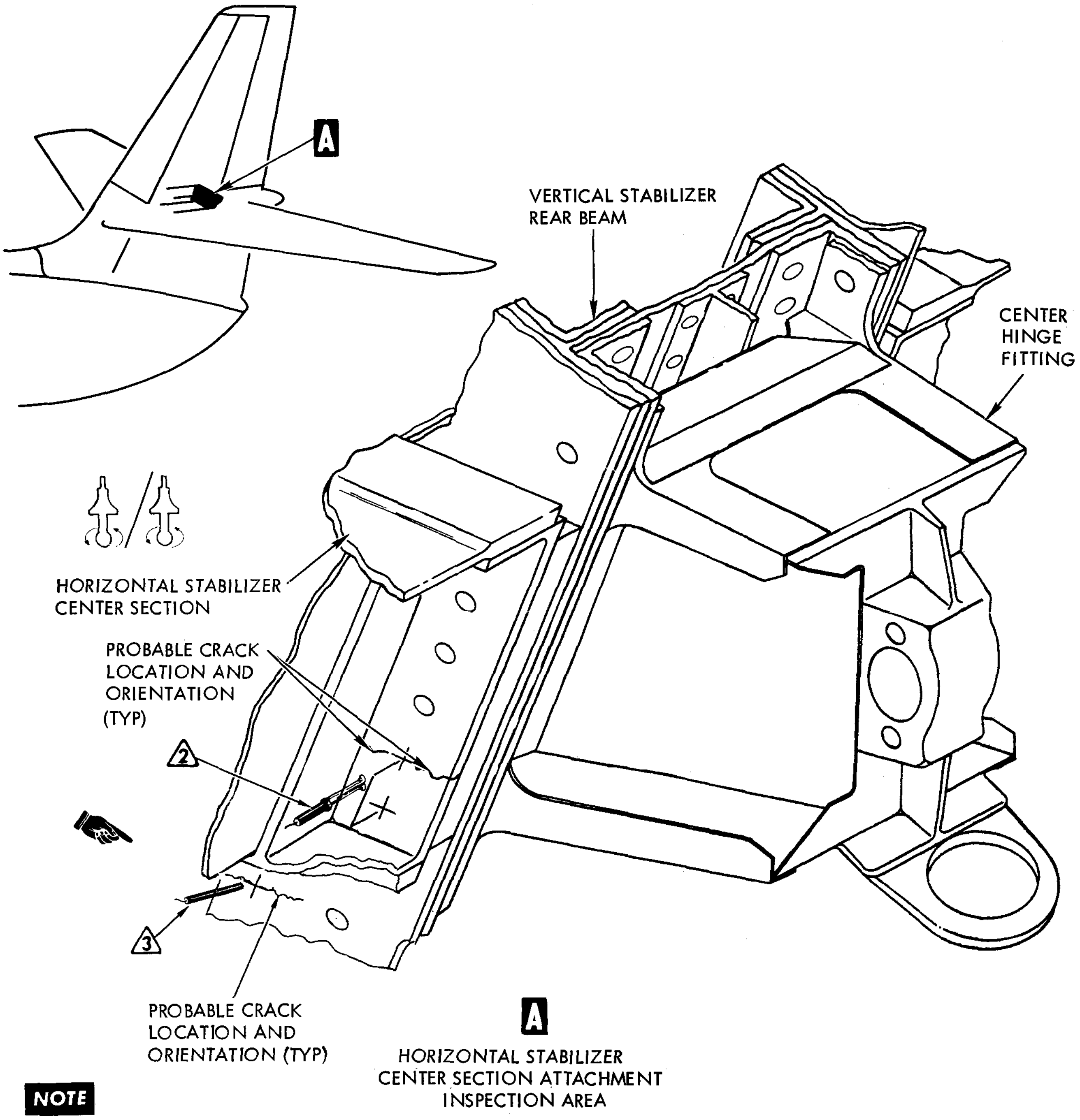
(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.3.8.4 BACKUP NDE PROCEDURE. In order to verify suspected defects, have a different inspector repeat the Primary NDE Procedure, inspecting only the suspect areas.

58.3.8.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


A
VERTICAL STABILIZER REAR BEAM
CENTER HINGE FITTING
HORIZONTAL STABILIZER CENTER SECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
2
3
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
A
HORIZONTAL STABILIZER CENTER SECTION ATTACHMENT INSPECTION AREA

NOTE

1. FASTENER HOLES CODED "+" TO BE INSPECTED; FASTENER HOLES CODED "0" FOR REFERENCE ONLY.

2 BOLT HOLE PROBE - SCAN ENTIRE INNER SURFACE OF EACH INDICATED FASTENER HOLE LEFT AND RIGHT OF CENTER HINGE FITTING.

3 EDDY CURRENT SURFACE SCAN AROUND FASTENER IN BOTH DIRECTIONS, LEFT AND RIGHT OF CENTER HINGE FITTING.

SMR151-58.3-11-007

Figure 58.3-7-4/0. Horizontal Stabilizer Center Section Aft Lower Attachment

58.3.9 EMPENNAGE PIVOT FITTING TO VERTICAL STABILIZER REAR BEAM ATTACHMENT.

58.3.9.1 DESCRIPTION. (See figure 58.3-8-4/0.) The vertical stabilizer rear beam assembly has a doubler and beam cap attached to each side of the empennage pivot fitting. This assembly is attached to the lower aft of the vertical stabilizer to provide a stationary pivot point for the empennage. The doubler and cap are fabricated from 7075-T6 aluminum and the pivot fitting is machined from 2014-T6 aluminum forging. These parts are finished with zinc chromate primer.

58.3.9.2 DEFECTS. Fatigue cracks may occur in the upper fastener holes common to the doubler, beam cap and pivot fitting and propagate in a random direction.

58.3.9.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 5/16-inch diameter, VM Products P/N VM101B, or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with the appropriate sections in this manual.

(2) Remove the vertical stabilizer lower fairing, refer to Section 52.3 of this manual for access panel locations.

c. Access: The attachment fasteners are accessible with the lower fairing removed and with use of maintenance stands or appropriate AGE.

d. Preparation of part:

(1) Remove the two attachment fasteners, left and right, common to the upper end of the pivot fitting on the lower vertical stabilizer.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

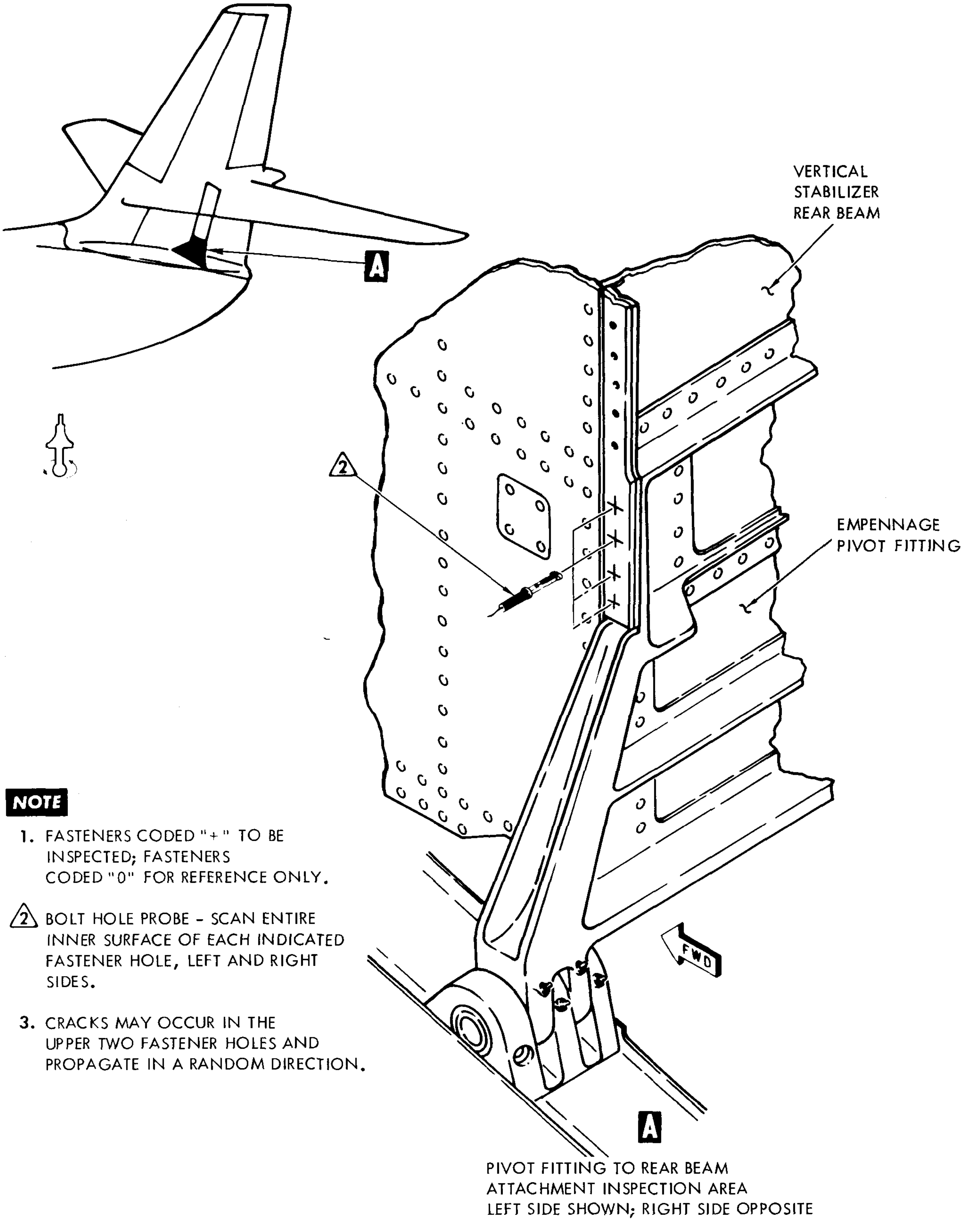
(1) Using the bolt hole probe, scan the entire inner surface of the two bolt holes in the upper end of the pivot fitting on the lower vertical stabilizer.

(2) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

58.3.9.4 BACKUP NDE PROCEDURE. None required.

58.3.9.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

2. BOLT HOLE PROBE - SCAN ENTIRE INNER SURFACE OF EACH INDICATED FASTENER HOLE, LEFT AND RIGHT SIDES.

3. CRACKS MAY OCCUR IN THE UPPER TWO FASTENER HOLES AND PROPAGATE IN A RANDOM DIRECTION.

PIVOT FITTING TO REAR BEAM ATTACHMENT INSPECTION AREA
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

Figure 58.3-8-4/0. Empennage Pivot Fitting to Vertical Stabilizer Rear Beam Attachment

58.3.10 EMPENNAGE PIVOT FITTING.

58.3.10.1 DESCRIPTION. (See figure 58.3-9-4/4) The vertical stabilizer rear beam pivot fitting is attached, through a doubler and spar cap, to the lower aft end of the vertical stabilizer. The pivot fitting is machined from 2014-T6 or 7049-T73 aluminum forging. The pivot fitting is finished with zinc chromate primer.

58.3.10.2 DEFECTS. Fatigue cracks may occur in the radius of the pivot fitting immediately below the lower end of the spar cap, left and/or right outboard side. The fatigue crack propagates across the outboard end of the pivot fitting, and then propagates inboard.

Note

If crack is found, perform visual inspection of the lower twelve (12) inches of the JE-22-2L/R vertical stabilizer rear beam caps; also, perform inspection in accordance with paragraph 58.3.8.3.f.(2).

58.3.10.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, surface, shielded, 1/8-inch diameter, VM Products P/N VM200-3, or equivalent.

(3) Probe, surface, right angle, shield, 1/8-inch diameter, VM Products P/N VM202-6X1/2.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with the appropriate sections in this manual.

(2) Remove the vertical stabilizer lower fairing, refer to Section 52.3 of this manual for access panel locations.

c. Access: The pivot fitting is accessible with the vertical stabilizer lower fairing removed and with use of maintenance stands or appropriate AGE.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the surface area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Surface scan the outboard and aft side of the hinge fitting both left and right side. Scan all accessible surfaces from the lower end of the spar cap to approximately two inches below the spar cap. Scan in a vertical direction. Refer to Section 58.1 of the HOMI for eddy current surface scanning techniques. A sharp meter deflection is noted during calibration will indicate a probable crack.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.3.10.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment: Same as Primary NDE Procedure.

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part: Same as Primary NDE Procedure.

e. Instrument settings/calibration: Same as Primary NDE Procedure.

f. Inspection: In order to verify suspect defects, repeat Primary NDE Procedure using one of the methods outlined below.

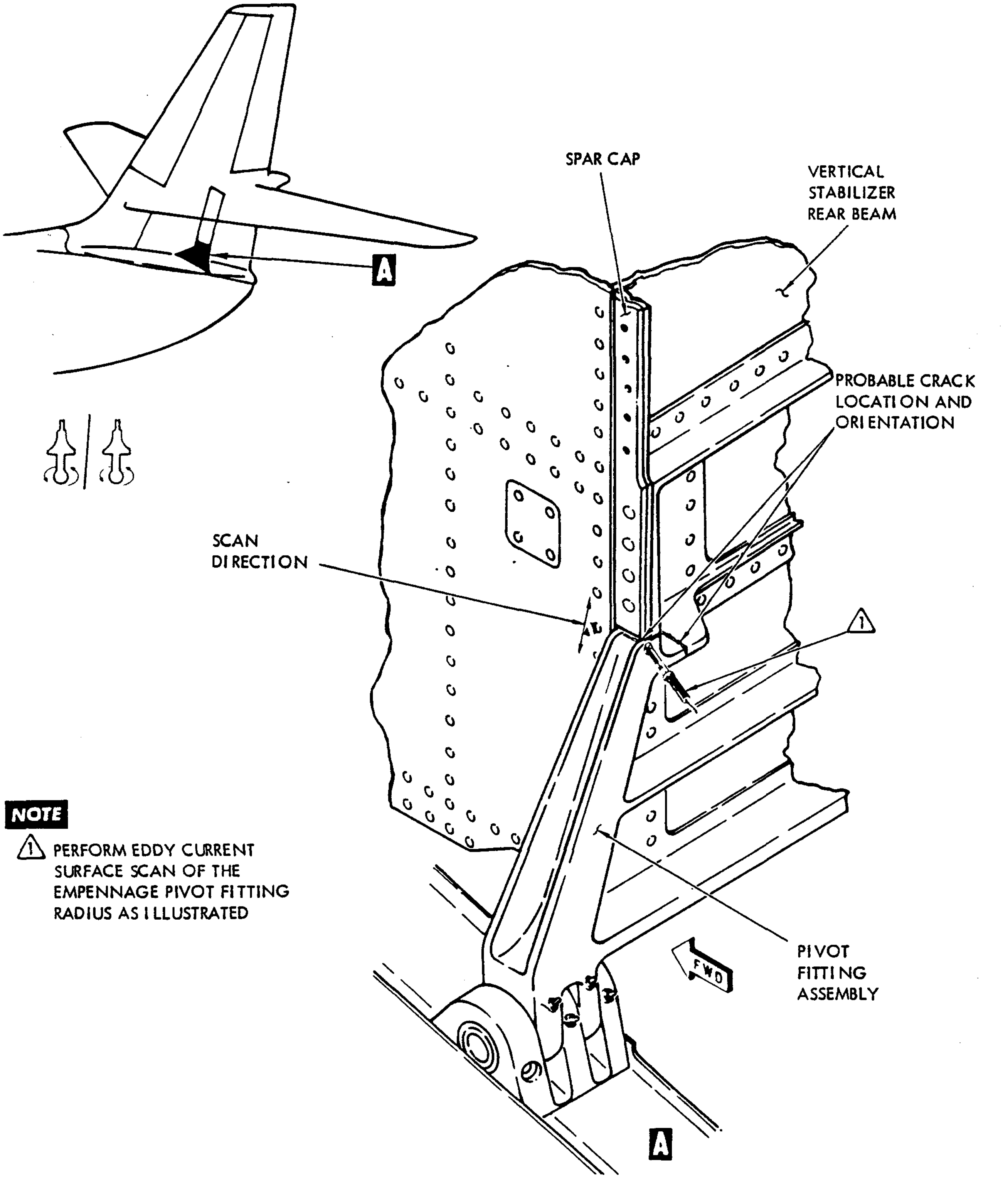
   (1) Preferred Method: Different inspector using different eddy current equipment.

   (2) Alternate Method No. 1: Different inspector using the same eddy current equipment.

   (3) Alternate Method No. 2: The same inspector using the same eddy current equipment after a one-hour delay.

g. Mark and report indicated defects.

58.3.10.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


A
SPAR CAP
VERTICAL STABILIZER REAR BEAM
PROBABLE CRACK LOCATION AND ORIENTATION
SCAN DIRECTION
1
PIVOT FITTING ASSEMBLY
FWD
A
PIVOT FITTING TO REAR BEAM ATTACHMENT INSPECTION AREA
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

NOTE

1 PERFORM EDDY CURRENT SURFACE SCAN OF THE EMPENNAGE PIVOT FITTING RADIUS AS ILLUSTRATED

JDM1270-0-001

Figure 58.3-9-4/4 Empennage Pivot Fitting

SECTION 58.4

FUSELAGE GROUP

Contents Page

GENERAL DESCRIPTION 58-129

Inspection Procedures 58-129

58.4.1 GENERAL DESCRIPTION.

The fuselage group consists of the nose section (Fus. Sta. 94 to 158), the forward section (Fus. Sta. 158 to 270), the center section (Fus. Sta. 270 to 570), and the aft section (Fus. Sta. 570 to 800). See Chapter 53 for complete description of the fuselage structure and detail components.

58.4.1.1 INSPECTION PROCEDURES. Fuselage Group areas and components to be inspected are tabulated in the Fuselage Group Inspections Index. (See Table 58.4-1.) Each inspection procedure has been assigned an inspection index number which is cross referenced to the relevant paragraph and page number.

Table 58.4-1. Fuselage Group Inspection Index

INDEX NO.	INSPECTION TITLE	PARAGRAPH NO.	PAGE NO.
F-1	Longeron, MLG Wheel Well.	58.4.2	58-130
F-2	Fuselage Main Frame Machined Cutouts.	58.4.3	58-132
F-3	Fuselage Main Frames and Side Frames.	58.4.4	58-134
F-4	Windshield Posts No. 1, No. 2, and No. 3.	58.4.5	58-137
F-5	Cabin Windows and Frames.	58.4.6	58-143
F-6	Hydraulic Accumulators.	58.4.7	58-153
F-7	Forward Cap Angle, Splice Doubler, and Skin Aft Pressure Bulkhead, Fus Sta 570.	58.4.8	58-155
F-8	Inner Cap and Channel Cap to Forward and Aft Channel Attachment to FS589.5 Frame.	58.4.9	58-160
F-9	Fuselage Main Frame to Pressure Skin Attachment, LBL9.0 to RBL9.0.	58.4.10	58-160C
F-10	Frame Outer Cap at Aft Engine Nacelle Mount, FS608.	58.4.11	58-160F
F-11	Emergency Exit Cutout Upper Corners, FS410 and FS430.	58.4.12	58-160L
F-12	Upper 30-degree Longeron Attachment, FS483 to FS496.	58.4.13	58-160R
F-13	Top Longeron Splice at BLO, FS490 to FS502.	58.4.14	58-160U

58.4.2 LONGERON, MLG WHEEL WELL.

58.4.2.1. DESCRIPTION. (See figure 58.4-1-4/4.) The shelf structure over the main landing gear wheel well contains a longeron member and forms part of the fuselage shell between fuselage stations 490 and 526 at water line 84. The longeron is fabricated from 7075-T6 aluminum alloy and is finished on exposed surfaces with wash primer, zinc chromate primer, and enamel.

58.4.2.2 DEFECTS. Cyclic pressurization of the fuselage shell produces stress which may cause cracks to develop in the outer bend radius of the longeron member. These cracks will propagate in a fore and aft direction.

58.4.2.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, 90-degree, V.M. Products P/N VM202RA-3/8 or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the main gear air bottle from the left MLG wheel well upper forward area in accordance with Section 32.1 of this manual.

c. Access: The inspection area is accessible through the MLG wheel well opening with the airplane in the normal parked condition.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean accessible surface of outer bend radius as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

Balance may need readjusting when inspecting painted surfaces. Check the balance before scanning.

f. Inspection: Inspect all accessible surface of the outer bend radius from fuselage station 497 to 520 by scanning in both directions. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the longeron.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.2.4. BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment: Same as for Primary NDE Procedure.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

d. Preparation of part: Same as for Primary NDE Procedure.

e. Instrument settings/calibration: Same as for Primary NDE Procedure.

f. Inspection: Inspect suspected defective areas only. Refer to Section 58.1 for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the longeron.

g. Mark and report indicated defects.

58.4.2.5. SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

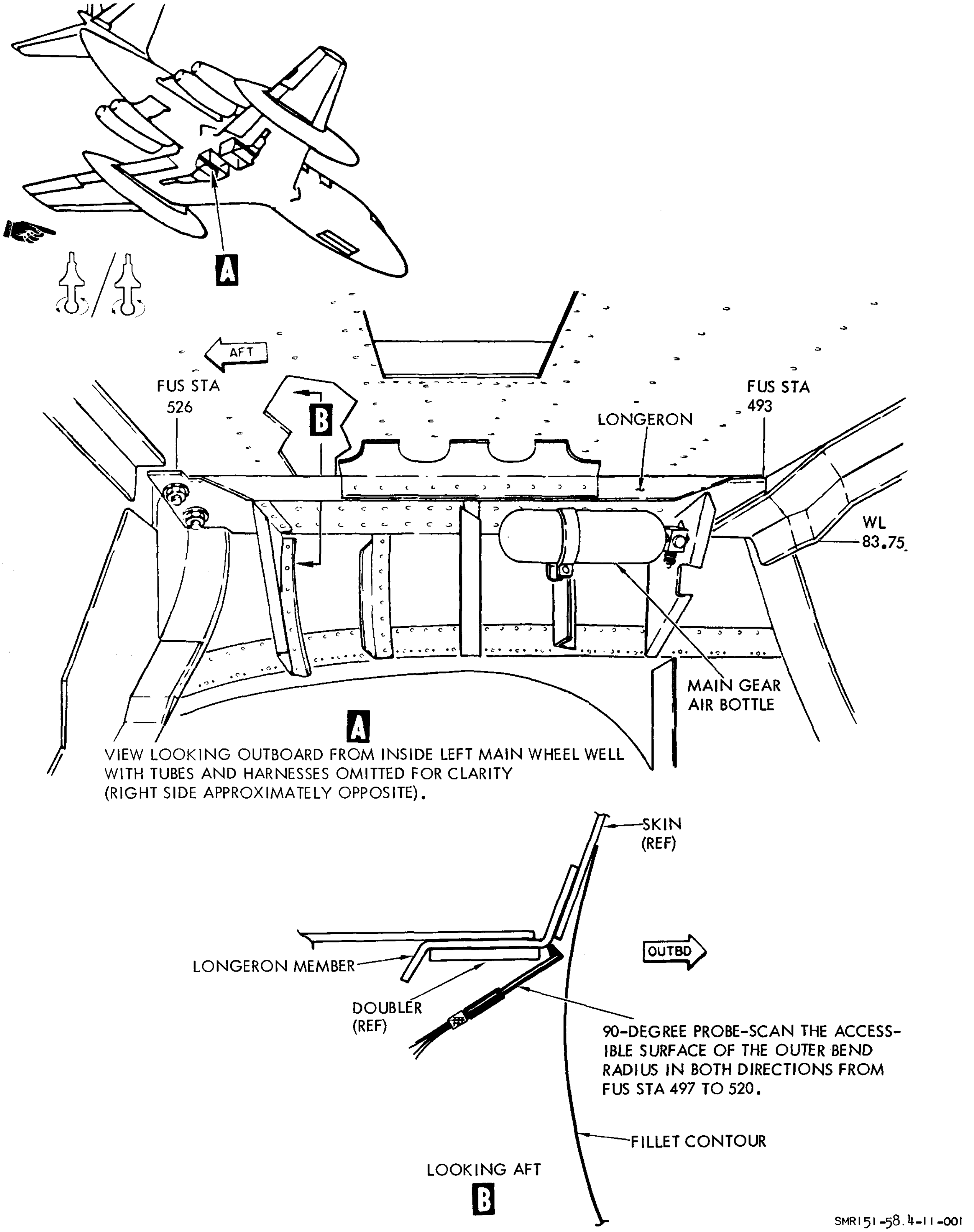

A
AFT
FUS STA 526
B
FUS STA 493
LONGERON
WL 83.75
MAIN GEAR AIR BOTTLE
A
VIEW LOOKING OUTBOARD FROM INSIDE LEFT MAIN WHEEL WELL WITH TUBES AND HARNESSES OMITTED FOR CLARITY (RIGHT SIDE APPROXIMATELY OPPOSITE).
SKIN (REF)
OUTBD
LONGERON MEMBER
DOUBLER (REF)
90-DEGREE PROBE-SCAN THE ACCESSIBLE SURFACE OF THE OUTER BEND RADIUS IN BOTH DIRECTIONS FROM FUS STA 497 TO 520.
FILLET CONTOUR
LOOKING AFT
B
SMR151-58.4-11-001

Figure 58.4-1-4/4. Longeron, MLG Wheel Well

58.4.3 FUSELAGE MAIN FRAME MACHINED CUTOUTS.

58.4.3.1 DESCRIPTION. (See figure 58.4-2-4/2.) The fuselage main frames located at fuselage stations 410, 430, 450, 470, and 490 are fabricated from 7075-T6 aluminum forgings and finished with zinc chromate primer. The frame lower surfaces have one or two machined cutouts at each end. The outboard cutouts provide access to fasteners which were installed to secure the wing mating plane apron to the main frames during aircraft assembly.

58.4.3.2 DEFECTS. The machined cutout surfaces of main frames that have not been shot peened are susceptible to stress corrosion cracking. These cracks will initiate in and propagate along the parting plane.

58.4.3.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Surface, General Purpose, 1/8-inch diameter, Ideal Specialty Co. P/N 6100-1/8-S-BNC or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM102AS or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

> Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove belly fairings as required to provide access to wing joint fitting-to-fuselage frame attachments at fuselage stations 410, 430, 450, 470, and 490. Refer to Section 52.3 of this manual.

c. Access: Access to the inspection areas can be gained while the airplane is in the normal parked condition and with appropriate fairings removed.

d. Preparation of part:

(1) Remove and retain the nuts and washers from each outboard cutout and the fillers from the outboard cutouts at fuselage station 410. Back the screws out from each outboard cutout.

(2) Refer to Section 58.0 for cleaning instructions. Clean surfaces of both machined cutouts at each end of the main frames as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

> The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using either the right angle or general purpose probe, inspect the entire interior surface of each machined cutout. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the frame.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.3.4 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

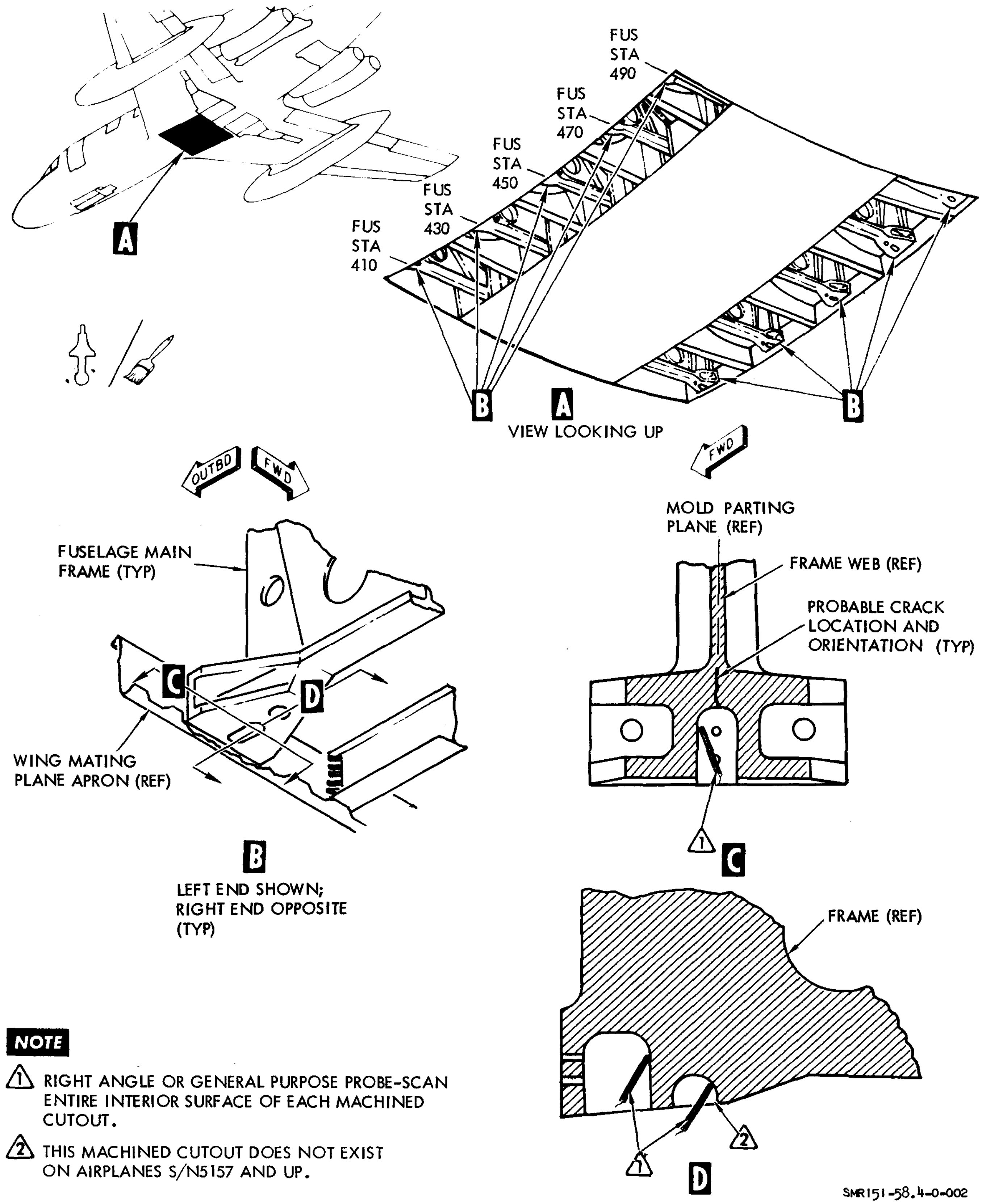
d. Preparation of part: Same as for Primary NDE Procedure, except strip finish and clean suspected defective areas.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. For disposition of indicated defects, refer to Chapter 5 of this manual.

58.4.3.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

⚠1 RIGHT ANGLE OR GENERAL PURPOSE PROBE-SCAN ENTIRE INTERIOR SURFACE OF EACH MACHINED CUTOUT.

⚠2 THIS MACHINED CUTOUT DOES NOT EXIST ON AIRPLANES S/N5157 AND UP.

Figure 58.4-2-4/2. Fuselage Main Frame Machined Cutouts

58.4.4 FUSELAGE MAIN FRAMES AND SIDE FRAMES.

58.4.4.1 DESCRIPTION. (See figure 58.4-3-4/2.) The fuselage main frames and side frames at fuselage stations 410, 430, 450, 470, and 490 are fabricated from 7075-T6 aluminum forgings. The mating surfaces and attachment holes are machined. The remaining surfaces are as forged and finished with zinc chromate primer.

58.4.4.2 DEFECTS. Stress corrosion cracking may occur in the attachment areas of main frames and side frames. Cracks will initiate in the fillet radius and surface of the main frame bathtub fittings and side frame clevis fitting and propagate in a random direction.

58.4.4.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded Pencil, High Sensitivity, VM Products P/N VM100 SS or equivalent.

(3) Probe, Surface, General Purpose; Flexible, 8 inches long, VM Products P/N VM100P-8 or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove belly fairings as required to provide access to the main frame bathtub fittings and side frame clevis fittings at fuselage stations 410, 430, 450, 470, and 490. Refer to Section 52.3 of this manual.

c. Access: Access to the inspection areas is from beneath the airplane through the access openings with the airplane in the normal parked condition.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean local inspection areas of the main frame bathtub fittings and side frame clevis fittings as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Scan the fillet radii and surfaces of the main frame upper and lower bathtub fittings of fuselage stations 410, 430, 450, 470, and 490.

(2) Scan the fillet radii and surfaces of the side frame clevis fittings at fuselage stations 430, 450, 470, and 490.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.4.4 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

d. Preparation of part:

CAUTION

Do not remove more than one upper and one lower wing-to-fuselage attachment bolt at each fuselage station at any time. One upper and one lower attachment bolt at each station must be fully installed at all times. Failure to comply could result in difficulty in reinstalling bolts and/or damage to structure.

CAUTION

Do not allow personnel access to top of wing or fuselage while the attachment bolts are removed. Failure to comply could result in movement between the wing joint fittings and fuselage frames.

Note

Removal of a wing-to-fuselage attachment bolt will provide better access for paint stripping/cleaning prior to inspection.

(1) Remove wing-to-fuselage attachment bolt when suspected defective area is in a main frame bathtub fitting.

58.4.4.4 (Cont).

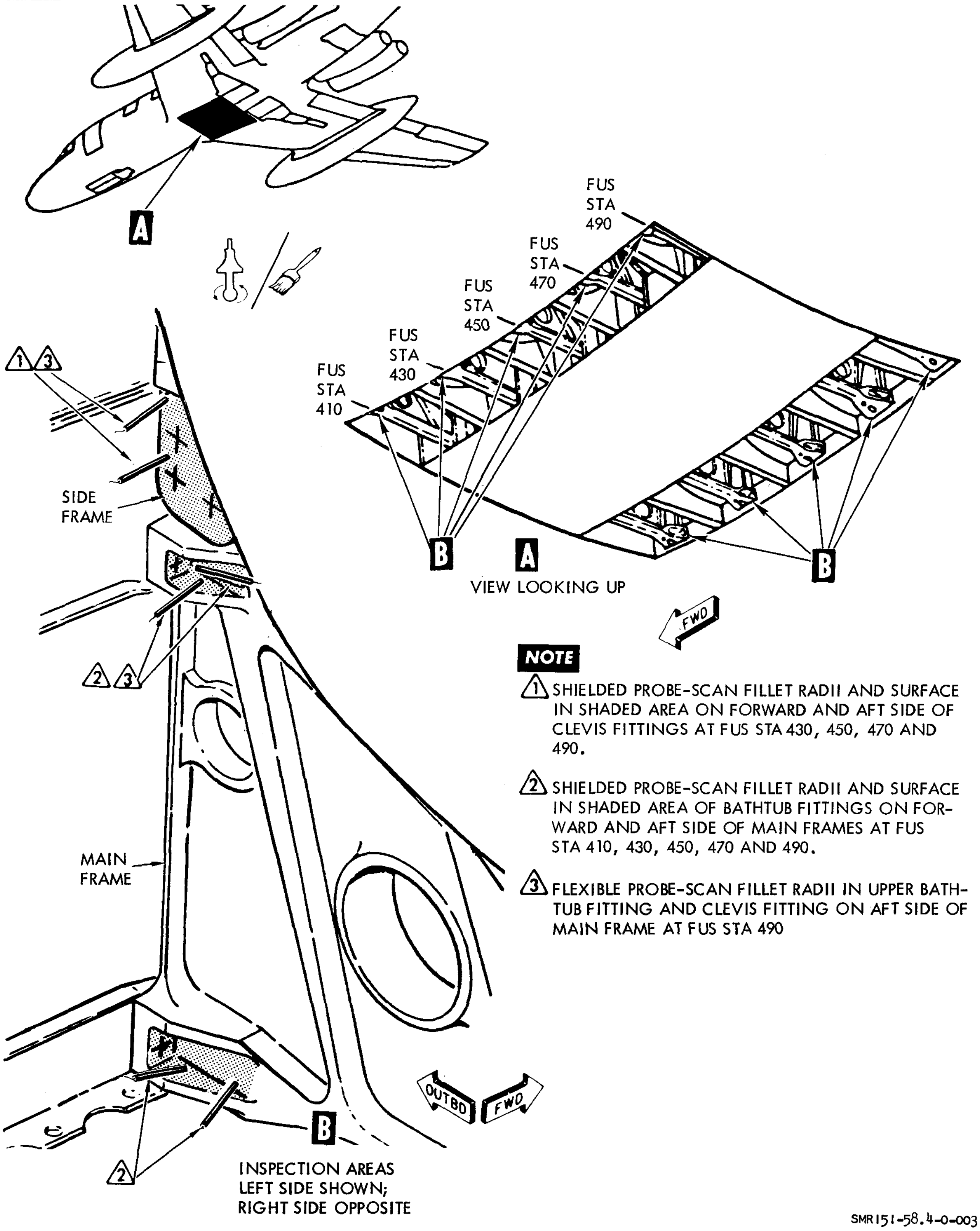
(2) Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective area in the main frame bathtub fitting and/or the side frame clevis fitting.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. For disposition of indicated defects, refer to Chapter 5 in this manual.

58.4.4.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

△1 SHIELDED PROBE-SCAN FILLET RADII AND SURFACE IN SHADED AREA ON FORWARD AND AFT SIDE OF CLEVIS FITTINGS AT FUS STA 430, 450, 470 AND 490.

△2 SHIELDED PROBE-SCAN FILLET RADII AND SURFACE IN SHADED AREA OF BATHTUB FITTINGS ON FORWARD AND AFT SIDE OF MAIN FRAMES AT FUS STA 410, 430, 450, 470 AND 490.

△3 FLEXIBLE PROBE-SCAN FILLET RADII IN UPPER BATHTUB FITTING AND CLEVIS FITTING ON AFT SIDE OF MAIN FRAME AT FUS STA 490

Figure 58.4-3-4/2. Fuselage Main Frames and Side Frames

58.4.5 WINDSHIELD POSTS NO. 1, NO. 2, AND NO. 3.

58.4.5.1 DESCRIPTION. (See figure 58.4-4-6/4.) In the upper section of the forward fuselage the cutout openings for the windshields are made in the outer skin, inner skin and internal doubler. The skins and doublers are fabricated from 2024-T4 aluminum sheet. The inner surfaces are finished with zinc chromate primer and the exterior surface finish will conform to customer peculiar paint scheme.

58.4.5.2 DEFECTS. Fatigue cracks may occur in the rivet or screw holes in the post area between the windshields.

58.4.5.3 PRIMARY NDE PROCEDURE NO. 1 - RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

(1) Open the clear vision windshield. Refer to Section 56.1 for windshield locations.

(2) Remove the interior trim for access to the post for film placement.

(3) Remove the windshield post support fitting from the inboard side of post No. 2 and No. 3. Refer to Figure 53-4, Index No. 43.

c. Access: The inspection areas are accessible from outside the aircraft with the use of appropriate maintenance stands. The interior surface of the windshield posts are accessible from inside the fuselage.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set X-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate X-ray unit and center X-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.5.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520, or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM102AS or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: No special preparation required.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the skin surface as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the ED-520 crack detector.

Note

Eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan around each fastener hole in the skin in No. 1, No. 2, and No. 3 windshield posts on both the left and right side of the airplane. Also scan the forward and aft edge extending the scan to cover the upper and lower radii at each post. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.5.5 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, Ideal Specialty Co. P/N 6200-3/16-BH-BNC or equivalent.

58.4.5.5 (Cont).

(3) Calibration Standard, Aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

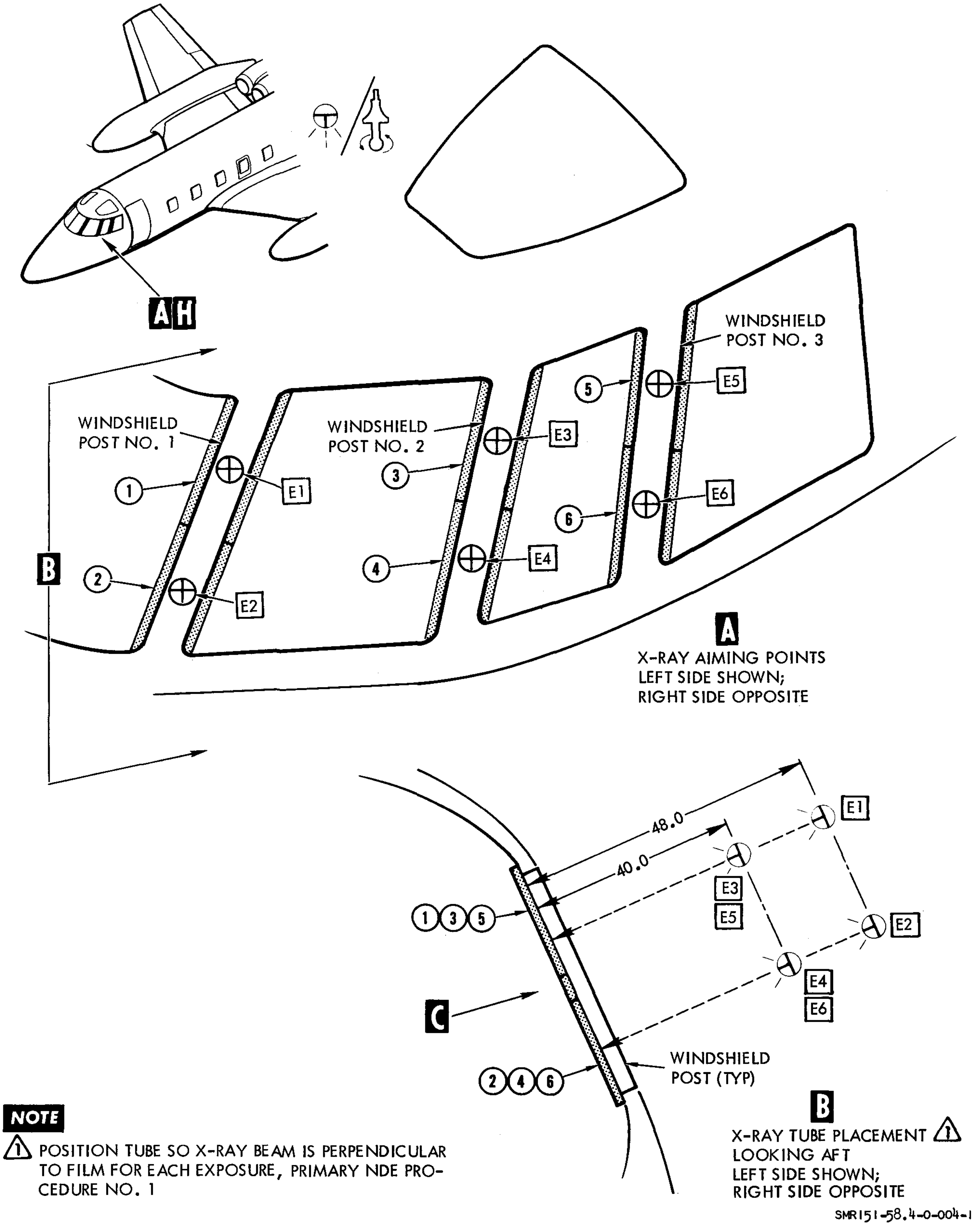
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

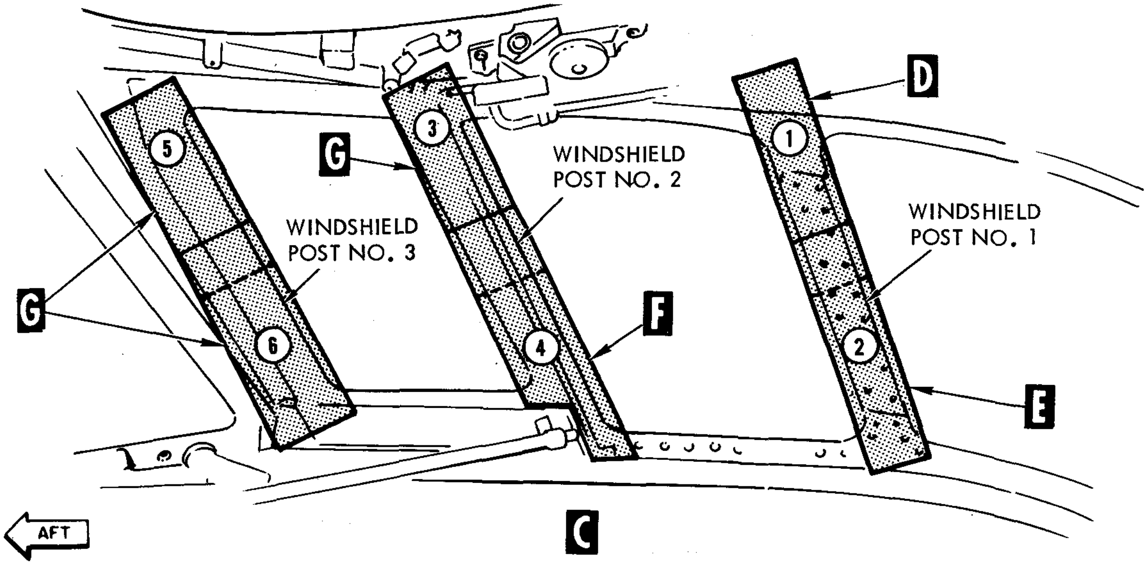
58.4.5.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

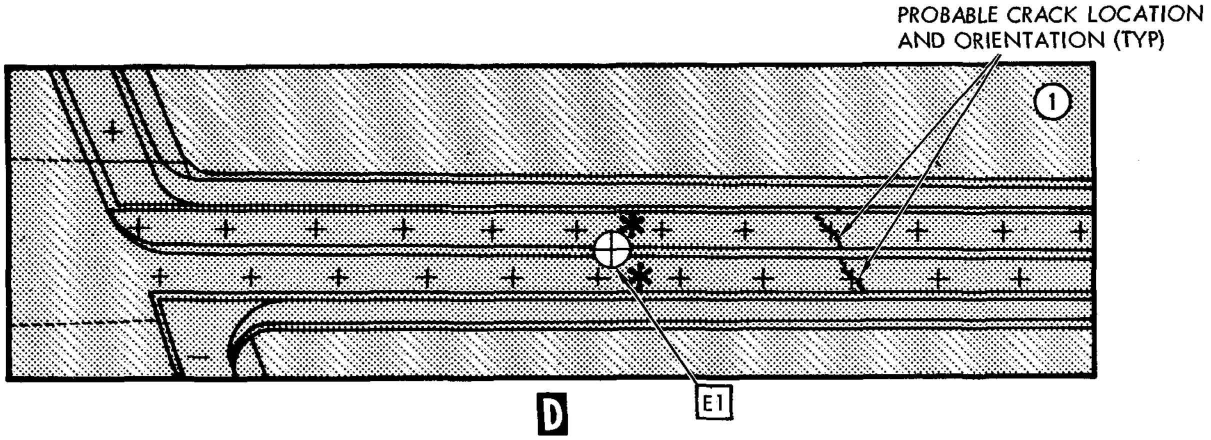
⚠1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO FILM FOR EACH EXPOSURE, PRIMARY NDE PROCEDURE NO. 1

Figure 58.4-4-6/4. Windshield Posts No. 1, No. 2, and No. 3 (Sheet 1 of 4)



C

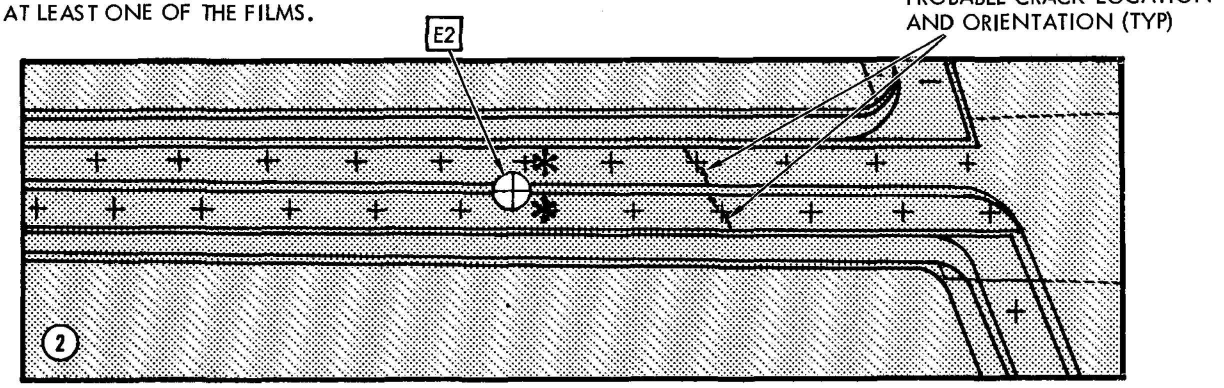
FILM PLACEMENT
LOOKING OUTBOARD
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



D

FILM PLACEMENT, AIMING POINT,
AND DENSITY CHECK POINTS

*CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

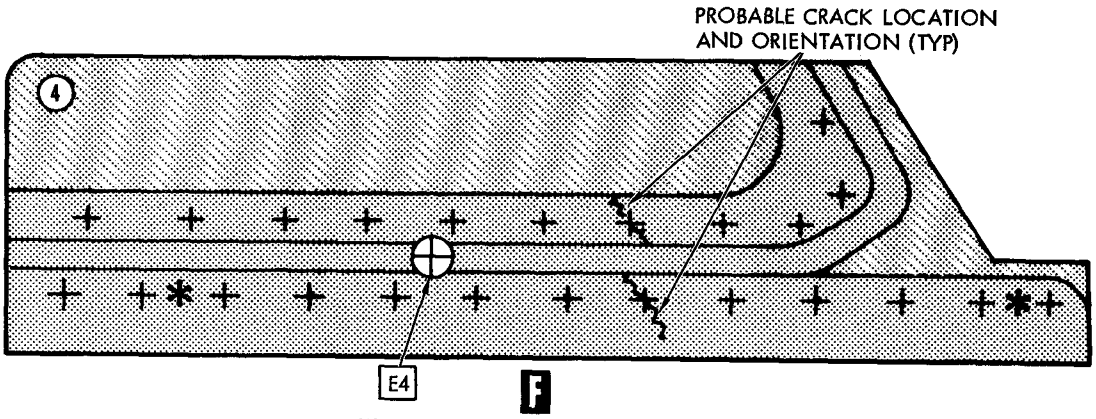


E

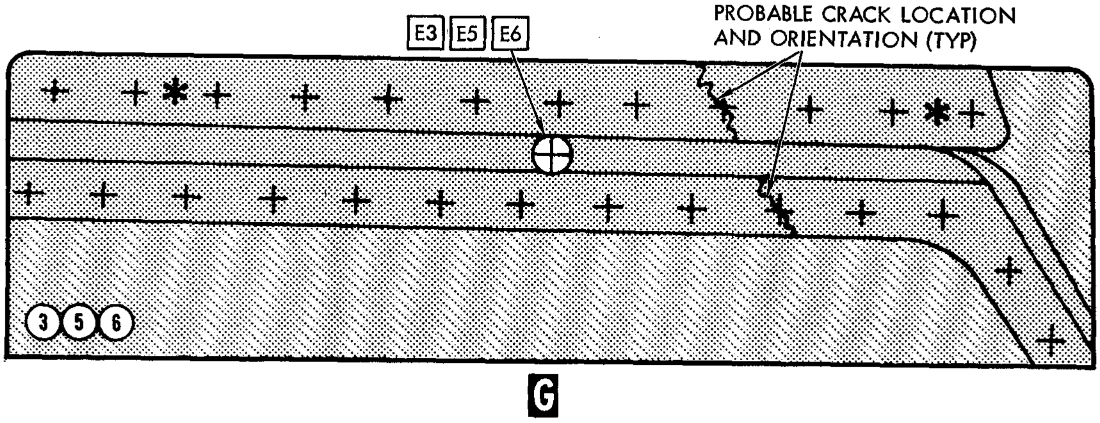
FILM PLACEMENT, AIMING POINT,
AND DENSITY CHECK POINTS

SMR151-58.4-0-004-2

Figure 58.4-4-6/4. Windshield Posts No. 1, No. 2, and No. 3 (Sheet 2 of 4)

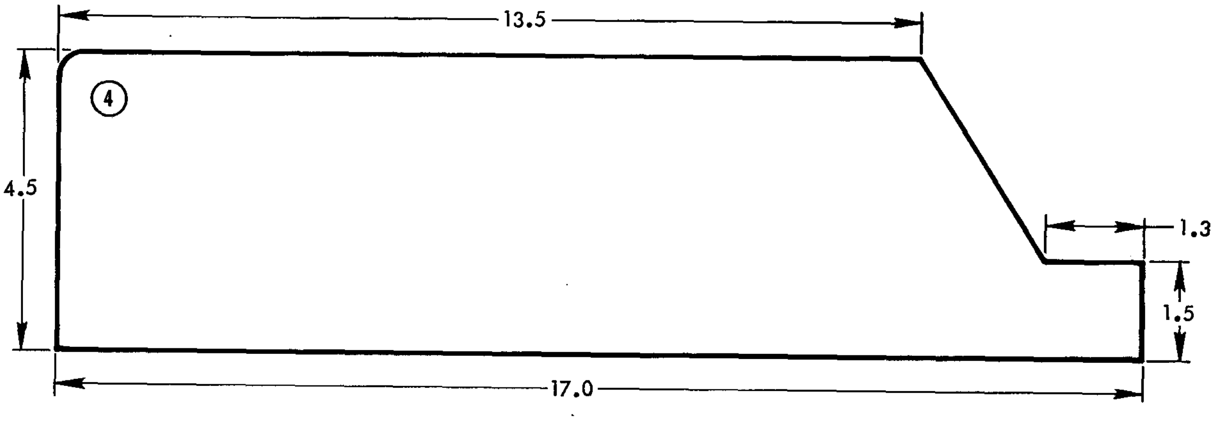


FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS



FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

✱ CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

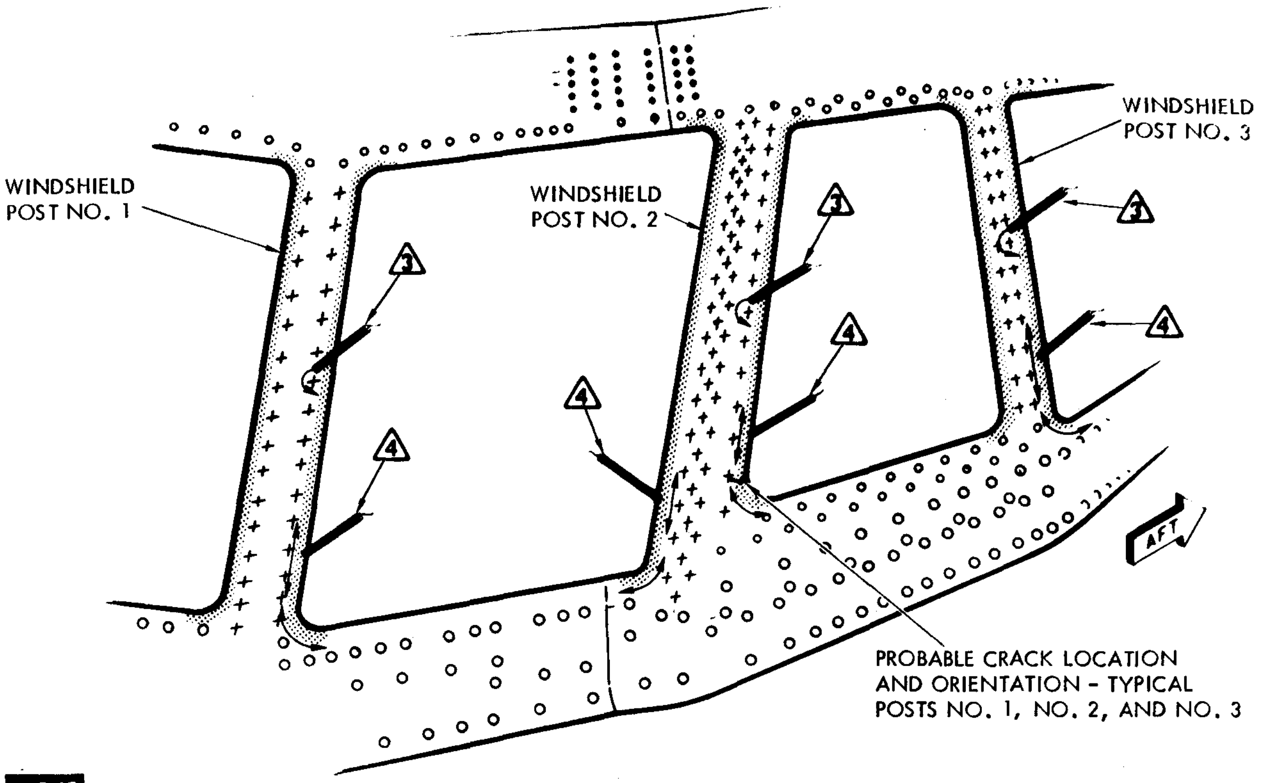


FILM CUT

SMR151-58.4-0-004-3

Figure 58.4-4-6/4. Windshield Posts No. 1, No. 2, and No. 3 (Sheet 3 of 4)

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	125	4.5	48	120	NDT 45/ NDT 55	4.5 X 17	FILM NUMBER SAME EXPOSURE NUMBER
E2	125	4.5	48	120	↑	4.5 X 17	
E3	100	5	40	90		4.5 X 17	
E4	100	5	40	90		SEE CUT	
E5	100	5	40	90	↓	4.5 X 17	
E6	100	5	40	90	NDT 45/ NDT 55	4.5 X 17	


WINDSHIELD POST NO. 1
WINDSHIELD POST NO. 2
WINDSHIELD POST NO. 3
3
4
AFT
PROBABLE CRACK LOCATION AND ORIENTATION - TYPICAL POSTS NO. 1, NO. 2, AND NO. 3

NOTE

2. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "O" FOR REFERENCE ONLY.

△3 SHIELDED PENCIL PROBE-SCAN AROUND INDICATED FASTENERS IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION, PRIMARY NDE PROCEDURE NO. 2.

△4 SHIELDED PENCIL PROBE-SCAN SHADED EDGES OF POSTS (INCLUDING CORNERS) IN BOTH DIRECTIONS, PRIMARY NDE PROCEDURE NO. 2.

H

EDDY CURRENT INSPECTION
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

JS-58-0-58.1

Figure 58.4-4-6/4. Windshield Posts No. 1, No. 2, and No. 3 (Sheet 4 of 4)

58.4.6 CABIN WINDOWS AND FRAMES.

58.4.6.1 DESCRIPTION. (See figure 58.4-5-6/2.) There are five cabin windows on each side of the fuselage. Each cabin window consists of a JF 442 (5001 through 5085) or a JF 764 (5086 and up) plexiglass window panel assembly attached to a 7075-T6 aluminum alloy window frame. The window frame is finished with zinc chromate primer and overcoated with a customer peculiar paint system.

58.4.6.2 DEFECTS. Cracks may occur in the plexiglass window and in the window frame at the attaching fastener holes.

Note

Perform either Primary NDE Procedure No. 1 Radiographic or Both Primary NDE Procedure No. 2 Ultrasonic and Primary NDE Procedure No. 3 Eddy Current.

58.4.6.3 PRIMARY NDE PROCEDURE NO. 1 - RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

(1) Remove the interior upper vertical panels around the windows.

(2) Pull the insulation blankets back from around the window frames.

c. Access: The inspection areas are accessible from outside the airplane with the use of appropriate maintenance stands. The interior surface of the windows is accessible from inside the fuselage.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set X-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

Note

Film Interpretation: Cracks may appear less dense rather than more dense due to foreign material in crack. The crack image would be lighter than surrounding structure instead of darker if foreign material is in the crack.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate X-ray unit and center X-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Primary NDE Procedure No. 2 Ultrasonic for cracks in the plexiglass or Primary NDE Procedure No. 3 Eddy Current for cracks in the frame. If no defect is suspected, proceed to System Securing.

58.4.6.4 PRIMARY NDE PROCEDURE NO. 2 - ULTRASONIC.

a. NDE equipment:

(1) Portable Ultrasonic Flaw Detector, Sonic Mark I or equivalent.

(2) Transducer, 5.0 MHz, 0.250 × 0.250, 70 degree Shear Wave, Mastercraft P/N HMT 7-223 or equivalent.

(3) Cable, 6-foot, Microdot/BNC Connectors, Mastercraft P/N HMT 7-41-BM or equivalent.

(4) Calibration Standard, as illustrated.

(5) Couplant, water.

b. Preparation of airplane: No special preparation required.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the window surfaces as required to permit good contact between part and transducer.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer to the calibration standard in Position No. 1 as illustrated with the front edge of the transducer approximately one-half inch from the step down radius.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT. Adjust sweep to display the signal from the radius near the center of the CRT and the fastener hole signal near the right side of the CRT.

(3) Slide the transducer to Position No. 2 and note that the signal from the radius is near the center of the CRT and the sawcut signal is near the right side of the CRT, immediately to the right of the fastener hole signal.

(4) Maximize the signal from the sawcut by adjusting the transducer position and adjust the signal amplitude to obtain 80 percent saturation on the CRT display.

58.4.6.4 (Cont).

(5) Slide the transducer to Position No. 3 and note the radius signal and the loss of the fastener hole signal on the CRT display.

Note

Review all previous calibration positions to observe the CRT display if sweep is readjusted.

(6) Slide and rotate the transducer to Position No. 4 to set up on the fastener hole.

(7) Observe the fastener hole signal on the right side of the CRT display and note the loss of the radius signal.

(8) Slide the transducer to Position No. 5 and note the sawcut signal to the immediate right of the fastener hole signal.

(9) Slide the transducer to Position No. 6 and note the sawcut signal to the immediate left of the fastener hole signal.

(10) Slide the transducer to Position No. 7 and note the loss of fastener hole and sawcut signal.

(11) Slide and rotate the transducer to Position No. 8, No. 9, No. 10, and No. 11 to observe the CRT display for the sawcut oriented in the opposite 45 degree direction.

(12) Repeat steps (1) thru (11) to observe the transducer locations, fastener hole signals and sawcut signals.

(13) The CRT signal from the sawcuts simulate signals from cracks at the fastener holes.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 of this manual.

f. Inspection:

(1) Apply couplant to part and couple transducer to inspection area as illustrated.

(2) Scan on each side of all the attaching fastener holes around the perimeter of the ten cabin windows.

(3) Any signal received similar to and at approximately the same location on the CRT display as the sawcut signals observed during calibration will indicate a probable crack in the part.

g. Mark and report suspected defects.

h. To confirm suspected defects, remove the window panel assembly and perform a close visual inspection of the suspect defect area. If no defect is suspected proceed to Primary NDE Procedure No. 3. If no additional inspections are required, proceed to System Securing.

58.4.6.5 PRIMARY NDE PROCEDURE NO. 3 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520, or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM102AS or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: No special preparation required.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the frame surface as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the ED-520 crack detector.

Note

Eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan around each fastener hole in the frame enclosing the five cabin windows on both the left and right side of the aircraft. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.6.6 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of aircraft: No special preparation required.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective area.

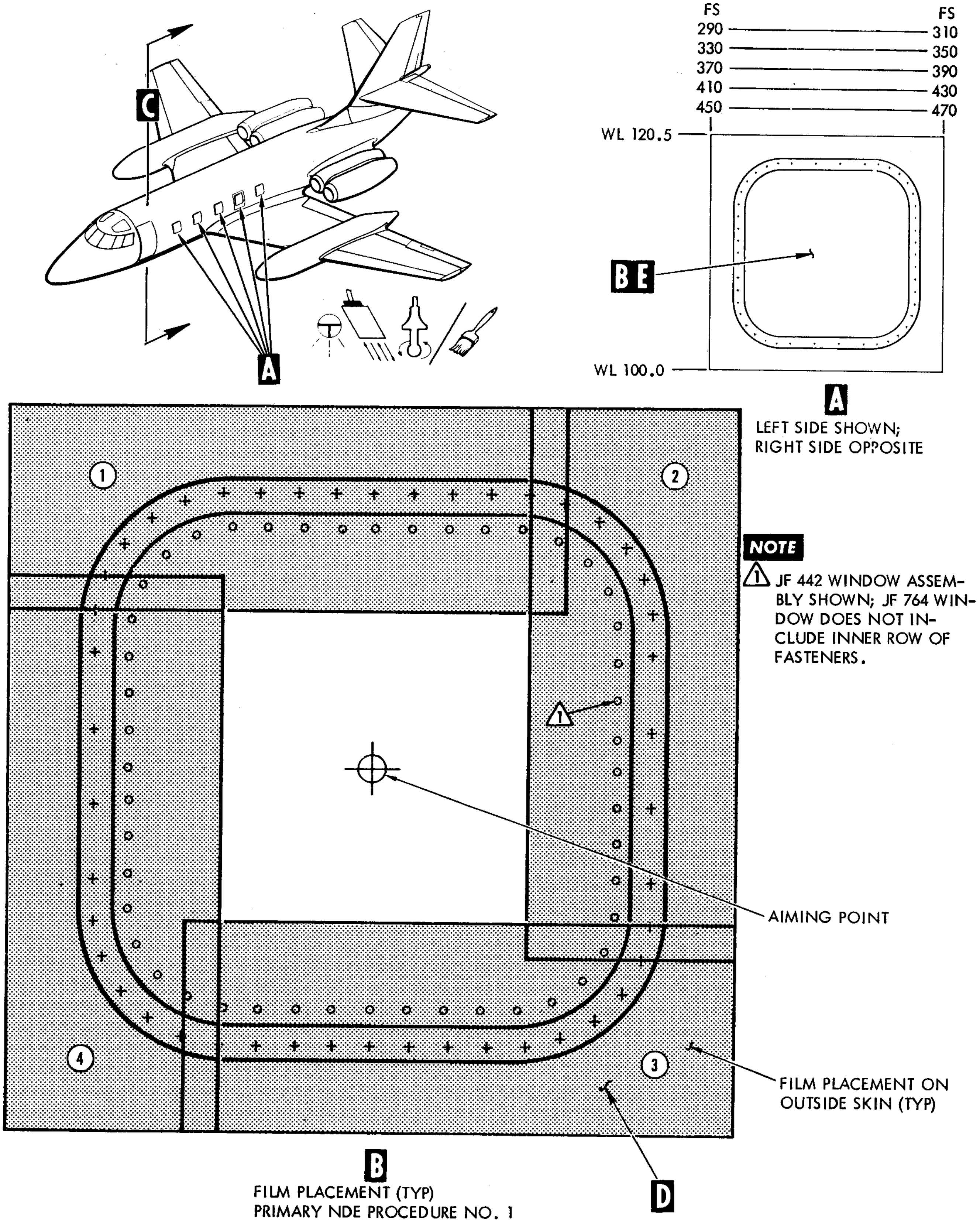
e. Instrument settings/calibration: Not applicable.

58.4.6.6 (Cont).

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.4.6.7 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

NOTE

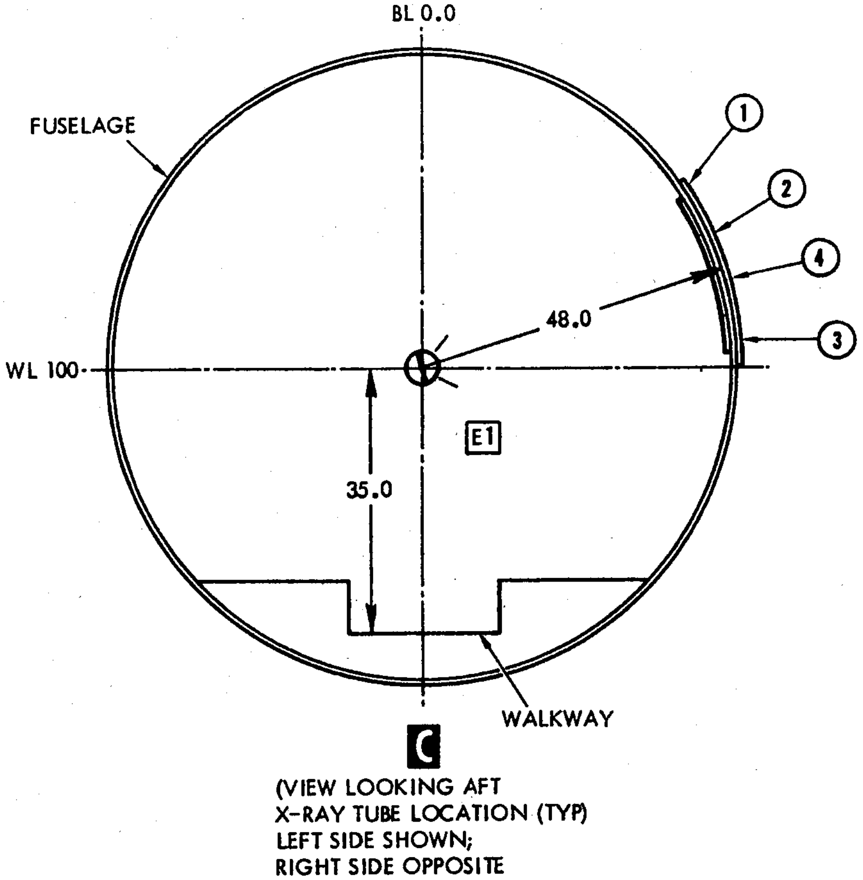
⚠1 JF 442 WINDOW ASSEMBLY SHOWN; JF 764 WINDOW DOES NOT INCLUDE INNER ROW OF FASTENERS.

AIMING POINT

FILM PLACEMENT ON OUTSIDE SKIN (TYP)

FILM PLACEMENT (TYP)
PRIMARY NDE PROCEDURE NO. 1

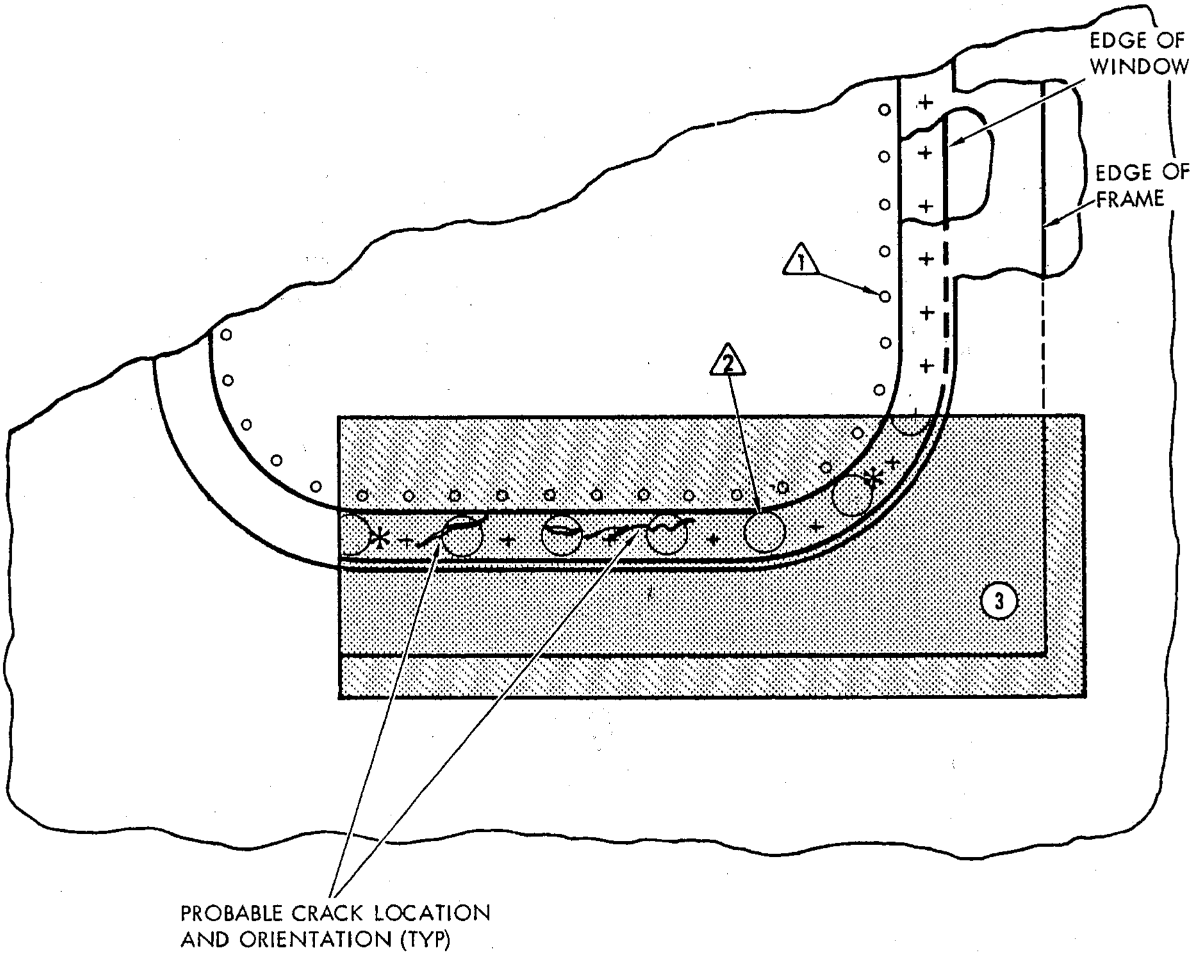
Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 1 of 7)



RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	75	5	48	240	NDT 35 NDT 45	7 X 17	EXPOSE FILM ① ③ ② ④

SMR151-58.4-0-005-2

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 2 of 7)



D

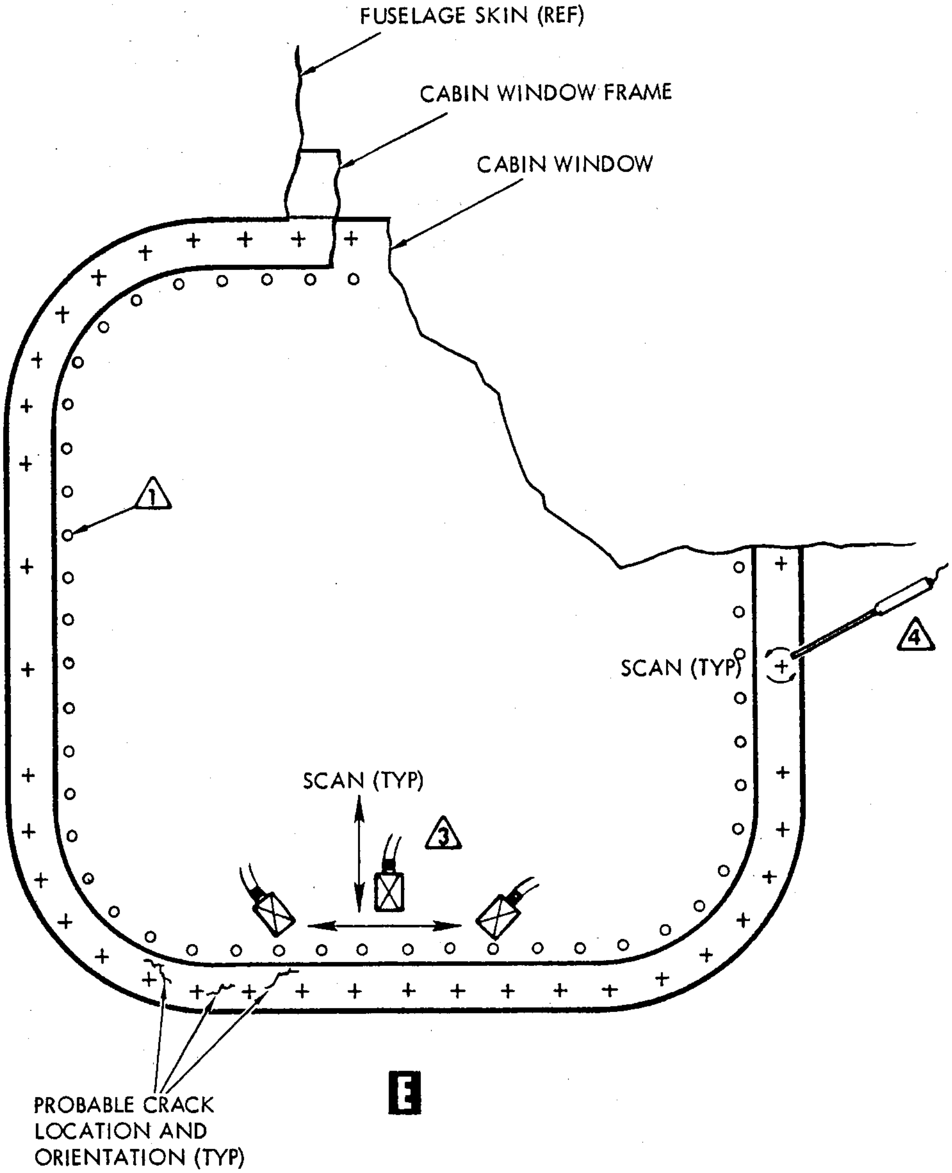
NOTE

1. JF 442 WINDOW ASSEMBLY SHOWN; JF 764 WINDOW DOES NOT INCLUDE INNER ROW OF FASTENERS.
2. LIGHTENING HOLES IN JF 442 WINDOW ASSEMBLY ONLY.

✻ CHECK MEDIAN FILM DENSITY AT NOTED POINTS. WHERE MULTIPLE FILM LOADING IS USED, THE MEDIAN FILM DENSITY IS REQUIRED IN EACH CRITICAL AREA ON AT LEAST ONE OF THESE FILMS.

SMR151-58.4-0-005-3

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 3 of 7)

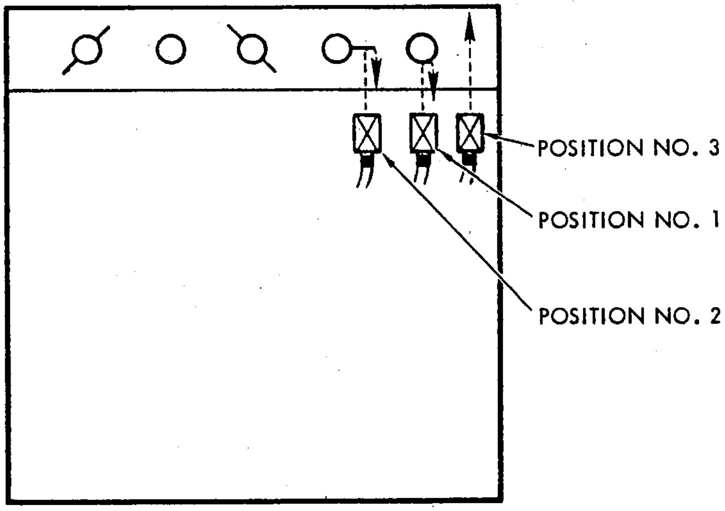


NOTE

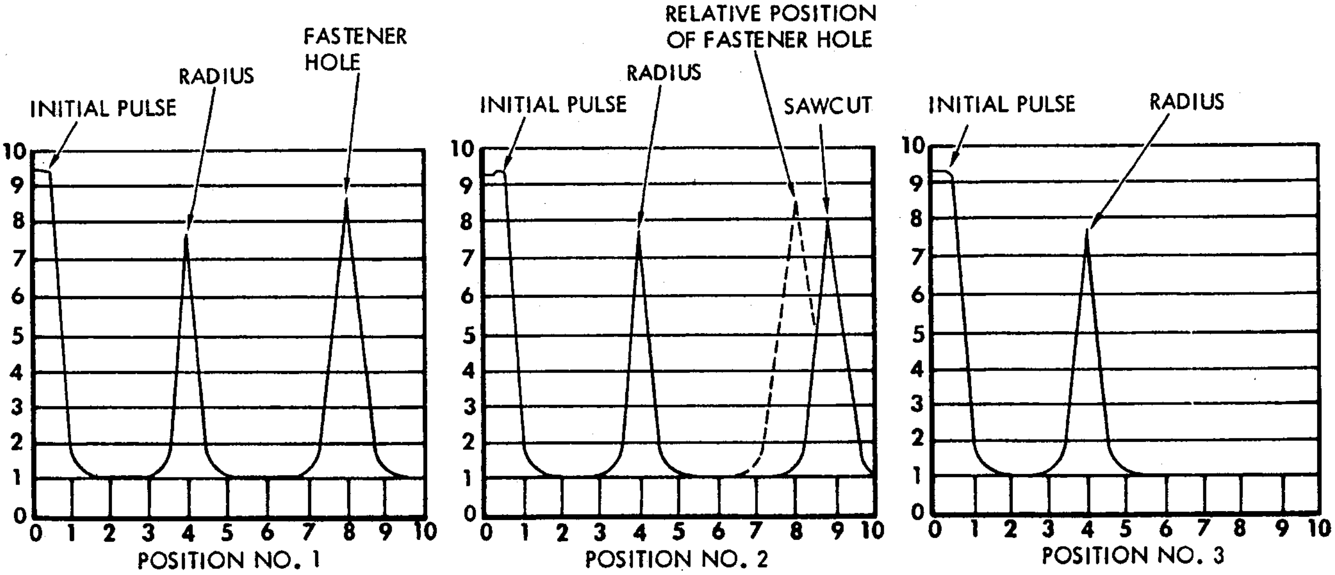
1. JF 442 WINDOW ASSEMBLY SHOWN; JF 764 WINDOW ASSEMBLY DOES NOT INCLUDE INNER ROW OF FASTENERS.
3. TRANSDUCER-SCAN BOTH SIDES OF ALL FASTENERS HOLES IN THE CABIN WINDOW, PRIMARY NDE PROCEDURE NO. 2.
4. PROBE-SCAN AROUND ALL FASTENERS IN THE CABIN WINDOW FRAME, PRIMARY NDE PROCEDURE NO. 3.
5. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "O" FOR REFERENCE ONLY.

SMR151-58.4-0-005-4

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 4 of 7)



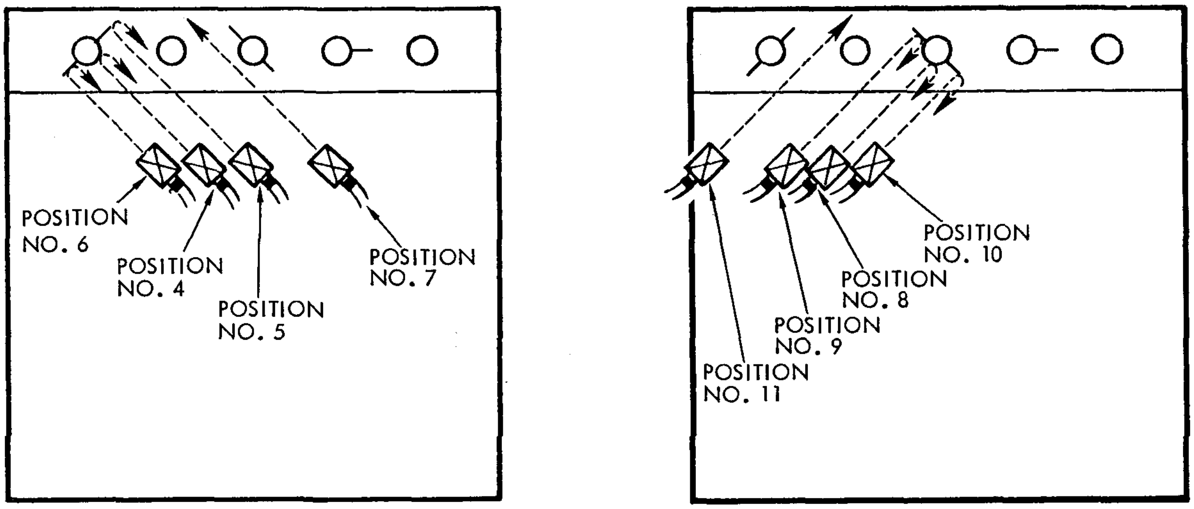
CALIBRATION SETUP PRIMARY
NDE PROCEDURE NO. 2



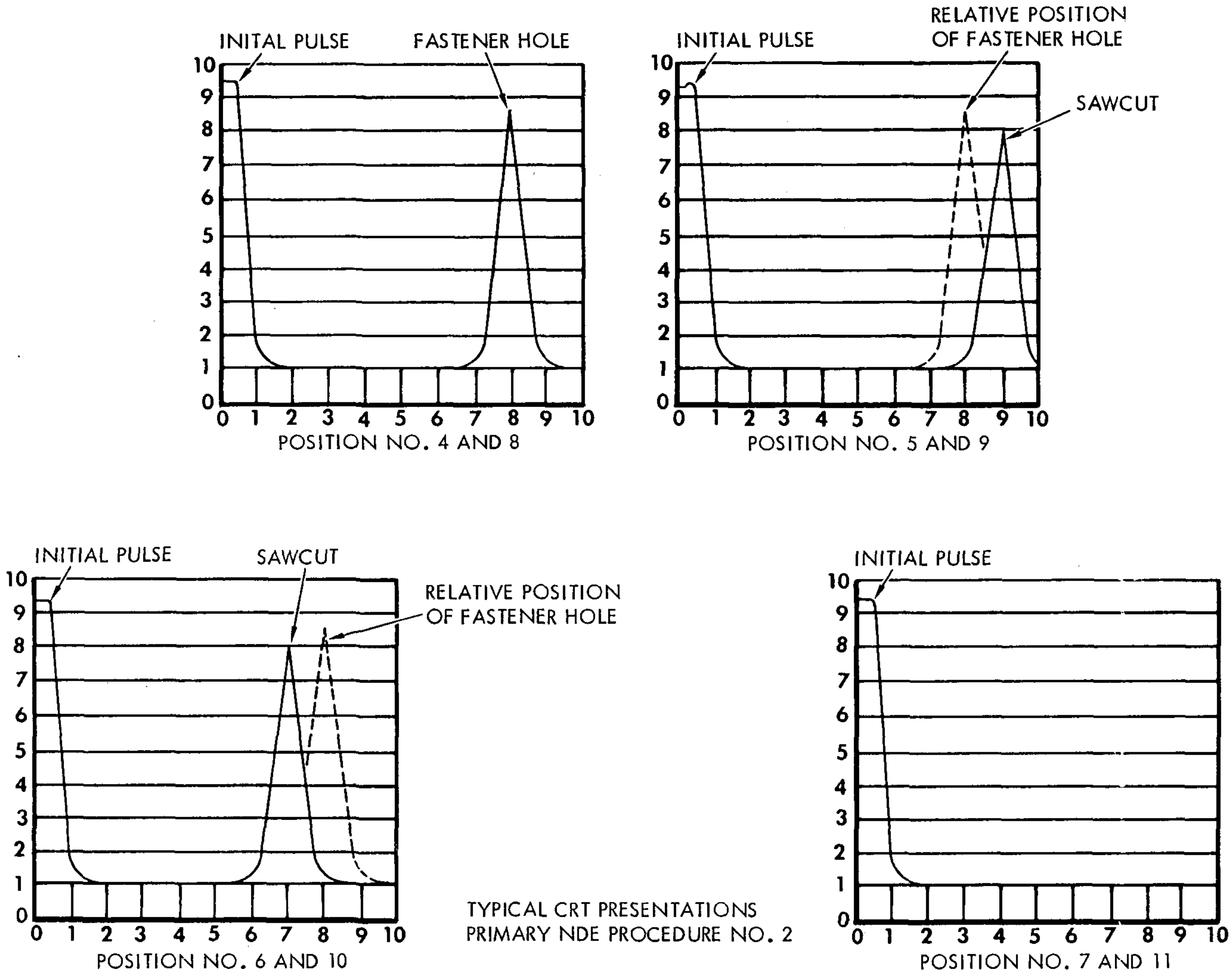
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 2

SMR151-58.4-0-005-5

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 5 of 7)

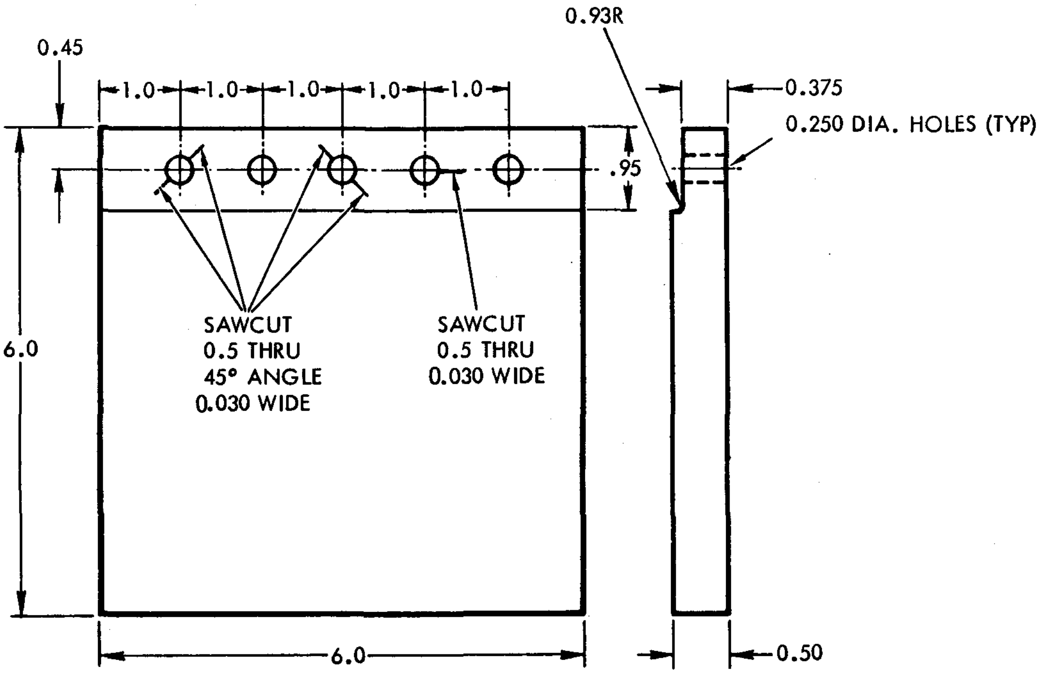


CALIBRATION SETUP
PRIMARY NDE PROCEDURE
NO. 2



TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 2

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 6 of 7)



CALIBRATION STANDARD
MATERIAL: PLEX 55
ROHM AND HAAS OR
SECTION OF A SCRAPPED
WINDOW

SMR151-58.4-0-005-7

Figure 58.4-5-6/2. Cabin Windows and Frames (Sheet 7 of 7)

58.4.7 HYDRAULIC ACCUMULATORS.

58.4.7.1 DESCRIPTION. (See figure 58.4-6-2/0.) The hydraulic system contains six accumulators which are chrome plated 4130 steel cylinders with 2024-T4 anodized aluminum alloy end caps. These are the brake system accumulator, the thrust reverser accumulator, the normal hydraulic system accumulator, the standby hydraulic system accumulator and the two landing gear accumulators. Refer to Section 12.4 of this manual for location of the accumulators.

58.4.7.2 DEFECTS. The end caps of the accumulators are susceptible to fatigue cracking on the inner and outer surfaces in random orientations.

58.4.7.3 PRIMARY NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE Equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Not applicable. The accumulators are inspected during overhaul.

c. Access: The caps are accessible after the accumulators have been disassembled.

WARNING

Provide adequate ventilation when using solvents. Avoid prolonged breathing of vapor and skin contact.

d. Preparation of Part: Clean the inner and outer surface of the end caps with dry cleaning solvent (Federal Specification P-D-680, Type II) and dry off thoroughly.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect the entire inner and outer surfaces of the twelve end caps. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.4.7.4 BACKUP NDE PROCEDURE. None required.

58.4.7.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

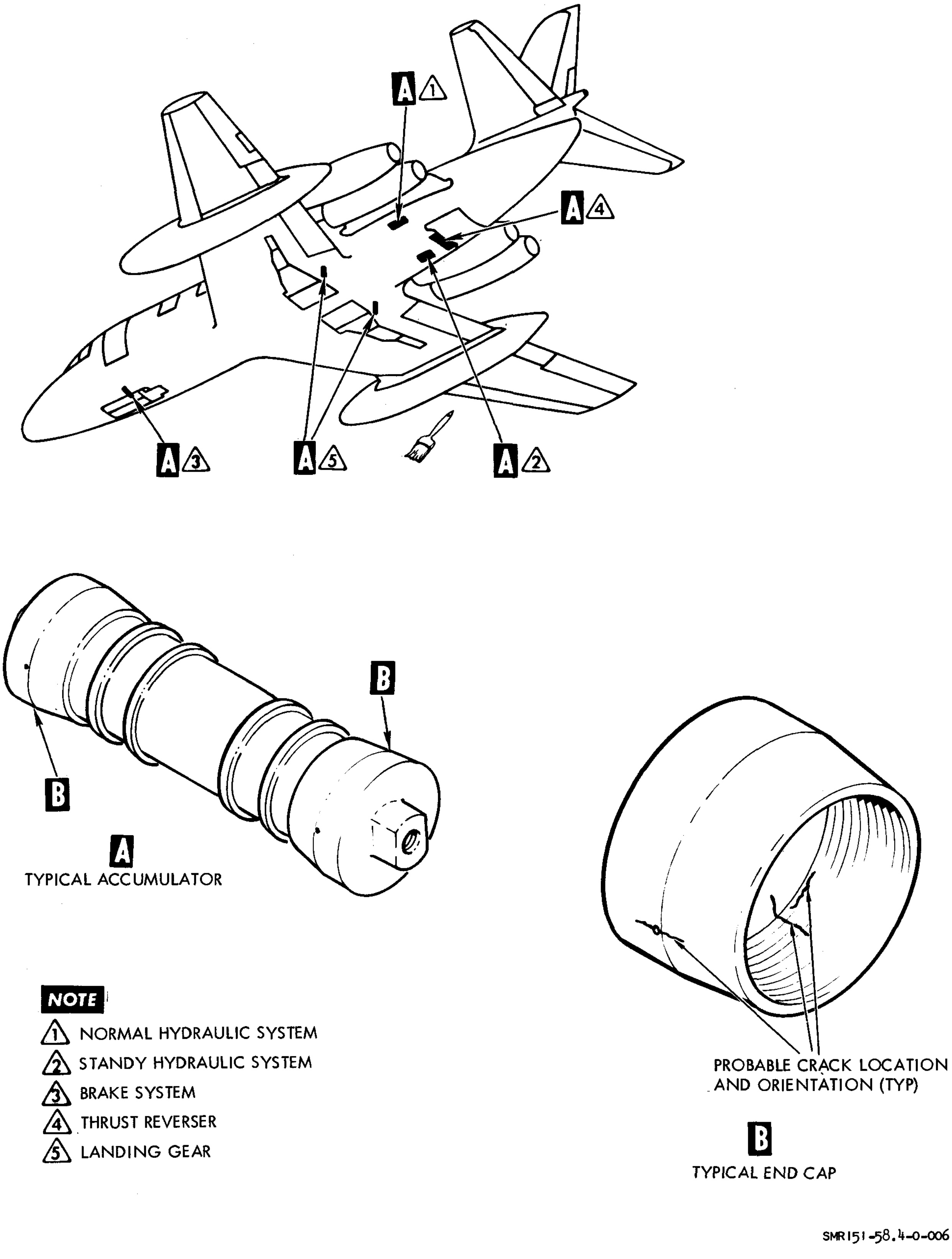


Figure 58.4-6-2/0. Hydraulic Accumulators

58.4.8 FORWARD CAP ANGLE, SPLICE DOUBLER, AND SKIN, AFT PRESSURE BULKHEAD, FUS STA 570.

58.4.8.1 DESCRIPTION. (See figure 58.4-7-6/4.) The outer cap is formed by the forward cap angle and the aft cap angle attached back to back to the FS 570 diaphragm. A splice doubler is added between the angles and fuselage skin. The angles, splice doubler and fuselage skins are fabricated from 2024-T4 aluminum sheet material. The angles, splice doubler and interior of the fuselage skin are finished with zinc chromate primer. The exterior surface finish will conform to customer peculiar paint scheme.

58.4.8.2 DEFECTS. Flexing of the fuselage may cause fatigue cracks in the fastener holes on the forward side of the FS 570 joint.

58.4.8.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD

Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray unit, Portable, 40 degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

Note

The interior equipment and trim will conform to customer peculiar scheme.

(1) Remove the interior equipment and trim on the forward side of the bulkhead as required for access to the forward cap angle.

(2) Remove the two walkway panel assemblies between FS 530 and 570. Refer to Section 53.2 for panel locations.

c. Access: The inspection areas are accessible from outside the aircraft with the use of appropriate maintenance stands. The interior surface is accessible from inside the fuselage.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set X-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate X-ray unit and center X-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.8.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, Ideal Specialty Co. P/N 6200-3/16-BH-BNC or equivalent.

(3) Calibration Standard, Aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

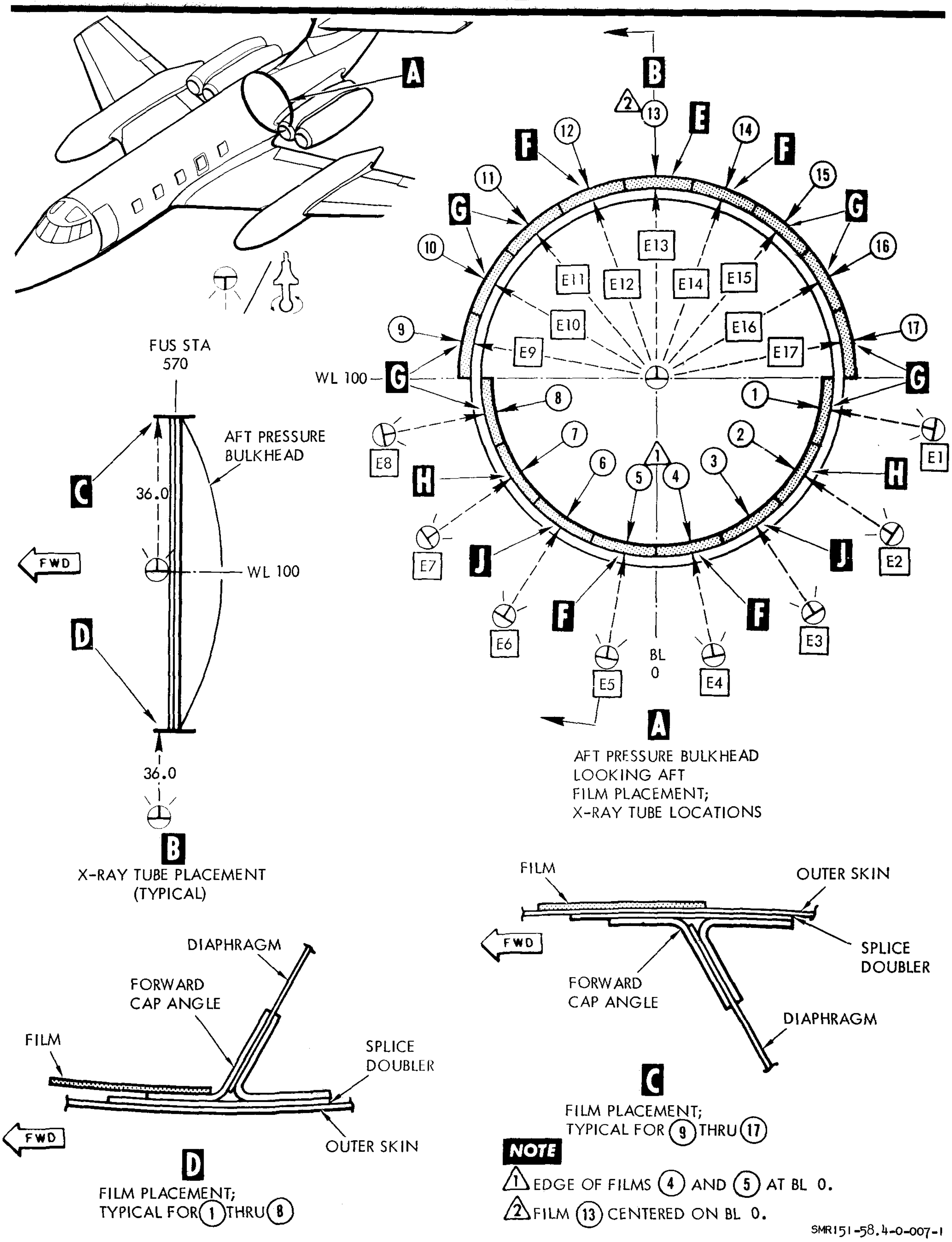
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.8.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



A
AFT PRESSURE BULKHEAD
LOOKING AFT
FILM PLACEMENT;
X-RAY TUBE LOCATIONS

B
X-RAY TUBE PLACEMENT
(TYPICAL)

C
FILM PLACEMENT;
TYPICAL FOR ⑨ THRU ⑰

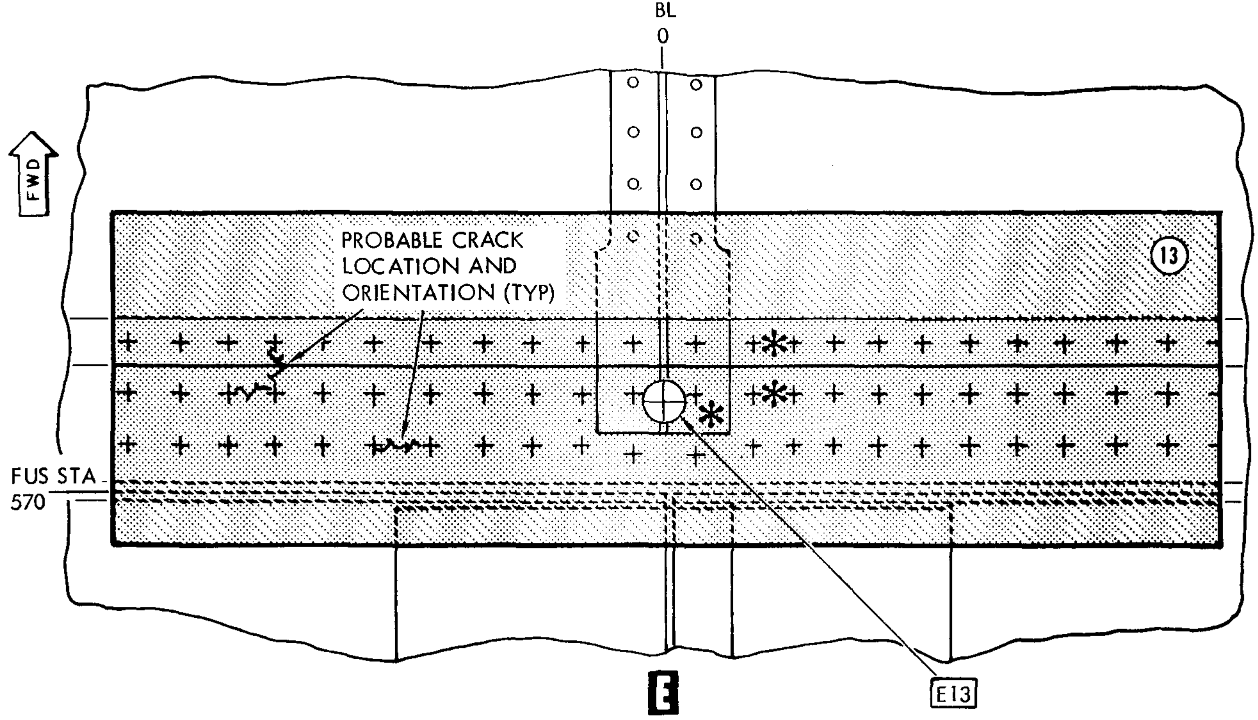
D
FILM PLACEMENT;
TYPICAL FOR ① THRU ⑧

NOTE

△1 EDGE OF FILMS ④ AND ⑤ AT BL 0.

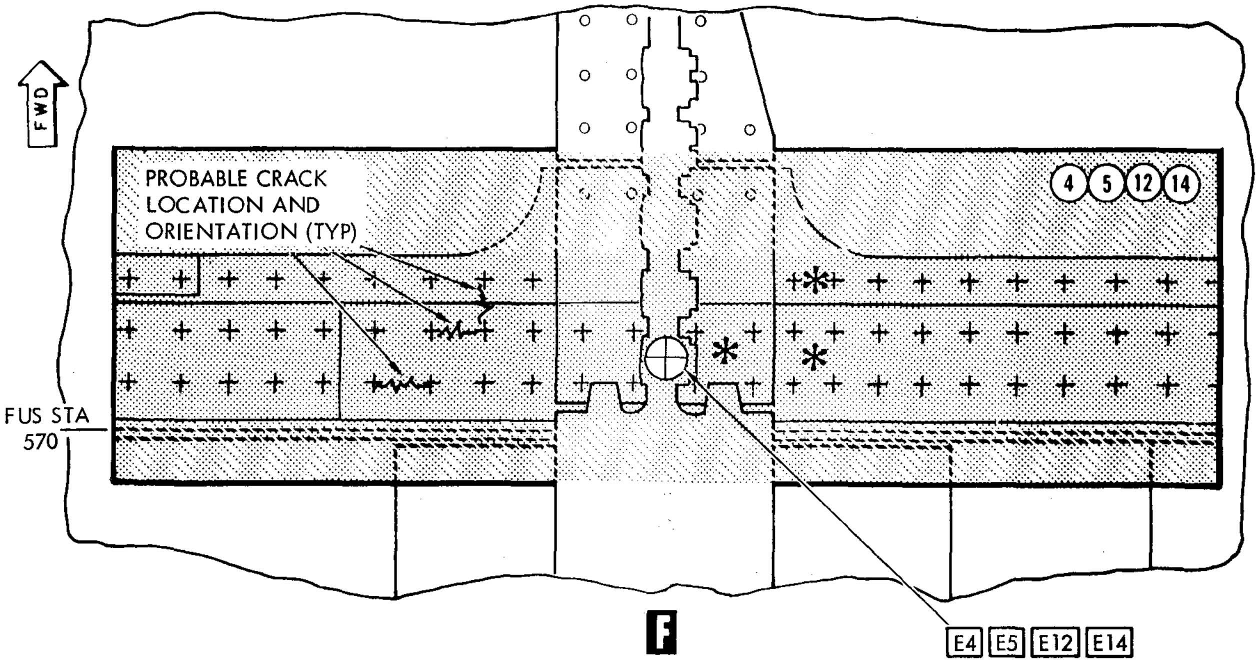
△2 FILM ⑬ CENTERED ON BL 0.

Figure 58.4-7-6/4. Forward Cap Angle, Splice Doubler, and Skin, Aft Pressure Bulkhead, Fus Sta 570 (Sheet 1 of 4)



E

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS



F

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

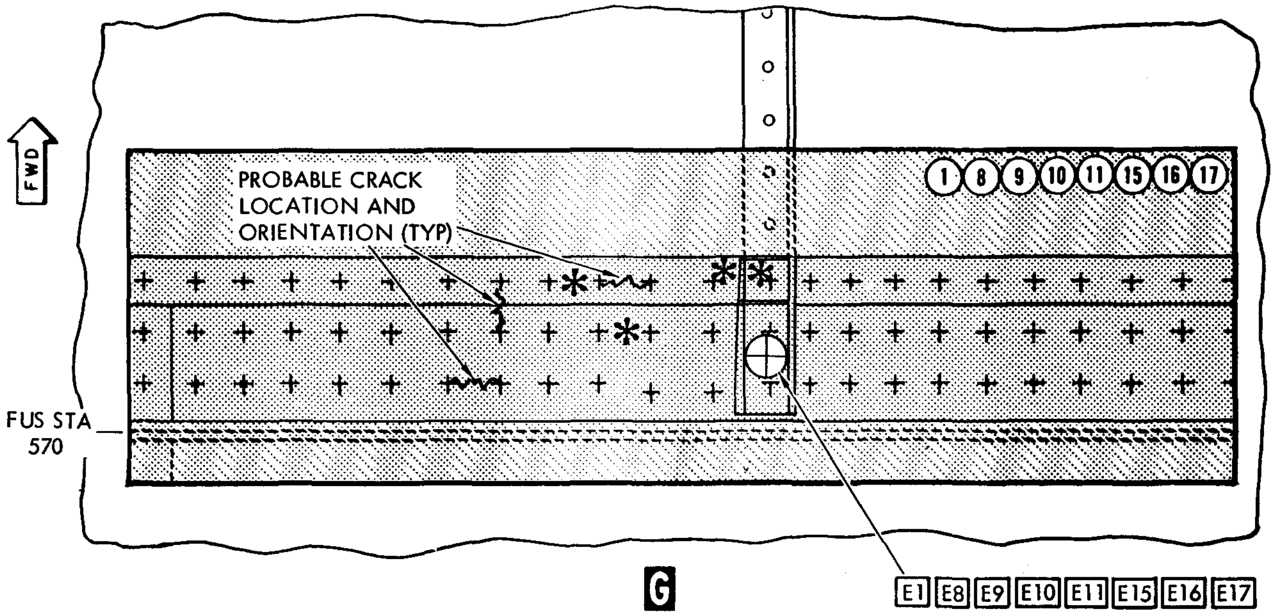
✱ CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

NOTE

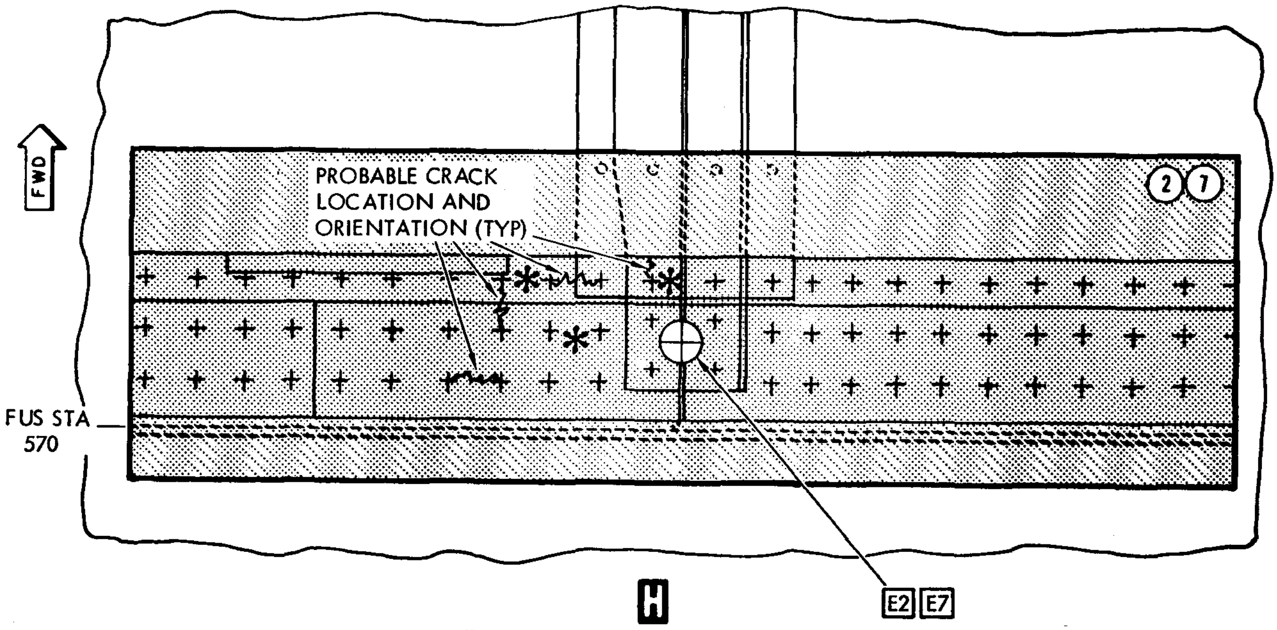
3. CRACKS MAY ORIGINATE AT FASTENER HOLES AND PROPAGATE IN ANY DIRECTION.

SMR151-58.4-0-007-2

Figure 58.4-7-6/4. Forward Cap Angle, Splice Doubler, and Skin, Aft Pressure Bulkhead, Fus Sta 570 (Sheet 2 of 4)


FWD
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
1 8 9 10 11 15 16 17
FUS STA 570
E1 E8 E9 E10 E11 E15 E16 E17
G

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS



FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

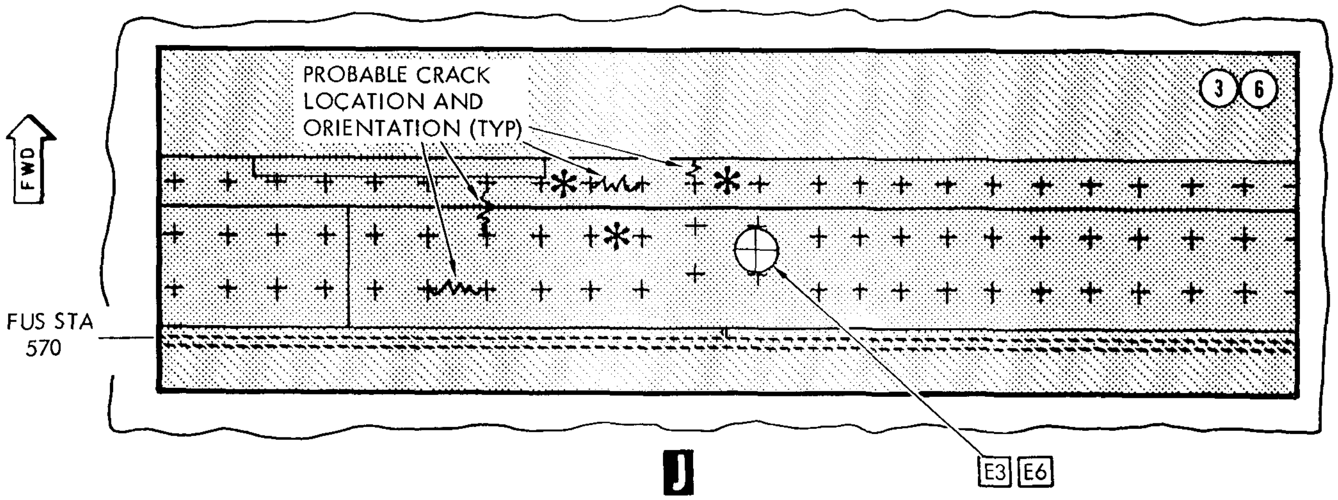
✱ CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

NOTE

3. CRACKS MAY ORIGINATE AT FASTENER HOLES AND PROPAGATE IN ANY DIRECTION.

SMR151-58.4-0-007-3

Figure 58.4-7-6/4. Forward Cap Angle, Splice Doubler, and Skin, Aft Pressure Bulkhead, Fus Sta 570 (Sheet 3 of 4)



J

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

✱ CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

NOTE

3. CRACKS MAY ORIGINATE AT FASTENER HOLES AND PROPAGATE IN ANY DIRECTION.

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	90	3.5	36	60	NDT 45 NDT 55	4.5 X 17.0	FILM NUMBER SAME AS EXPOSURE NUMBER
E2							
E3							
E4							
E5							
E6							
E7							
E8							
E9							
E10							
E11							
E12							
E13							
E14							
E15							
E16							
E17	90	3.5	36	60	NDT 45 NDT 55	4.5 X 17.0	

SMR151-58.4-0-007-4

Figure 58.4-7-6/4. Forward Cap Angle, Splice Doubler, and Skin, Aft Pressure Bulkhead, Fus Sta 570 (Sheet 4 of 4)

58.4.9 INNER CAP AND CHANNEL CAP TO FORWARD AND AFT CHANNEL ATTACHMENT OF FS 589.5 FRAME.

58.4.9.1 DESCRIPTION. (See figure 58.4-8-4/4.) The inboard cap of the forward engine mount frame is formed by attaching the inner cap and channel cap to the forward and aft channels. These parts are fabricated from 7075-T6 clad sheet and finished with zinc chromate primer.

58.4.9.2 DEFECTS. Fatigue cracks may occur in the inner cap, channel cap, forward and aft channel. Cracks will initiate in the fastener holes and propagate forward and aft.

58.4.9.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded Pencil, High Sensitivity, VM Products P/N VM100SS or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

b. Preparation of airplane: Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

c. Access: Access to the inspection areas is from beneath the airplane through the lower aft access door opening.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean local inspection areas of the main frame and skins as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using the shielded pencil probe, scan the forward and aft channels between WL 85.0 and WL 112.0, in a clockwise and counterclockwise direction. Scan around all fasteners common to the channel cap and inner cap.

(2) Using the shielded pencil probe, scan the inner cap between WL 85.0 and WL 112.0, in a clockwise and counterclockwise direction. Scan around all fasteners common to the channel cap and the forward and aft channels.

(3) Using the shielded pencil probe, scan the channel cap between WL 85.0 and WL 112.0, along the forward and aft edges of the inner cap.

(4) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects in the forward channel, aft channel, or inner cap, perform Backup NDE Procedure No. 1. To confirm suspected defects in the channel cap, perform Backup NDE Procedure No. 2. If no defect is suspected, proceed to System Securing.

58.4.9.4 BACKUP NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.9.5 BACKUP NDE PROCEDURE NO. 2 - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

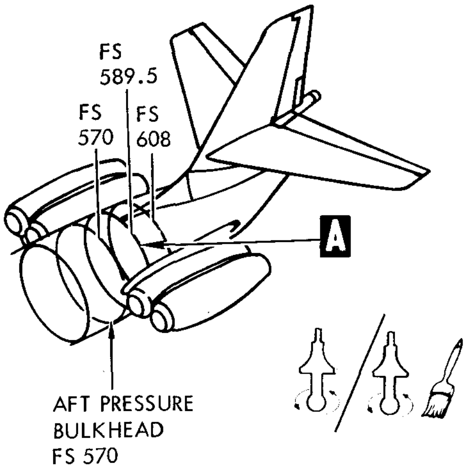
d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean local suspected defective areas.

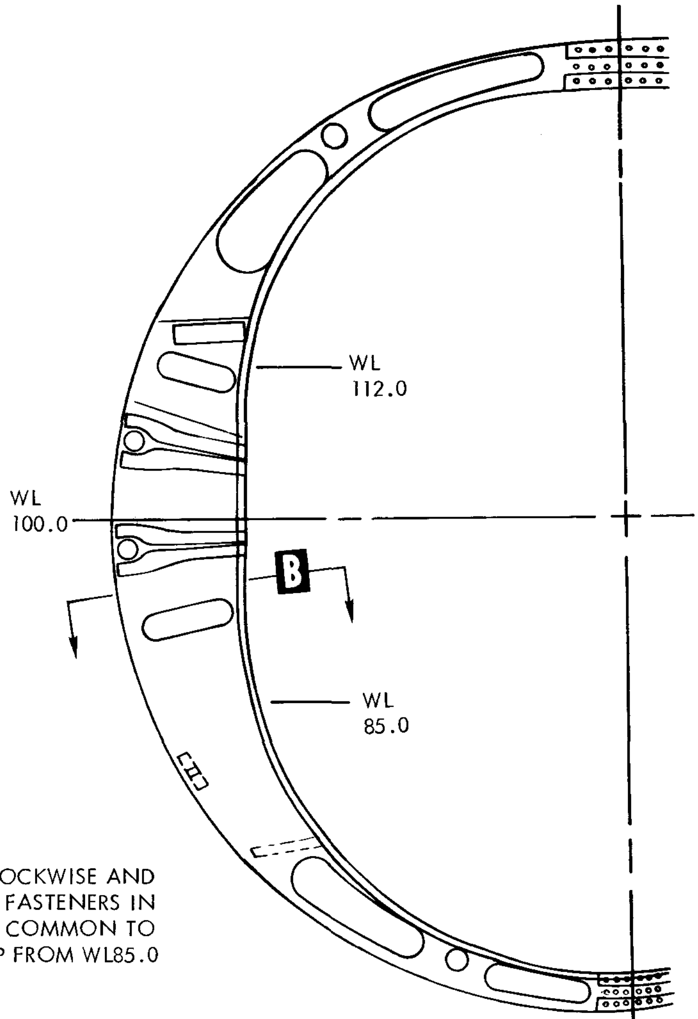
e. *Instrument settings/calibration:* Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

58.4.9.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.





A

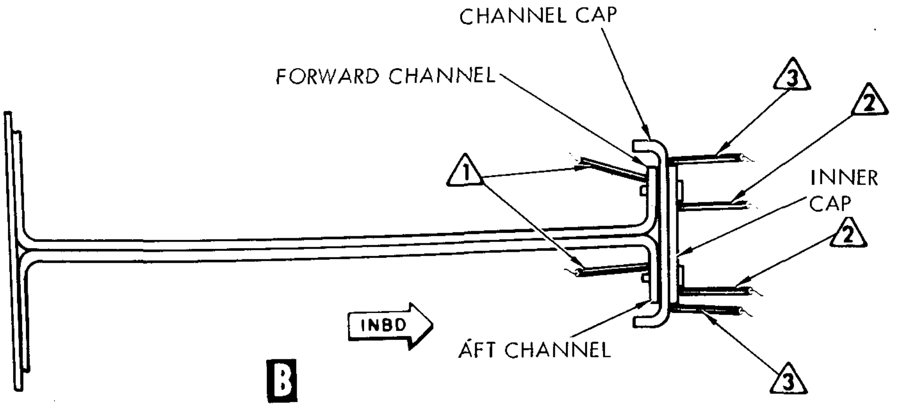
FS589.5 FRAME
LOOKING FORWARD
LEFT SIDE SHOWN:
RIGHT SIDE OPPOSITE

NOTE

△1 SHIELDED PENCIL PROBE - SCAN CLOCKWISE AND COUNTERCLOCKWISE AROUND ALL FASTENERS IN THE FORWARD AND AFT CHANNELS COMMON TO THE CHANNEL CAP AND INNER CAP FROM WL85.0 TO WL112.0.

△2 SHIELDED PENCIL PROBE - SCAN CLOCKWISE AND COUNTER CLOCKWISE AROUND ALL FASTENERS IN THE INNER CAP COMMON TO THE CHANNEL CAP AND FORWARD AND AFT CHANNELS FROM WL85.0 TO WL 112.0.

△3 SHIELDED PENCIL PROBE - SCAN THE CHANNEL CAP IN BOTH DIRECTIONS ALONG THE FORWARD AND AFT EDGES OF THE INNER CAP FROM WL85.0 TO WL112.0.



B

SMR151-58.4-0-008

Figure 58.4-8-4/4. Inner Cap and Channel Cap to Forward and Aft Channel Attachment of FS 589.5 Frame

58.4.10 FUSELAGE MAIN FRAME TO PRESSURE SKIN ATTACHMENT, LBL 9.0 TO RBL 9.0.

58.4.10.1 DESCRIPTION. (See figure 58.4-9-4/4.) The fuselage pressure skins in the underfloor area are attached to the forward and aft sides of the fuselage main frames at fuselage stations 410, 430, 450, 470 and 490. On airplanes 5046 and up, the pressure skin attachments to the forward and aft sides of FS 430, 450, and 470 main frames and the forward side of FS 490 main frame are reinforced with straps on the upper surface of the pressure skin. The frames are fabricated from 7075-T6 or 7075-T73 aluminum forgings and finished with zinc chromate primer. The skins and straps are fabricated from 2024-T3 clad aluminum sheet and finished with zinc chromate primer.

58.4.10.2 DEFECTS. Fatigue cracks may occur in the fastener holes where the fuselage pressure skin is attached to the fuselage main frames. Cracks will initiate in the fastener hole and propagate fore and aft.

58.4.10.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded Pencil, High Sensitivity, VM Products P/N VM100SS or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove ADF sense antennas to provide access to the main frame from left BL9.0 to right BL9.0 at fuselage stations 410, 430, 450, 470 and 490. Refer to Section 34.5 of this manual.

(3) Remove the walkway panel assemblies as required to provide access to the internal main frame from left BL9.0 to right BL9.0 at fuselage station 410, 430, 450, 470 and 490. Refer to Chapter 53 of this manual.

c. Access: Access to the inspection areas is from beneath the airplane and from inside the fuselage through the underfloor structure.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean local inspection areas of the main frame and skins as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

f. Inspection:

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

(1) Using the shielded surface probe, scan around the fasteners attaching the fuselage pressure skin and/or strap to the main frame from BL9.0 left to BL9.0 right at fuselage stations 410, 430, 450, 470 and 490. Scan around the fasteners on the lower surface of the outside member and around the upper surface of the inside member of the joint.

(2) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.10.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 5/32-inch diameter, VM Products P/N VM101B or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

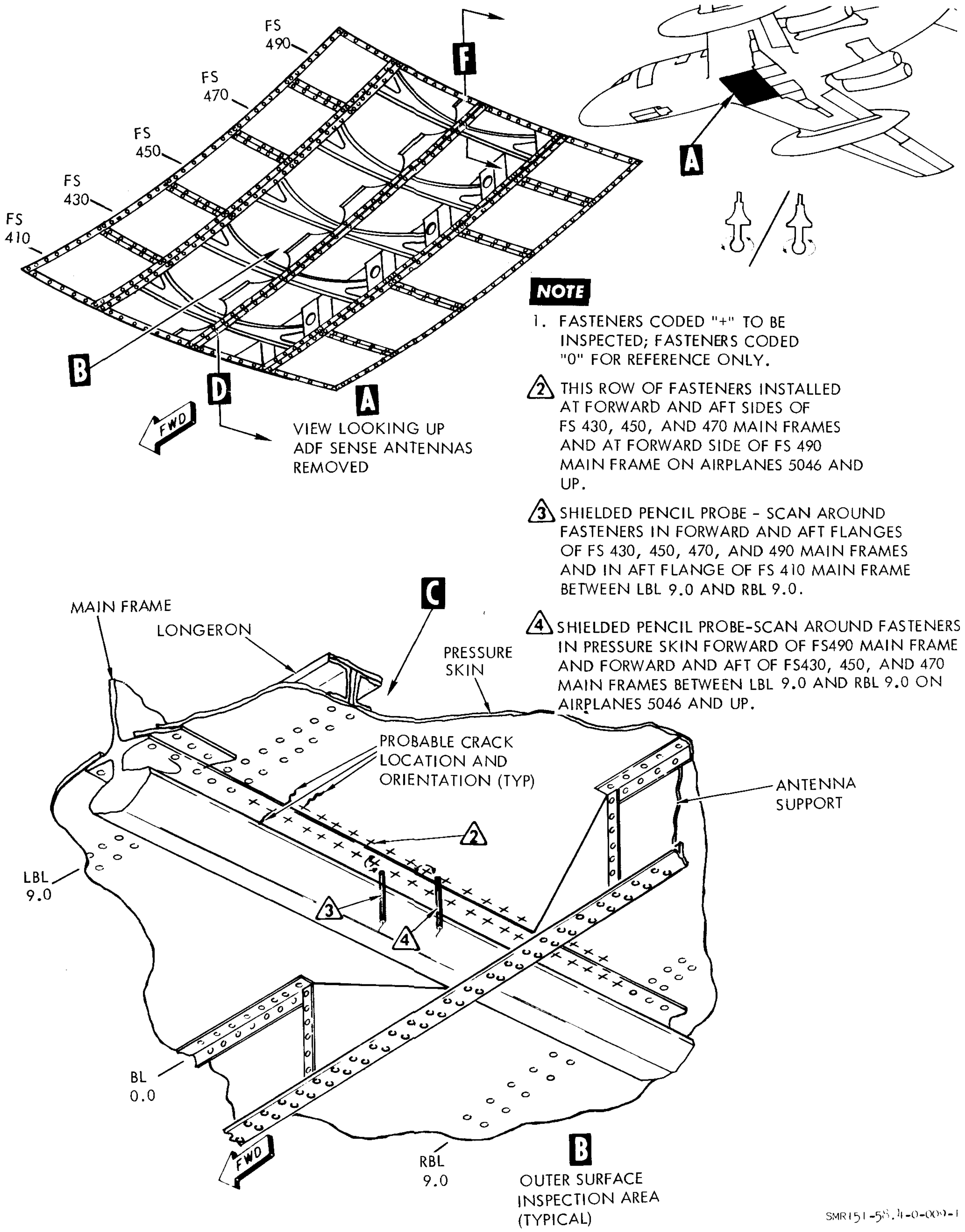
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using appropriate size probe, scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.10.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

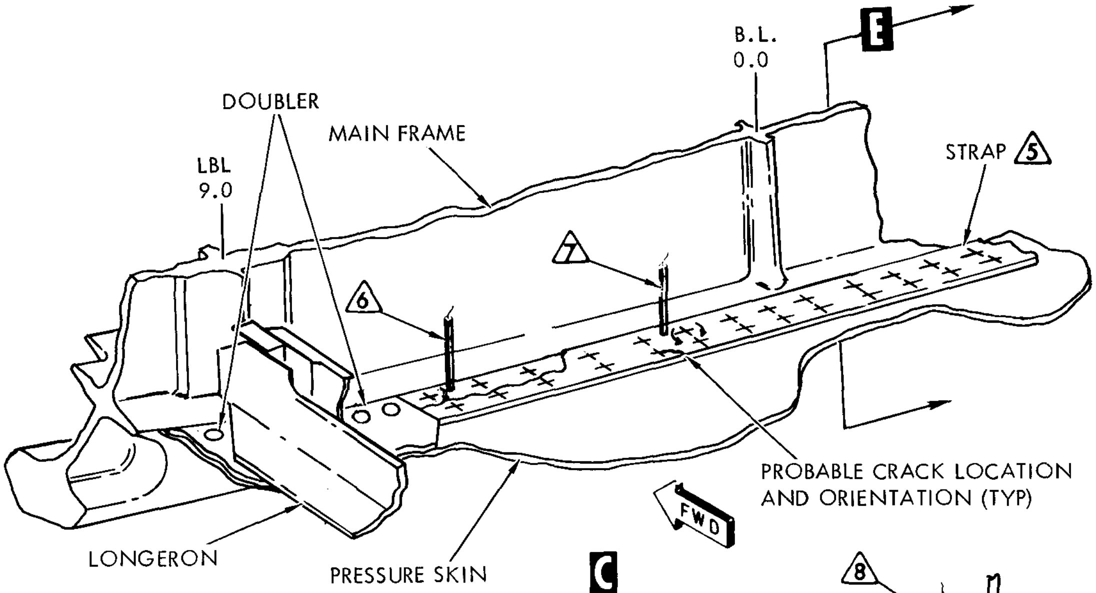
1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

⚠2 THIS ROW OF FASTENERS INSTALLED AT FORWARD AND AFT SIDES OF FS 430, 450, AND 470 MAIN FRAMES AND AT FORWARD SIDE OF FS 490 MAIN FRAME ON AIRPLANES 5046 AND UP.

⚠3 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN FORWARD AND AFT FLANGES OF FS 430, 450, 470, AND 490 MAIN FRAMES AND IN AFT FLANGE OF FS 410 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0.

⚠4 SHIELDED PENCIL PROBE-SCAN AROUND FASTENERS IN PRESSURE SKIN FORWARD OF FS490 MAIN FRAME AND FORWARD AND AFT OF FS430, 450, AND 470 MAIN FRAMES BETWEEN LBL 9.0 AND RBL 9.0 ON AIRPLANES 5046 AND UP.

Figure 58.4-9-4/4. Fuselage Main Frame to Pressure Skin Attachment, LBL 9.0 to RBL 9.0 (Sheet 1 of 2)



C

INNER SURFACE
INSPECTION AREA
(TYPICAL)

NOTE

1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

3 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN FORWARD AND AFT FLANGES OF FS430, 450, 470, AND 490 MAIN FRAMES AND IN AFT FLANGE OF FS410 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0.

4 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN PRESSURE SKIN FORWARD OF FS490 MAIN FRAME AND FORWARD AND AFT OF FS430, 450, AND 470 MAIN FRAMES BETWEEN LBL 9.0 AND RBL 9.0 ON AIRPLANES 5046 AND UP.

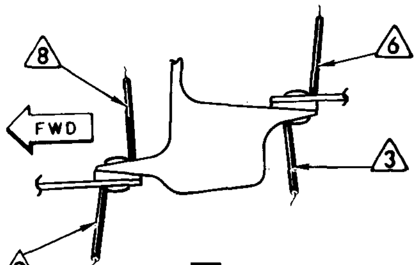
5 STRAP INSTALLED AT FORWARD AND AFT SIDES OF FS430, 450, AND 470 MAIN FRAMES AND AT FORWARD SIDE OF FS490 MAIN FRAME ON AIRPLANES 5046 AND UP.

6 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN PRESSURE SKIN COMMON TO MAIN FRAME ON FORWARD AND AFT SIDES OF FS430, 450, 470, AND 490 MAIN FRAMES AND ON AFT SIDE OF FS410 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0 ON AIRPLANES PRIOR TO 5046.

7 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN STRAP COMMON TO PRESSURE SKIN AND MAIN FRAME ON FORWARD AND AFT SIDES OF FS430, 450, AND 470 MAIN FRAMES AND ON FORWARD SIDE OF FS490 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0 ON AIRPLANES 5046 AND UP.

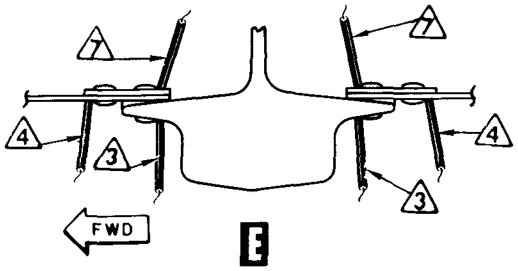
8 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN FORWARD FLANGE OF FS410 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0.

9 SHIELDED PENCIL PROBE - SCAN AROUND FASTENERS IN PRESSURE SKIN COMMON TO FORWARD FLANGE OF FS410 MAIN FRAME BETWEEN LBL 9.0 AND RBL 9.0.



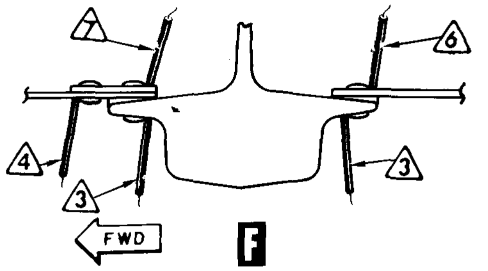
D

MAIN FRAME AT FS 410



E

MAIN FRAME (TYP)
FS 430, 450, 470



F

MAIN FRAME AT FS 490

SMR151-58.4-0-009-2

Figure 58.4-9-4/4. Fuselage Main Frame to Pressure Skin Attachment, LBL 9.0 to RBL 9.0 (Sheet 2 of 2)

58.4.11 FRAME OUTER CAP AT AFT ENGINE NACELLE MOUNT, FS 608.

58.4.11.1 DESCRIPTION. (See figure 58.4-10-6/4.) The nacelle box beam frames are attached to the frame at fuselage stations 589 and 608. The fuselage station 608 outer frame is made by joining channels back to back to the outer cap with the exterior fuselage skin. Engine mount fittings are attached to the forward and aft side of the frame to mate with the nacelle box beams. The channels and outer cap are made from 7075-T6 and finished with zinc chromate primer.

58.4.11.2 DEFECTS. Fatigue cracks may occur in the members of the frame outer cap above, between, and below the attachment of the nacelle box beam fittings. Cracks will be hole-to-hole or hole-to-edge of part.

58.4.11.3 PRIMARY NDE PROCEDURE - RADIOGRAPHIC.

WARNING

RADIATION HAZARD
Ensure compliance with all safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube Head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove upper box beam fillet left and right, and the lower forward box beam fairing left and right. Refer to Section 52.3 of this manual for fillet and fairing location.

c. Access: The area around the nacelle box beam fittings are accessible from outside the fuselage with the use of appropriate AGE or maintenance stands. Access to the interior of the fuselage is through the lower aft access door opening.

d. Preparation of part: No special preparation is required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiograph Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.11.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520, or equivalent.

(2) Probe, Bolt Hole, 5/32-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(4) Probe, Bolt Hole, 1/4-inch diameter, VM Products P/N VM101B or equivalent.

(5) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: In addition to Primary NDE Procedure, if the suspected defective fastener hole is covered by the nacelle box beam, remove the engine and nacelle. Refer to Section 54.0 for removals.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe. The nacelle box beam frames are steel and should not be contaminated by the cleaning process.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

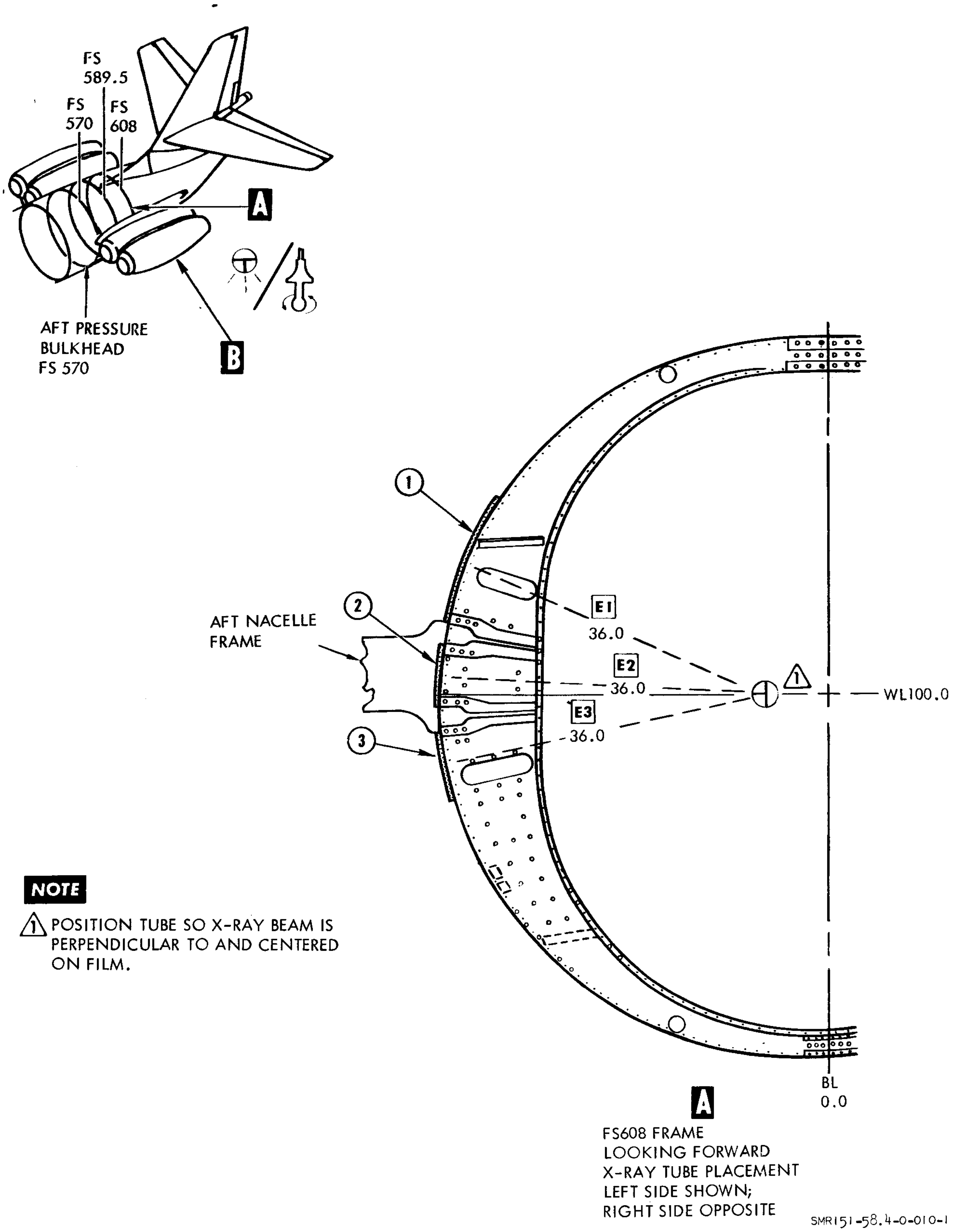
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using the appropriate bolt hole probe, scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.11.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND CENTERED ON FILM.

FS608 FRAME
LOOKING FORWARD
X-RAY TUBE PLACEMENT
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

SMR151-58.4-0-010-1

Figure 58.4-10-6/4. Frame Outer Cap at Aft Engine Nacelle Mount, FS 608 (Sheet 1 of 3)

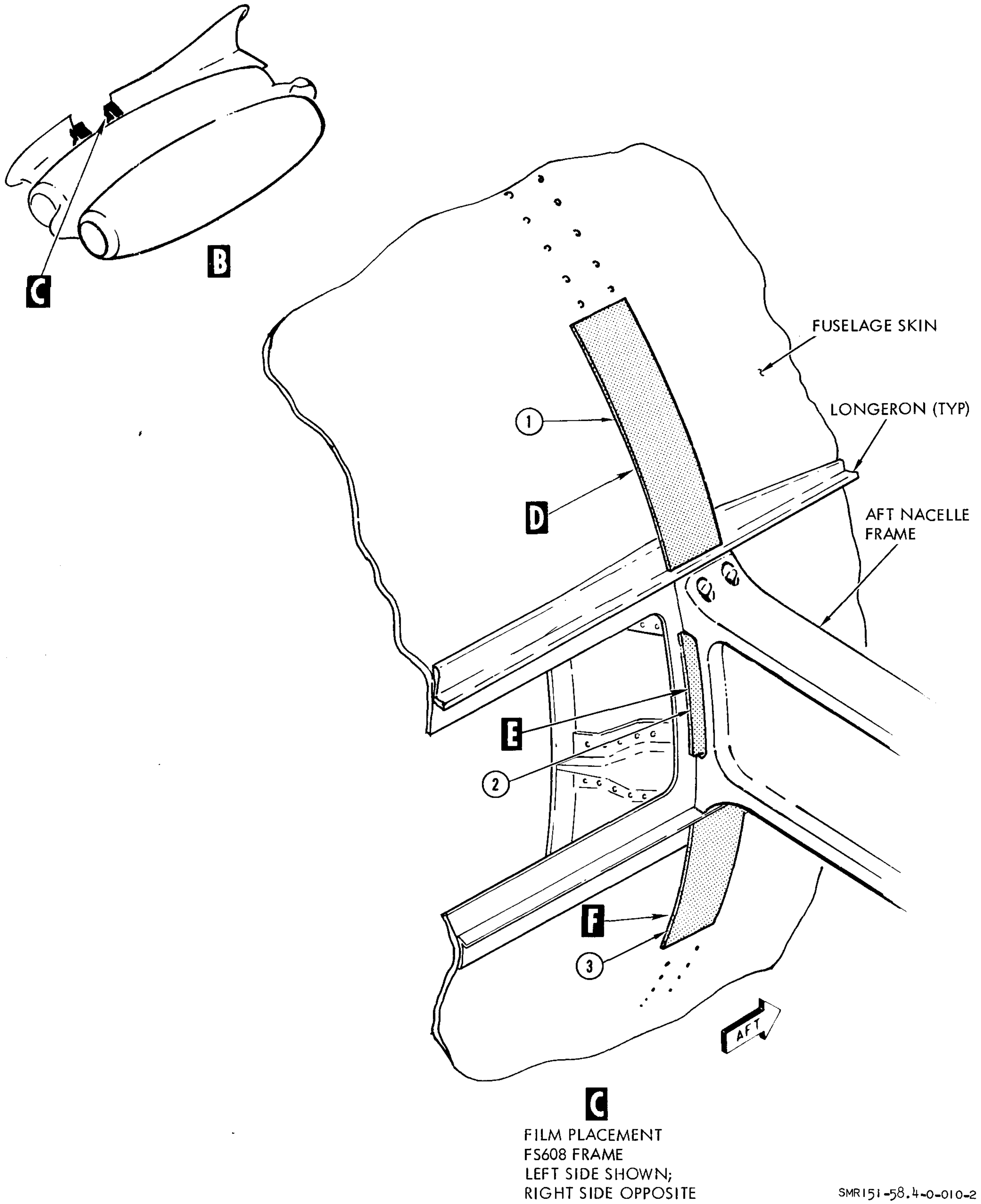
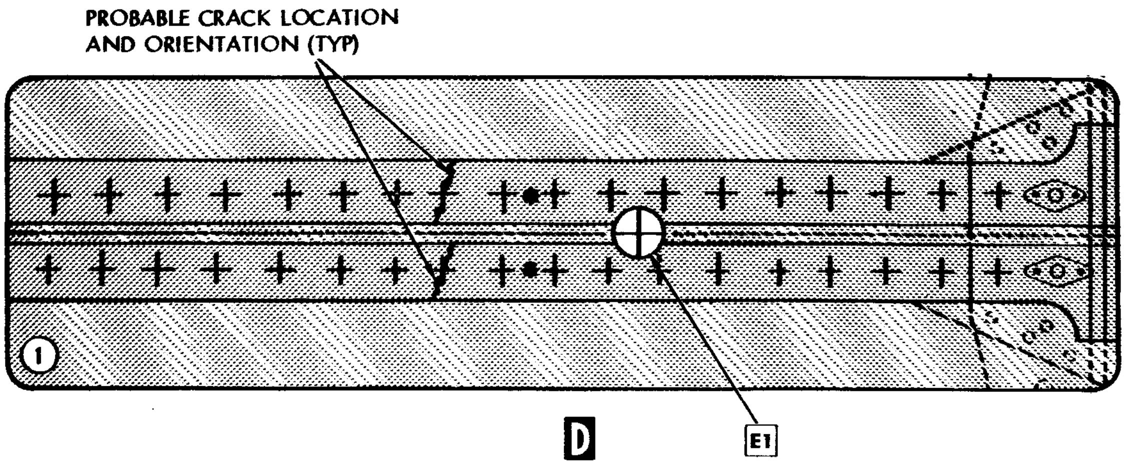
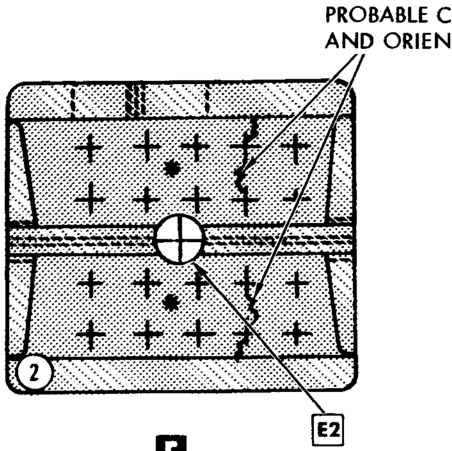


Figure 58.4-10-6/4. Frame Outer Cap at Aft Engine Nacelle Mount, FS 608
(Sheet 2 of 3)



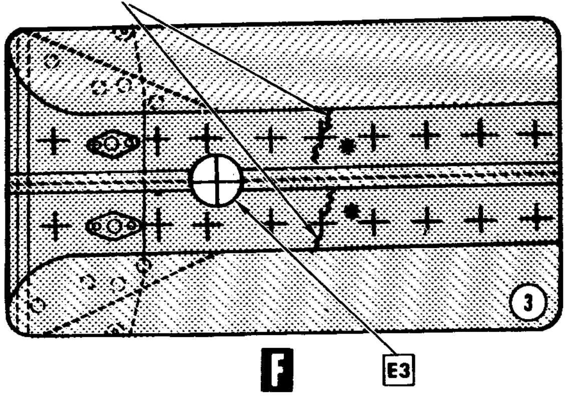
D

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS



E

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS



F

FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

✱ CHECK MEDIAN FILM DENSITY AT NOTED POINTS.

RADIOGRAPHIC INSPECTION DATA

EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	80	3	36	80	NDT 55	4.5 X 17	FILM NUMBER SAME AS EXPOSURE NUMBER
E2	80	3	36	80	NDT 55	4.5 X 5	
E3	80	3	36	80	NDT 55	4.5 X 8.5	

SMR151-58.4-0-010-3

Figure 58.4-10-6/4. Frame Outer Cap at Aft Engine Nacelle Mount, FS 608 (Sheet 3 of 3)

58.4.12 EMERGENCY EXIT CUTOUT UPPER CORNERS, FS410 AND FS430.

58.4.12.1 DESCRIPTION. (See figure 58.4-11-6/4.) The emergency exit cutout forward edge is common to the frame at FS410 and the aft edge is common to the frame at FS430. The frame at FS410 is fabricated from 7075-T6 aluminum sheet and plate. The frame at FS430 is fabricated from 7075-T6 aluminum forging. Underneath the fuselage skin there is a large 2024-T3 aluminum sheet doubler which reinforces the opening. There are corner elements at each of the four corners which are made from 2014-T6 forgings. At the top of the door there is a longeron, sill and angle extending forward and aft. The longeron and angle are made from 7075-T6 aluminum extrusions and the sill is 7075-T6 aluminum sheet. The inner surfaces are finished with zinc chromate primer and the exterior surface finish will conform to customer peculiar paint scheme.

58.4.12.2 DEFECTS. Fatigue cracks may occur in any of the layers in either of the upper corners of the emergency exit cutout. Cracks will be hole-to-hole and hole-to-edge of part. Also fatigue cracks may occur in the angle attached to the sill. The cracks will be oriented perpendicular to the length of the angle in the radius.

58.4.12.3 PRIMARY NDE PROCEDURE NO. 1 - RADIOGRAPHIC.

WARNING

RADIATION HAZARD
Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube head, 150 KVP, Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

(1) Remove the emergency exit hatch.

(2) Remove the interior trim from around the emergency exit opening.

(3) Pull the insulation blanket back from the interior side of the opening.

c. Access: The interior of the emergency exit opening is accessible from inside the fuselage. The inspection areas are accessible from outside the airplane with the use of appropriate maintenance stands.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed with Primary NDE Procedure No. 2.

58.4.12.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520, or equivalent.

(2) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM102AS or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the surface of angle three inches forward and aft of the frame at FS410 and 430 as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

f. Inspection:

(1) Using the shielded 90-degree probe, scan the angle around the fasteners three inches forward and aft of the FS410 and FS430 frames. Scan in a clockwise and counterclockwise direction.

(2) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE procedure. If no defect is suspected, proceed to System Securing.

58.4.12.5 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 1/4-inch diameter, VM Products P/N VM101B or equivalent.

(4) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

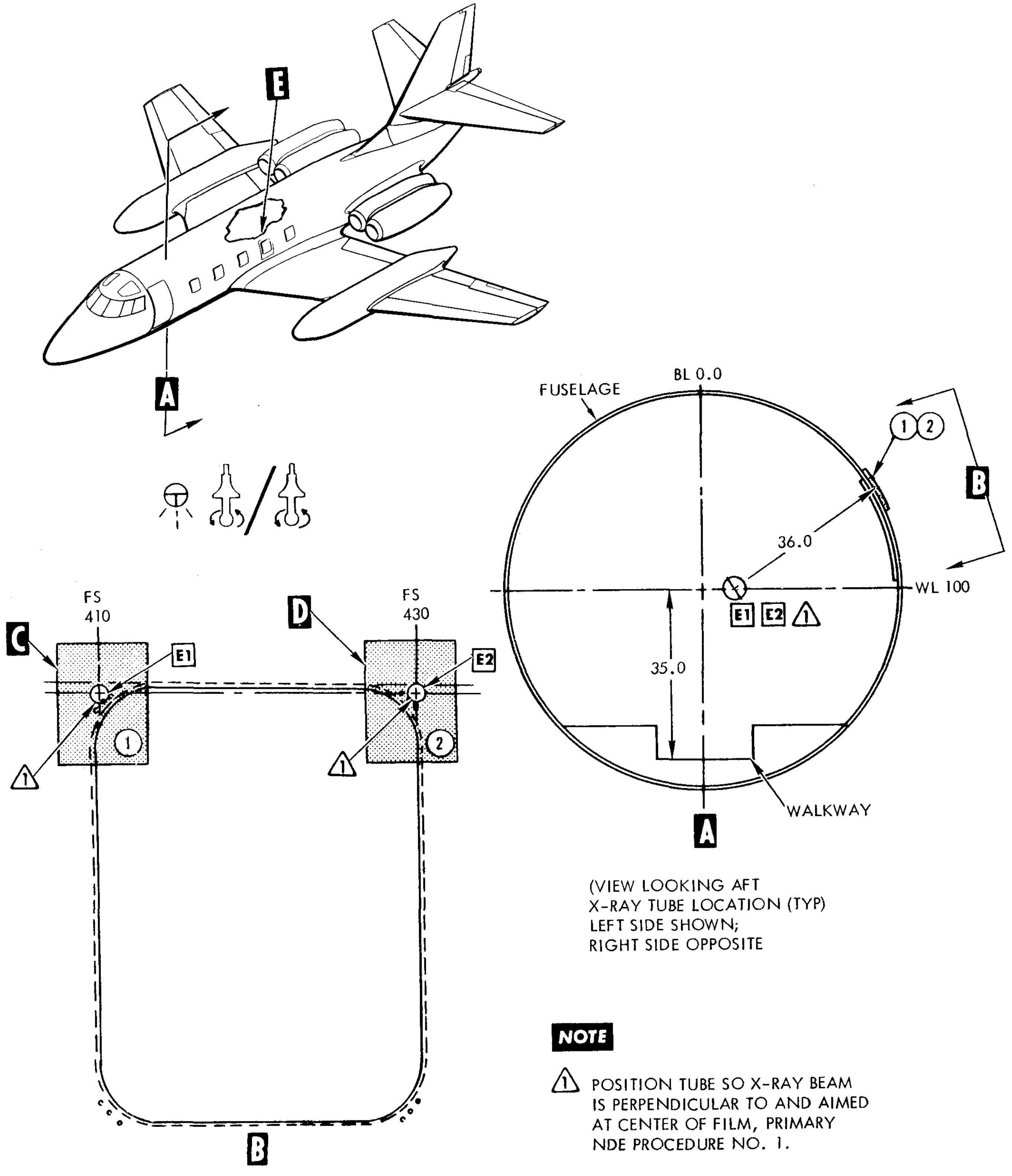
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using the appropriate bolt hole probe, scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.12.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



FILM PLACEMENT AND AIMING POINTS
LEFT SIDE SHOWN; RIGHT SIDE OPPOSITE

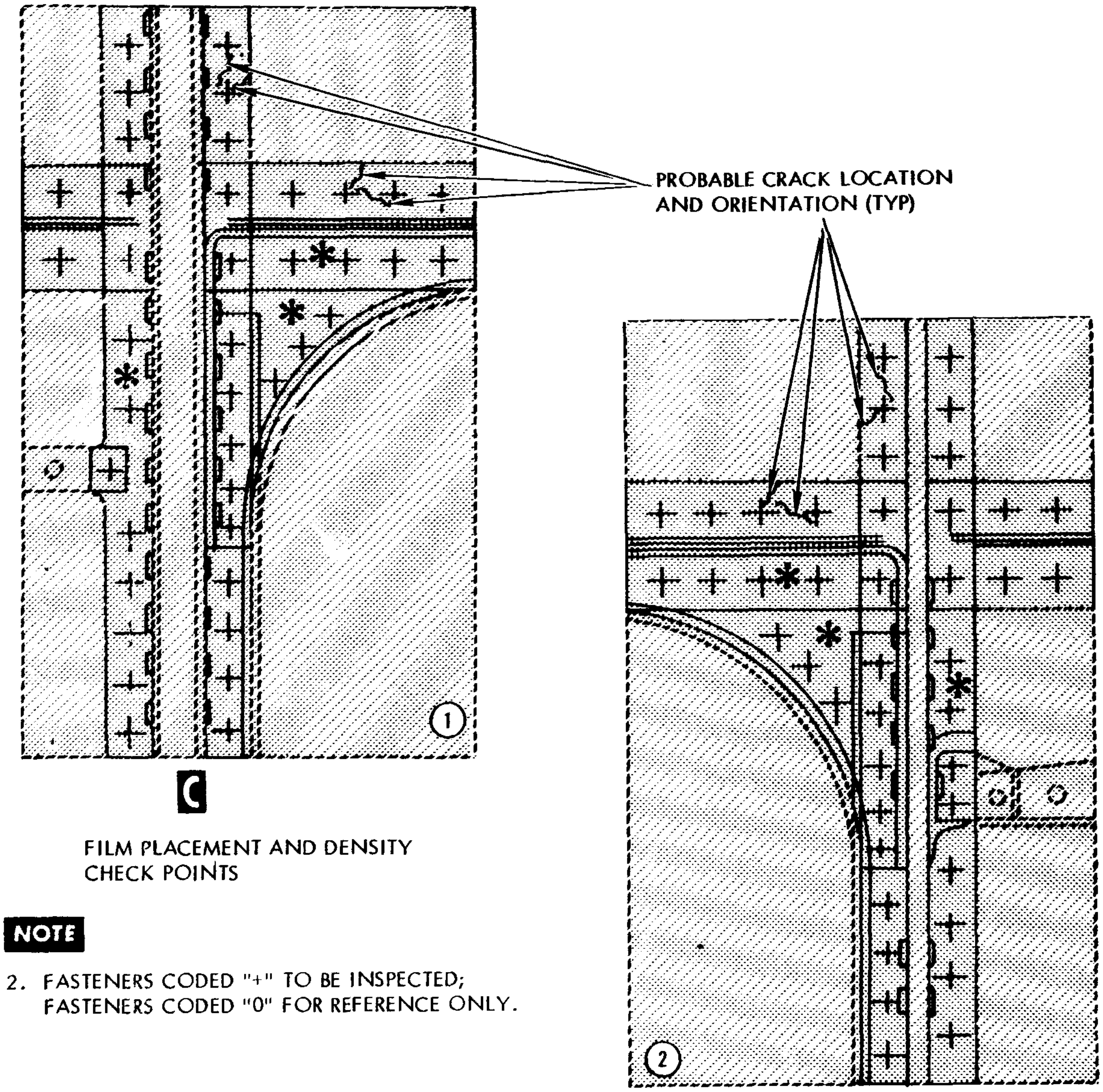
(VIEW LOOKING AFT
X-RAY TUBE LOCATION (TYP)
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

NOTE

⚠1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND AIMED AT CENTER OF FILM, PRIMARY NDE PROCEDURE NO. 1.

SMR151-58.4-0-011-1

Figure 58.4-11-6/4. Emergency Exit Cutout Upper Corners, FS 410 and FS 430 (Sheet 1 of 3)



C

FILM PLACEMENT AND DENSITY CHECK POINTS

NOTE

2. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

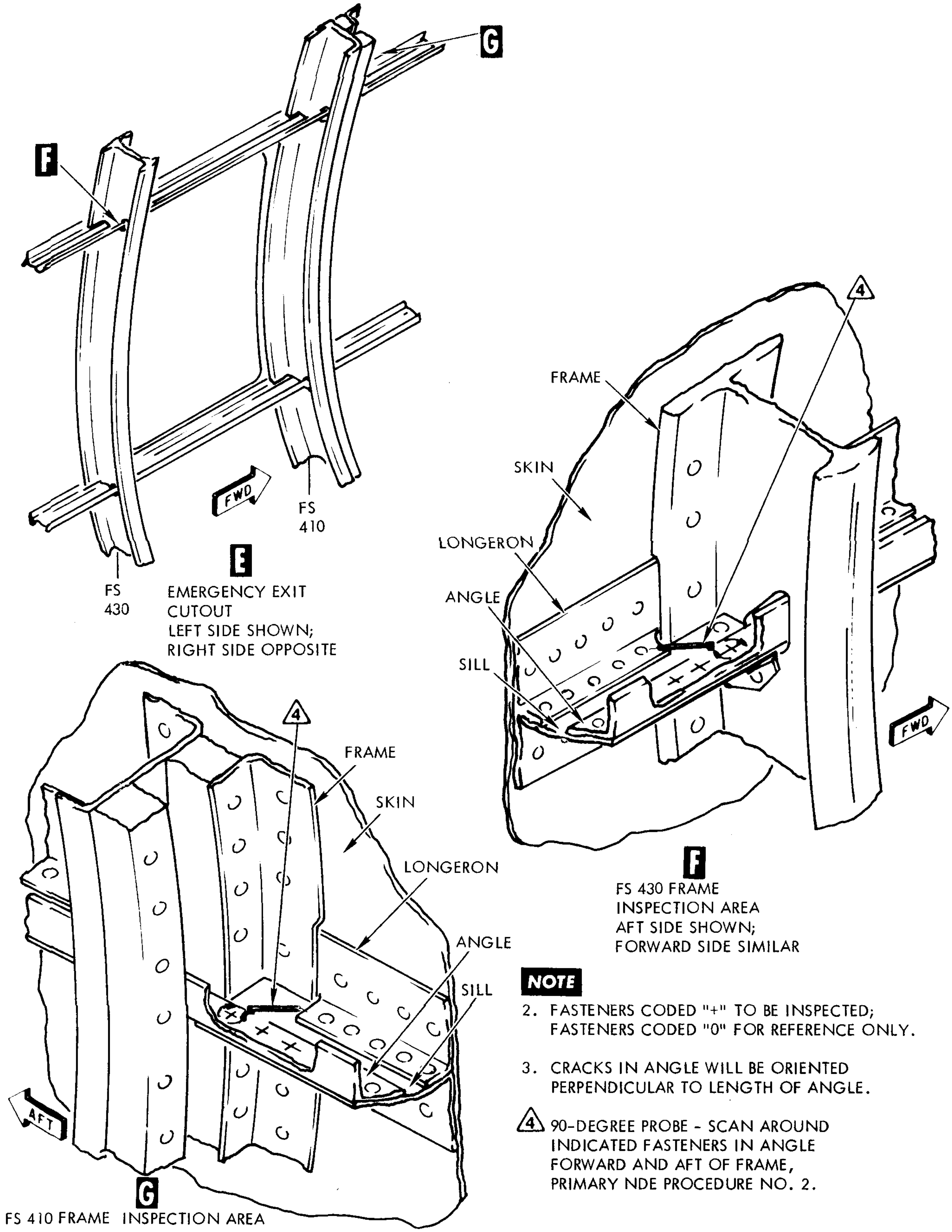
D

FILM PLACEMENT AND DENSITY CHECK POINTS

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1 E2	100	3	36.0	90	NDT 45/ NDT 55	8.5x14	FILM NUMBER SAME AS EXPOSURE NUMBER

SMR151-58.4-0-011-2

Figure 58.4-11-6/4. Emergency Exit Cutout Upper Corners, FS 410 and FS 430 (Sheet 2 of 3)



EMERGENCY EXIT CUTOUT
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

FS 430 FRAME
INSPECTION AREA
AFT SIDE SHOWN;
FORWARD SIDE SIMILAR

FS 410 FRAME INSPECTION AREA
FORWARD SIDE SHOWN; AFT SIDE SIMILAR

NOTE

2. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

3. CRACKS IN ANGLE WILL BE ORIENTED PERPENDICULAR TO LENGTH OF ANGLE.

⚠4 90-DEGREE PROBE - SCAN AROUND INDICATED FASTENERS IN ANGLE FORWARD AND AFT OF FRAME, PRIMARY NDE PROCEDURE NO. 2.

SMR151-58.4-0-011-3

Figure 58.4-11-6/4. Emergency Exit Cutout Upper Corners, FS 410 and FS 430 (Sheet 3 of 3)

58.4.13 UPPER 30-DEGREE LONGERON ATTACHMENT, FS 483 TO FS 496.

58.4.13.1 DESCRIPTION. (See figure 58.4-12-4/4.) The 30-degree longeron is located on the upper portion of the fuselage on the left and right side and extends the length of the fuselage. The outer cap of the longeron is a machined 7075-T6 aluminum extruded tee section which is attached to the fuselage skin. The cap is finished with zinc chromate primer. The exterior fuselage surface finish will conform to customer peculiar paint scheme.

58.4.13.2 DEFECTS. Flexing of the fuselage may cause fatigue cracks in the cap of the longeron or fuselage skin. Cracks will initiate in the fastener holes and propagate in a circumferential direction.

58.4.13.3 PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM100SS or equivalent.

(3) Probe, Shielded, 90-degree, High Sensitivity, VM Products P/N VM102AS or equivalent.

(4) Calibration Standard, aluminum (See figure 58.1-9).

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result serious injury to personnel.

b. Preparation of airplane:

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the interior trim for access to the longeron on the left and right side.

c. Access: The inspection area is accessible from outside the upper part of the fuselage with the use of appropriate maintenance stands. The interior surface of the longerons are accessible from inside the fuselage.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the surface area of the longeron and skin as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) From outside the fuselage, scan around the fasteners in the fuselage skin common to the longeron aft of the FS 482 frame back to the forward side of the FS 495 frame.

(2) From inside the fuselage, scan around the fasteners in the longeron aft of the FS 482 frame back to the forward side of the FS 495 frame. Omit the fasteners common to the FS 490 frame.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defects are suspected, proceed to System Securing.

58.4.13.4 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 5/32-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(4) Calibration Standard, aluminum (See figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure.

c. Access: Same as Primary NDE Procedure.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instructions. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

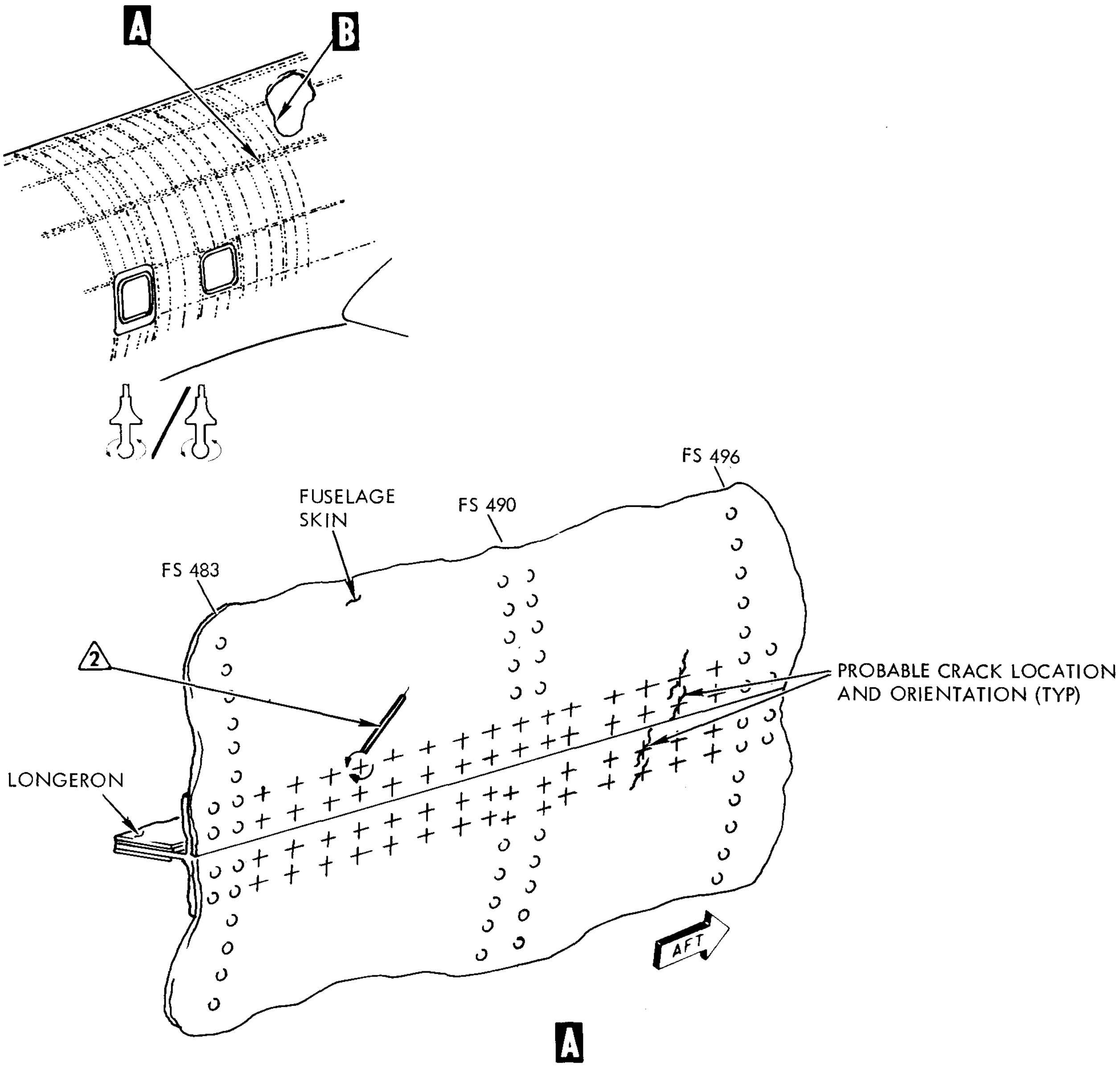
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using appropriate bolt-hole probe, scan the entire inner surface of the suspected defective fastener hole through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

58.4.13.5 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



UPPER 30-DEGREE LONGERON ATTACHMENT, EXTERNAL INSPECTION AREA
LEFT SIDE SHOWN:
RIGHT SIDE OPPOSITE

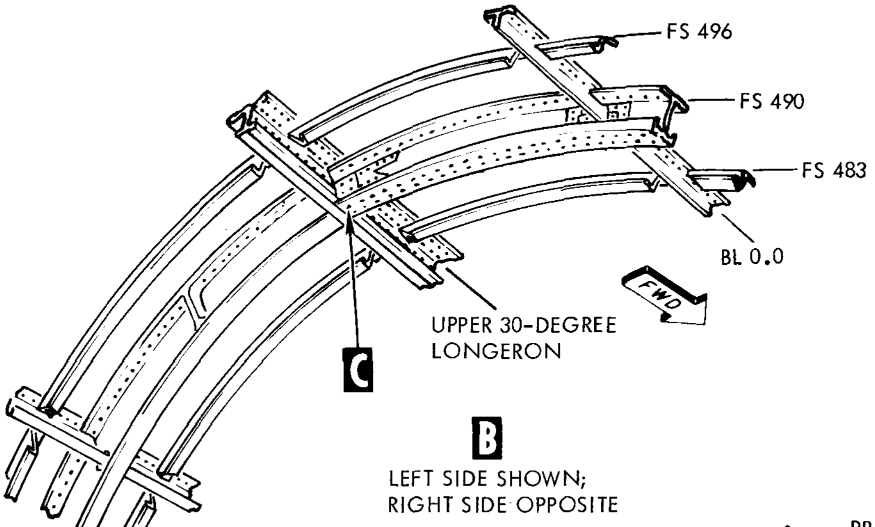
NOTE

1. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

⚠2 SHIELDED PENCIL PROBE - SCAN AROUND INDICATED FASTENERS IN A CLOCKWISE AND COUNTERCLOCKWISE DIRECTION.

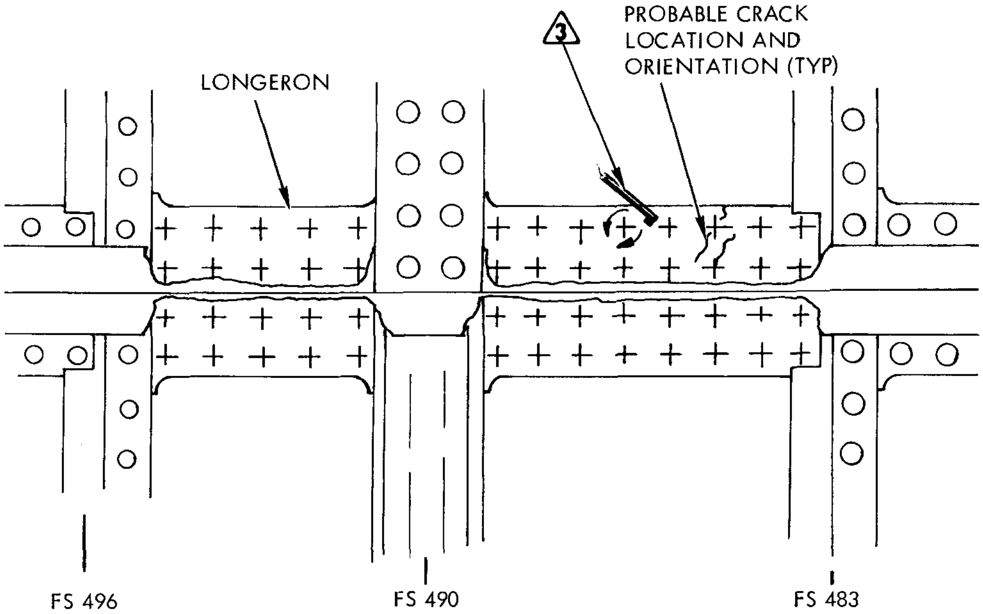
SMR151-58.4-0-012-1

Figure 58.4-12-4/4. Upper 30-Degree Longeron Attachment, FS 483 to FS 496 (Sheet 1 of 2)



B

LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE



C

UPPER 30-DEGREE LONGERON
ATTACHMENT, INTERNAL
INSPECTION AREA

NOTE

1. FASTENERS CODED "+" TO BE INSPECTED;
   FASTENERS CODED "0" FOR REFERENCE ONLY.

△3 90-DEGREE PROBE - SCAN AROUND INDICATED FASTENERS IN A CLOCKWISE AND COUNTER-CLOCKWISE DIRECTION.

SMR151-58.4-0-012-2

Figure 58.4-12-4/4. Upper 30-Degree Longeron Attachment, FS 483 to FS 496 (Sheet 2 of 2)

58.4.14 TOP LONGERON SPLICE AT BL 0, FS 490 TO FS 502.

58.4.14.1 DESCRIPTION. (See figure 58.4-13-6/4.) The top longeron assembly at BL 0 extends from FS 232 to FS 570 and includes a forward tee section, an aft tee section, and back-to-back angles. The tee sections are spliced to the angles at FS 402 and FS 495. The angles and tees are machined from 7075-T6 extrusions and finished with zinc chromate primer.

58.4.14.2 DEFECTS. Fatigue cracks may occur in the horizontal and/or vertical legs of the angles and tee at the aft longeron splice. Cracks will originate in fastener holes and propagate circumferentially in the horizontal legs and vertically in the vertical legs toward the edge of the parts.

Note

Primary NDE Procedures No. 1, No. 2 and No. 3 must be performed to complete this inspection.

58.4.14.3 PRIMARY NDE PROCEDURE NO. 1 RADIOGRAPHIC.

WARNING

RADIATION HAZARD
Ensure compliance with all applicable safety precautions set forth in Section 58.0 of this manual. Failure to comply may result in injury to personnel.

a. NDE equipment: X-ray Unit, Portable, 40-degree Tube Head, 150 KVP Magnaflux P/N MXR150A or equivalent.

b. Preparation of airplane:

(1) Lower or remove the overhead trim panel from the fuselage interior at FS 490 to 502.

(2) Pull the insulation blanket back from around the splice area.

c. Access: Access to the inspection area will be from inside the fuselage and from the top side of the fuselage with the use of appropriate maintenance stands.

d. Preparation of part: No special preparation required.

e. Instrument settings/calibration: Calibration not applicable. Set x-ray unit for each exposure in accordance with data in the Radiographic Inspection Data block illustrated. This procedure is established for a median film density of 2.0.

f. Inspection:

(1) Use film specified in the Radiographic Inspection Data block and place film in the sequence illustrated.

(2) Locate x-ray unit and center x-ray beam as illustrated for each exposure.

(3) Make radiographs of the inspection area in accordance with Radiographic Inspection Data as illustrated.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.4.14.4 PRIMARY NDE PROCEDURE NO. 2 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, Pencil, High Sensitivity, VM Products P/N VM102AS or equivalent.

(3) Calibration Standard, aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The inspection area is accessible from inside the airplane with the trim and insulation removed.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the splice surface as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the ED-520 crack detector.

Note

Eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Using the shielded pencil probe, scan around the six fasteners in the vertical legs of both angles of the longeron splice between FS 496 and FS 502. Scan in both clockwise and counterclockwise directions. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 3.

58.4.14.5 PRIMARY NDE PROCEDURE NO. 3 - ULTRASONICS.

a. NDE equipment:

(1) Portable Ultrasonic Flaw Detector, Sonic Mark I or equivalent.

(2) Transducer, 5.0 MHz, 70-degree Shear Wave, Mastercraft Industries, P/N HMT 7-123TC or equivalent.

(3) Cable, 6-foot, Microdot/BNC Connectors, Mastercraft Industries P/N HMT 7-41-BM or equivalent.

(4) Calibration Standard, Lockheed P/N PDL 990.

(5) Couplant, light oil.

b. Preparation of airplane: Same as Primary NDE Procedure No. 2.

c. Access: Same as Primary NDE Procedure No. 2.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the vertical leg of the tee just aft of the overlap of the splice as required to permit good contact between the part and transducer.

e. Instrument settings/calibration:

(1) Apply couplant and couple transducer to the vertical leg of calibration standard in Position No. 1 as illustrated.

(2) Refer to Section 58.1 of this manual for general calibration information and calibrate instrument to display the initial pulse near the left side of the CRT. Adjust sweep to display signal from the fastener hole near the center of the CRT.

(3) Slide the transducer to Position No. 2 and note that the sawcut signal is immediately to the right of the fastener hole signal.

(4) Maximize the signal from the sawcut by adjusting the transducer position and adjust the signal amplitude to obtain 80 percent saturation on the CRT display.

(5) Slide the transducer to Position No. 3 and note the loss of the fastener hole signal on the CRT display.

(6) The CRT signal from the sawcut simulates signals from cracks at the fastener hole.

Note

Attenuation by sealant, paint finishes, material differences, etc., may cause a difference in sensitivity between standard and part. This difference shall be determined and compensated for prior to inspection. To compensate for this difference, refer to instructions in Section 58.1 of this manual.

f. Inspection:

(1) Apply couplant to part and couple transducer to inspection area as illustrated.

(2) Scan above and below the aft most fastener hole in the vertical leg of the splice.

(3) Any signal received similar to and at approximately the same location on the CRT display as the sawcut signal observed during calibration will indicate a probable crack in the part.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform Backup NDE Procedure. If no defect is suspected, proceed to System Securing.

58.4.14.6 BACKUP NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 5/32-inch diameter, VM Products P/N VM101B or equivalent.

(3) Probe, Bolt Hole, 3/16-inch diameter, VM Products P/N VM101B or equivalent.

(4) Calibration Standard, aluminum (see figure 58.1-9).

b. Preparation of airplane: Same as Primary NDE Procedure No. 2.

c. Access: Same as Primary NDE Procedure No. 2.

d. Preparation of part:

(1) Remove the fastener from the suspected defective fastener hole.

(2) Refer to Section 58.0 for cleaning instruction. Remove sealant and clean the suspected defective hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

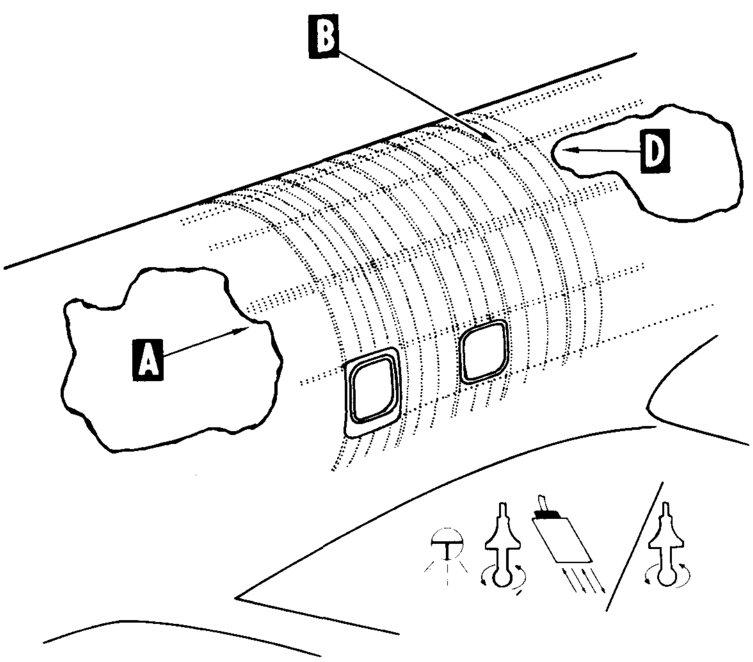
Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the suspected defective fastener hold through each layer of material. Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

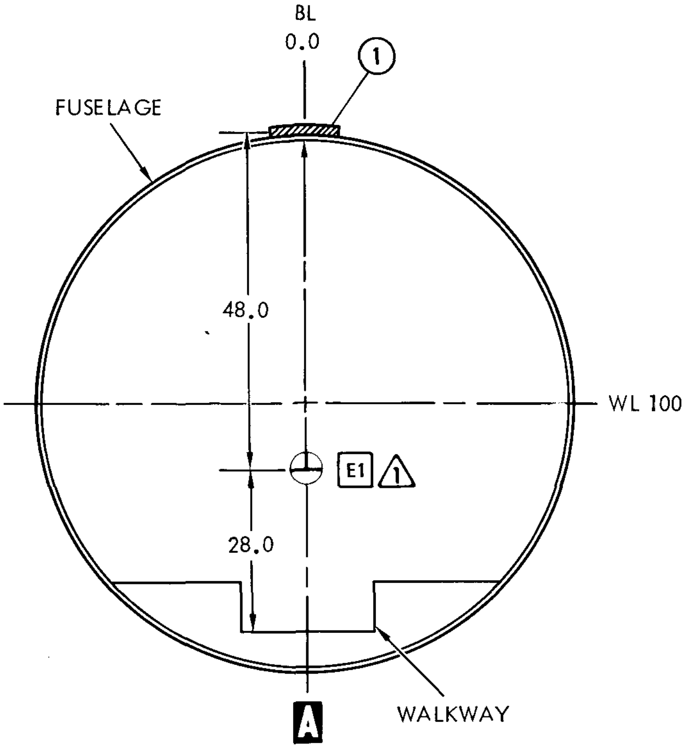
g. Mark and report indicated defects.

58.4.14.7 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



NOTE

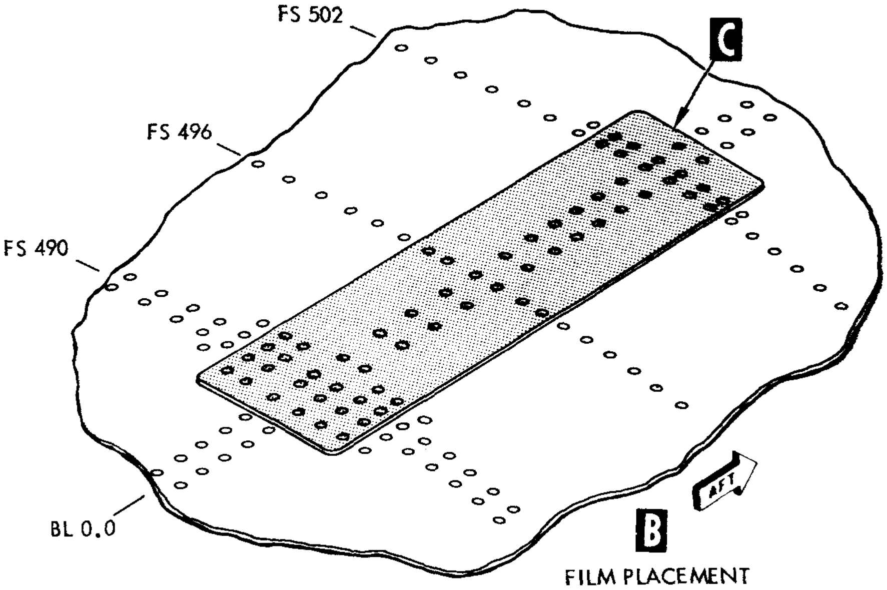
△1 POSITION TUBE SO X-RAY BEAM IS PERPENDICULAR TO AND AIMED AT CENTER OF FILM, PRIMARY NDE PROCEDURE NO. 1.

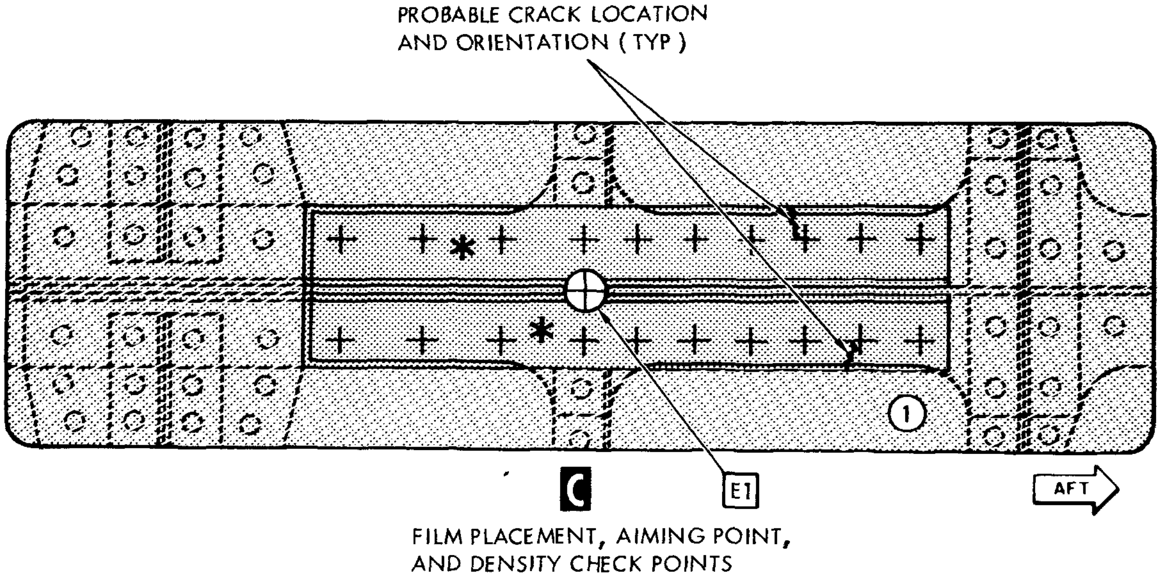


VIEW LOOKING AFT
X-RAY TUBE LOCATION

SMR 151-58.4-0-013-1

Figure 58.4-13-6/4. Top Longeron Splice at BL0, FS 490 to FS 502 (Sheet 1 of 5)




PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
1
C
E1
AFT
FILM PLACEMENT, AIMING POINT, AND DENSITY CHECK POINTS

NOTE

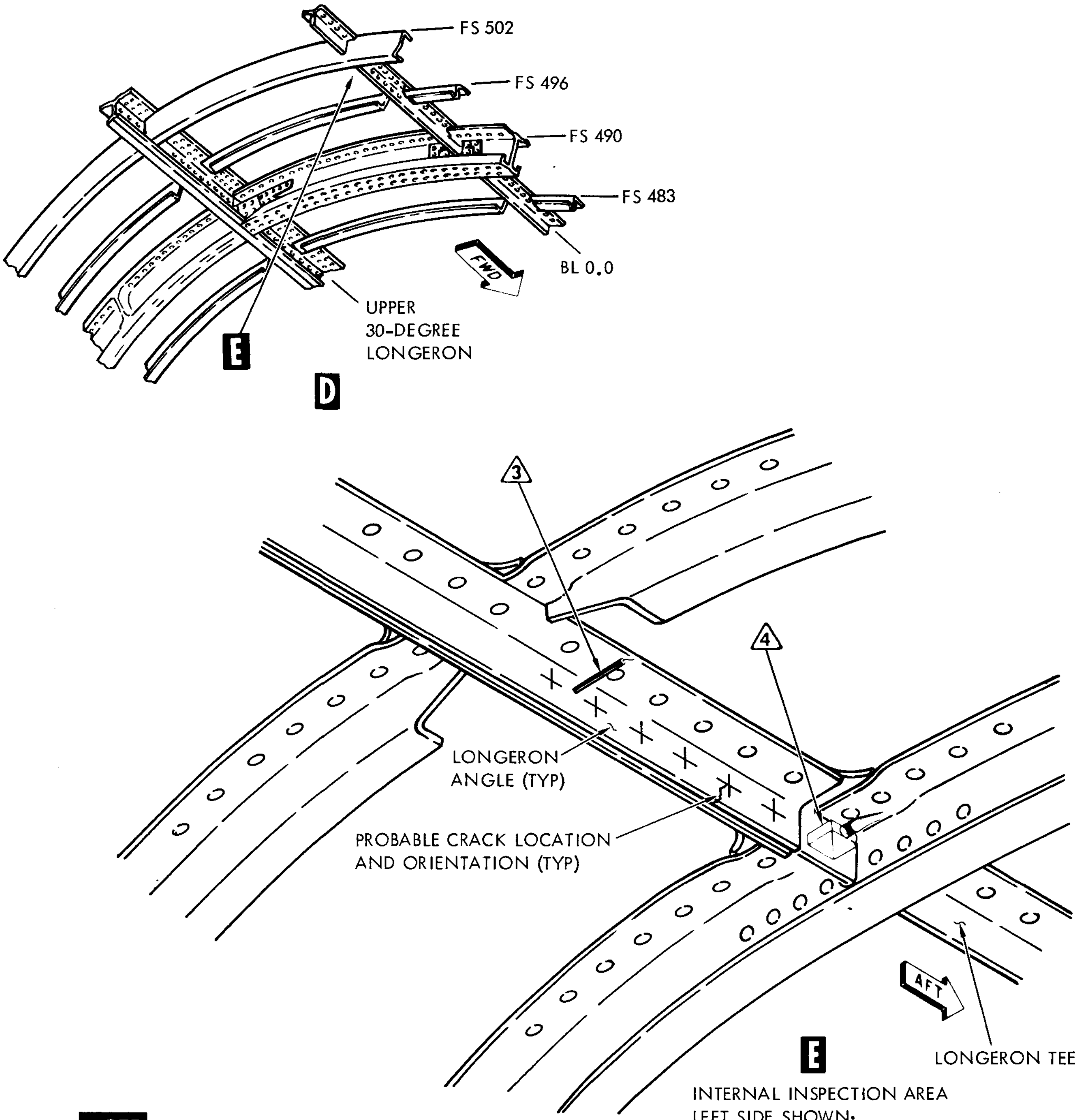
* CHECK FOR MEDIAN FILM DENSITY ON AT LEAST ONE OF THE FILMS.

2. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

RADIOGRAPHIC INSPECTION DATA							
EXPOSURE NUMBER	KV	MA	FFD(INCHES)	TIME (SEC)	FILM		REMARKS
					TYPE	SIZE	
E1	100	5	48	90	NDT 45/ NDT 55	4.5 x 17	EXPOSE ①

SMR 151-58.4-0-013-2

Figure 58.4-13-6/4. Top Longeron Splice at BL0, FS 490 to FS 502 (Sheet 2 of 5)



INTERNAL INSPECTION AREA
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

NOTE

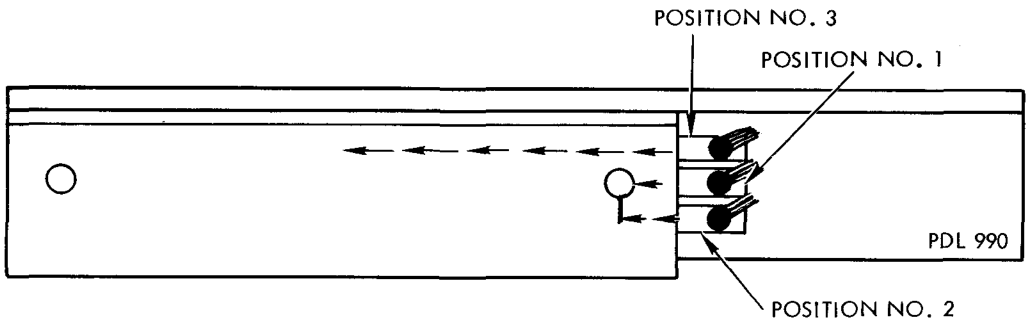
2. FASTENERS CODED "+" TO BE INSPECTED; FASTENERS CODED "0" FOR REFERENCE ONLY.

△3 SHIELDED PENCIL PROBE - SCAN AROUND INDICATED FASTENERS IN LONGERON ANGLES, LEFT AND RIGHT, PRIMARY NDE PROCEDURE NO. 2.

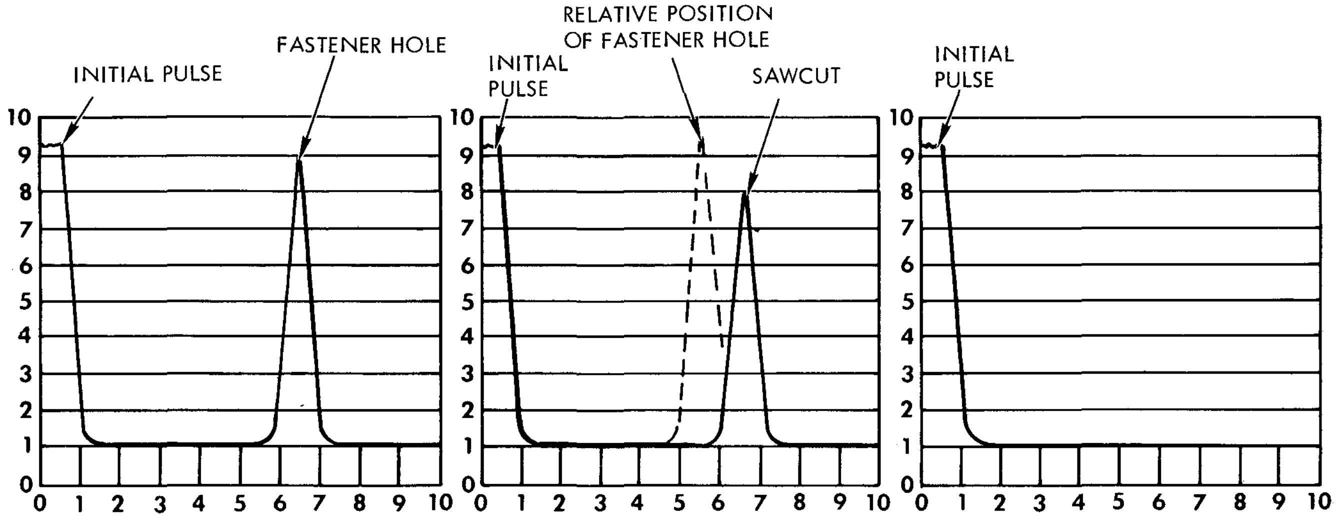
△4 SHEAR WAVE TRANSDUCER - SCAN ABOVE AND BELOW AFT MOST INDICATED FASTENER IN LONGERON TEE, PRIMARY NDE PROCEDURE NO. 3.

SMR 151-58.4-0-013-3

Figure 58.4-13-6/4. Top Longeron Splice at BL0, FS 490 to FS 502 (Sheet 3 of 5)



CALIBRATION SETUP
PRIMARY NDE PROCEDURE NO. 3

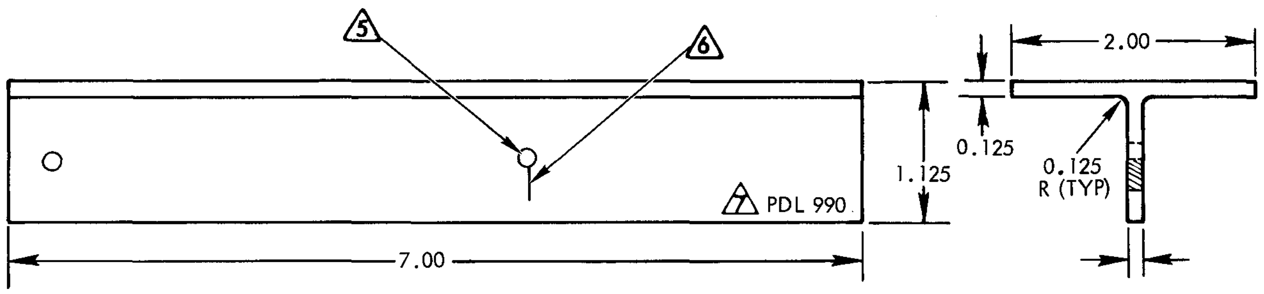


POSITION NO. 1 POSITION NO. 2 POSITION NO. 3

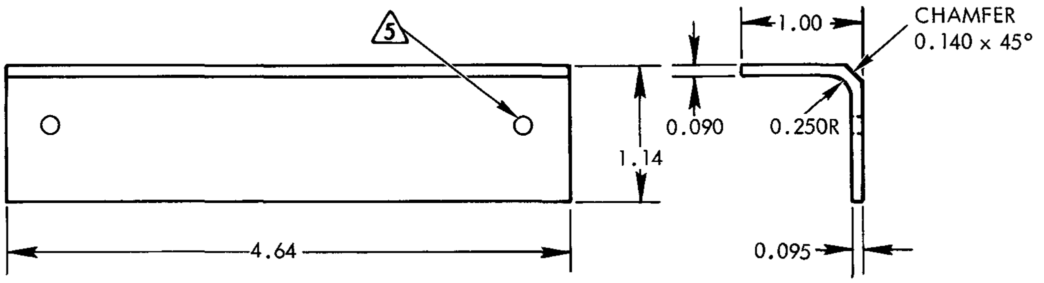
TYPICAL CRT PRESENTATIONS
PRIMARY NDE PROCEDURE NO. 3

SMR151-58.4-0-013-4

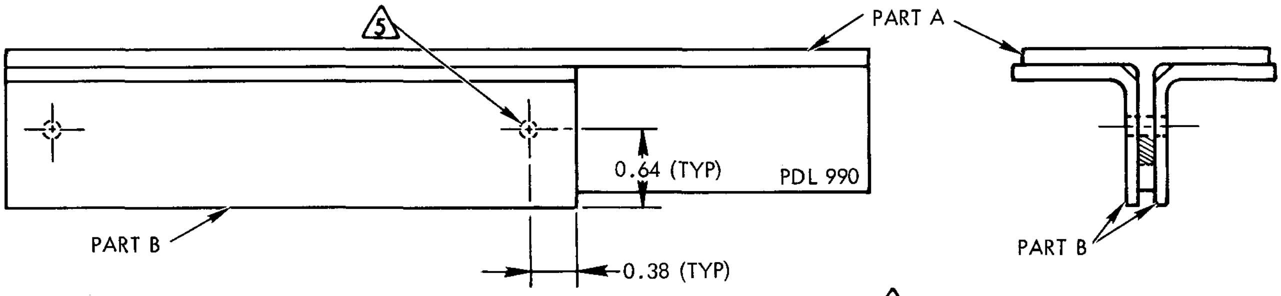
Figure 58.4-13-6/4. Top Longeron Splice at BLO, FS 490 to FS 502 (Sheet 4 of 5)



CALIBRATION STANDARD 8
PART A, 7075-T6 ALUMINUM



CALIBRATION STANDARD 8
PART B, 7075-T6 ALUMINUM
(2 REQ'D)



CALIBRATION STANDARD ASSEMBLY 9

NOTE

5 MATCH DRILL 0.187 ± 0.005 DIAMETER HOLE THROUGH PARTS A AND B (2 PLACES).

6 SAWCUT - 0.25 THRU, PART A ONLY. FILL SAWCUT WITH CORRECTION FLUID (WHITE OPAQUE) BEFORE ASSEMBLY.

7 SCRIBE ON PART.

8 SULFURIC ACID ANODIZE AND FINISH WITH TWO COATS OF ZINC CHROMATE PRIMER.

9 FAY SURFACE SEAL WITH MIL-S-8802 SEALANT AND ASSEMBLE WITH WET INSTALLED HL54-5-4 PINS AND HL70-5 COLLARS OR EQUIVALENT.

SMR151-58.4-0-013-5

Figure 58.4-13-6/4. Top Longeron Splice at BLO, FS 490 to FS 502 (Sheet 5 of 5)

SECTION 58.5

LANDING GEAR GROUP

Contents Page

GENERAL DESCRIPTION 58-161

Inspection Procedures 58-161

58.5.1 GENERAL DESCRIPTION.

The landing gear group consists of two main gear assemblies and one nose gear assembly. Each main gear assembly has three outboard doors and one inboard door and the nose gear assembly has two forward doors and one aft door. See Chapters 32 and 52 for complete description of the landing gear strut assemblies, doors, and detail components.

58.5.1.1 INSPECTION PROCEDURES. Landing Gear Group areas and components to be inspected are tabulated in the Landing Gear Group Inspections Index (See Table 58.5-1). Each inspection procedure has been assigned an inspection index number which is cross referenced to the relevant paragraph and page number.

Table 58.5-1. Landing Gear Group Inspection Index

INDEX NO.	INSPECTION TITLE	PARAGRAPH NO.	PAGE NO.
L-1	NLG Upper Link Assembly	58.5.2	58-162
L-2	NLG and MLG Piston Assemblies	58.5.3	58-165
L-3	NLG and MLG Strut Cylinders	58.5.4	58-169
L-4	NLG and MLG Wheels	58.5.5	58-176

58.5.2. NLG UPPER LINK ASSEMBLY.

58.5.2.1. DESCRIPTION. (See figure 58.5-1-4/2.) The Nose Landing Gear upper link assembly is fabricated from a 7079 aluminum alloy forging and finished with epoxy primer, epoxy enamel, and Corogard 14.

58.5.2.2. DEFECTS. Stress corrosion cracks will originate in the drag link holes and propagate in any direction.

58.5.2.3. PRIMARY NDE PROCEDURE - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 5/16-inch diameter, Ideal Specialty Co. P/N 6200-5/16-BH-BNC or equivalent.

(3) Probe, Bolt Hole, 5/8-inch diameter, Ideal Specialty Co. P/N 6200-5/8-BH-BNC or equivalent.

(4) Probe, Shielded, V.M. Products P/N VM202-1/8 or equivalent.

(5) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane.

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

WARNING

Ensure that the Nose Landing Gear ground lock pin is installed before performing this inspection. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual and check that the Nose Landing Gear ground lock pin is installed.

(2) Remove the screw and sleeve securing the retainer assembly to the upper link lug and disconnect the upper latch rod from the rocker assembly. Rotate the rod and retainer sufficiently to clear the retainer attachment lug.

(3) Remove the down lock switch assembly from the down lock switch support lug.

(4) Remove the upper end of the NLG linkage spring assembly from the NLG upper link.

c. Access: The inspection area is accessible with the airplane in the normal parked condition.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the fastener holes and surface areas of the NLG upper link as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Scan the entire inner surface of the fastener hole in each of the retainer attachment lugs and the switch support lug. Surface scan around the bolt in the linkage attach lever and the fasteners at the lower end of the NLG upper link assembly. Refer to Section 58.1 of this manual for bolt hole scanning of straight wall fastener holes and surface scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

h. To confirm suspected defects, perform the backup NDE Procedure. If no defect is suspected, proceed to system securing.

58.5.2.4. BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane: Same as for Primary NDE Procedure.

c. Access: Same as for Primary NDE Procedure.

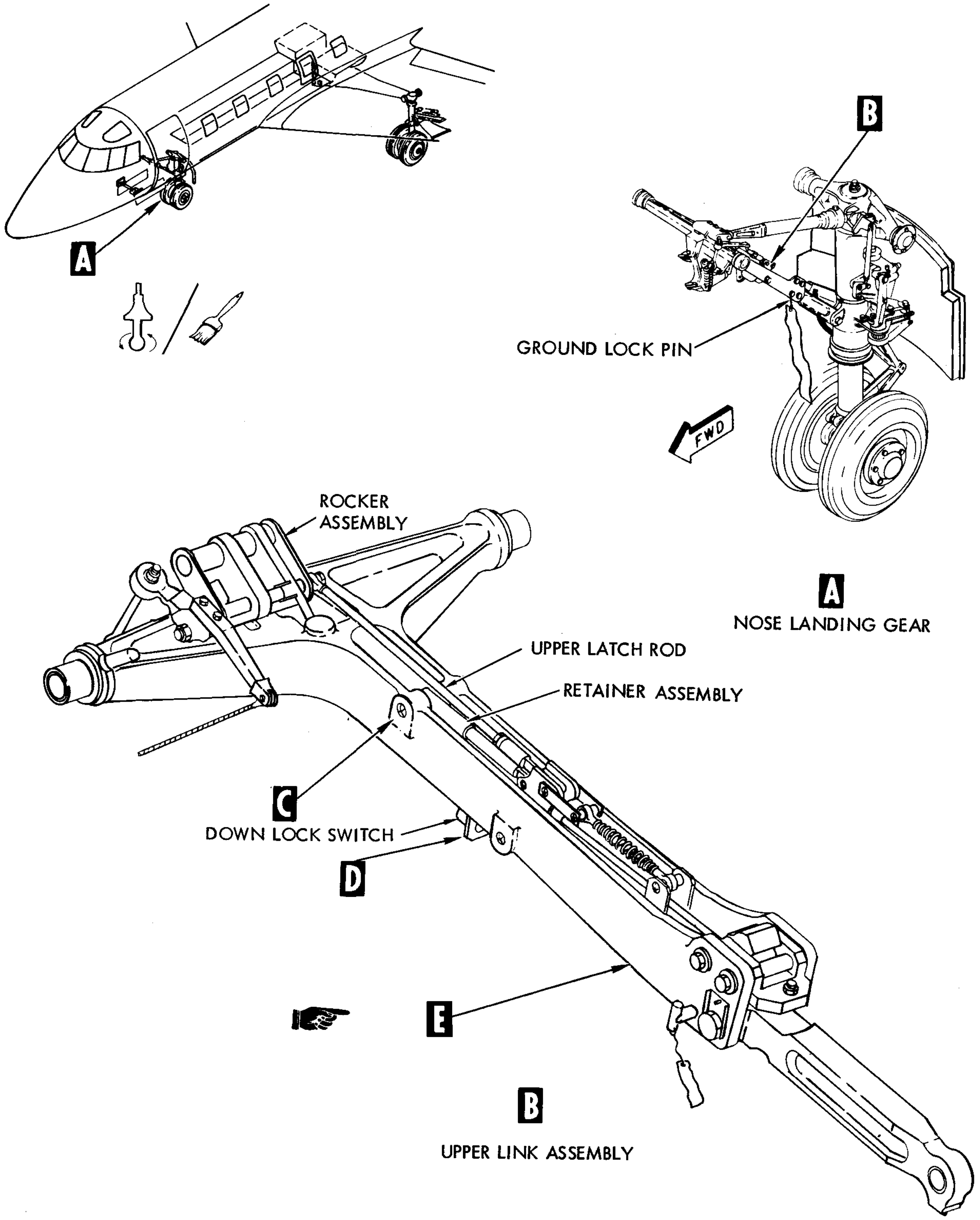
d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean surfaces of both sides of suspected defective area.

e. Instrument Settings/Calibration: Not applicable.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-3.

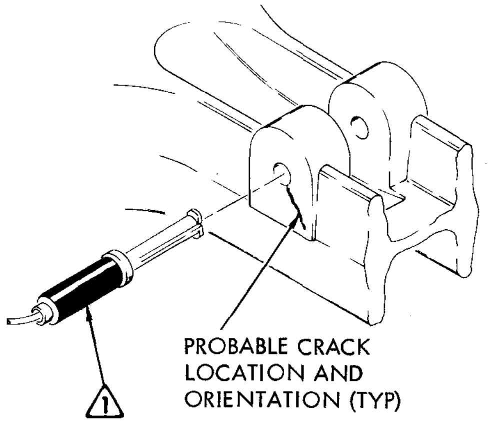
g. Mark and report indicated defects.

58.5.2.5. SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



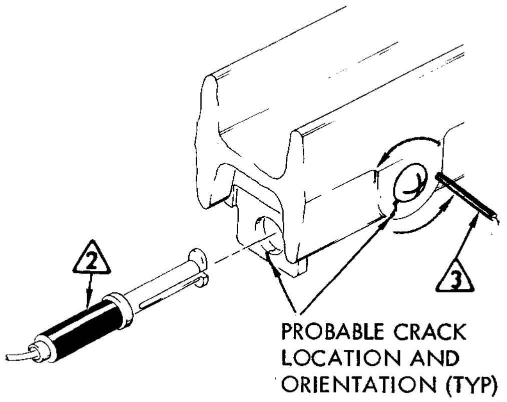
SMR151-58.5-11-001-1

Figure 58.5-1-4/2. NLG Upper Link Assembly (Sheet 1 of 2)



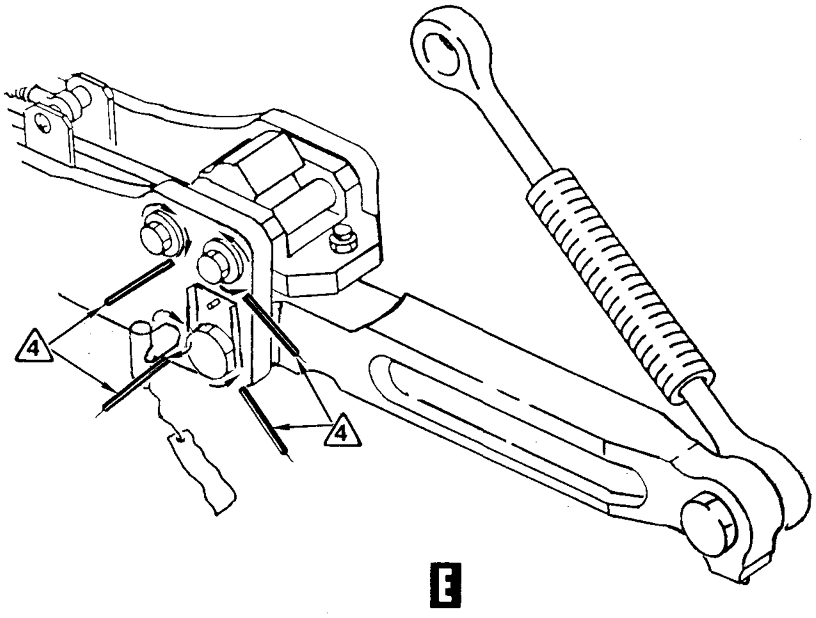
C

RETAINER ATTACHMENT
LUG INSPECTION



D

DOWN LOCK SWITCH SUPPORT LUG
AND LINKAGE LEVER ATTACHMENT
INSPECTION



E

LEFT SIDE SHOWN
RIGHT SIDE OPPOSITE

NOTE

1 5/16-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF FASTENER HOLES IN BOTH LUGS.

2 5/8-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF SWITCH SUPPORT HOLE.

3 EDDY CURRENT SURFACE SCAN AROUND THE LINKAGE LEVER ATTACHMENT BOLT IN BOTH DIRECTIONS ON BOTH LEFT AND RIGHT SIDES OF THE LINK ASSEMBLY.

4 EDDY CURRENT SURFACE SCAN AROUND FASTENERS IN BOTH DIRECTIONS ON BOTH LEFT AND RIGHT SIDES OF THE LINK ASSEMBLY.

SMR151-58-5-11-001-2

Figure 58.5-1-4/2. NLG Upper Link Assembly (Sheet 2 of 2)

58.5.3 NLG AND MLG PISTON ASSEMBLIES.

58.5.3.1 DESCRIPTION. (See figure 58.5-2-3/0.) The Nose and Main Landing Gear piston assemblies are fabricated from 4340 heat treated steel. The piston outer surface in the pressure weld area is chrome plated while the inner surface in the pressure weld area is as-welded. The radiused shoulder where the upper axle surface joins the piston tube and the radiused relief of the axle hubs are titanium-cadmium plated and finished with epoxy primer, epoxy enamel, and Corogard 14.

58.5.3.2 DEFECTS. Fatigue loading may cause circumferential cracks to initiate in the pressure weld areas, the radiused shoulder where the upper axle surface joins the piston tube, and the radiused relief at the base of the axle hub.

Note

Both Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.5.3.3 PRIMARY NDE PROCEDURE NO. 1 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the Nose and Main Landing Gear wheels and separate the piston assemblies from the strut assemblies in accordance with Section 32.0 of this manual.

c. Access: Access to the inspection areas requires removal of the Nose and Main Landing Gear wheels and separation of the piston assemblies from the struts.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish as necessary from the inspection areas on the outer surfaces and clean the inspection areas on the inner and outer surfaces of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current for longitudinal field as follows:

(1) NLG Piston Assembly

(a) Piston - 1800 amperes

(b) Axle - 2400 amperes

(2) MLG Piston Assembly

(a) Piston - 2250 amperes

(b) Axle - 2000 amperes

f. Inspection: Inspect the Nose and Main Landing Gear piston assemblies completely around the piston and axle for circumferential cracks by the magnetic particle wet continuous method using longitudinal magnetic field. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

(1) Using the current settings given, inspect each piston a minimum of 15 inches from the centerline of the axle. Special attention is to be given to the pressure weld area.

(2) Using the current settings given, inspect each side of each axle. Special attention is to be given to the radiused areas.

(3) Demagnetize parts after completion of each inspection.

g. Mark and report indicated defects. If no defect is indicated, proceed to Primary NDE Procedure No. 2.

58.5.3.4 PRIMARY NDE PROCEDURE NO. 2 - VISUAL.

a. NDE equipment:

(1) Borescope, 0.394 O.D. x 34.53 working length, right angle, Lenox P/N L309MC or equivalent.

58.5.3.4 (Cont)

(2) Cable, 72 inch, Lenox P/N CL1G18 or equivalent.

(3) Light, Lenox P/N 6127-2 or equivalent.

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

c. Access: Same as for Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the inspection areas on the inner surfaces as required.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect the internal surfaces of the Nose and Main Landing Gear piston assemblies for pitting and circumferential cracks. Special attention is to be given to the pressure weld area of the pistons and to areas opposite the radius areas of the axles.

g. Mark and report indicated defects.

58.5.3.5 BACKUP NDE PROCEDURE. None required.

58.5.3.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.

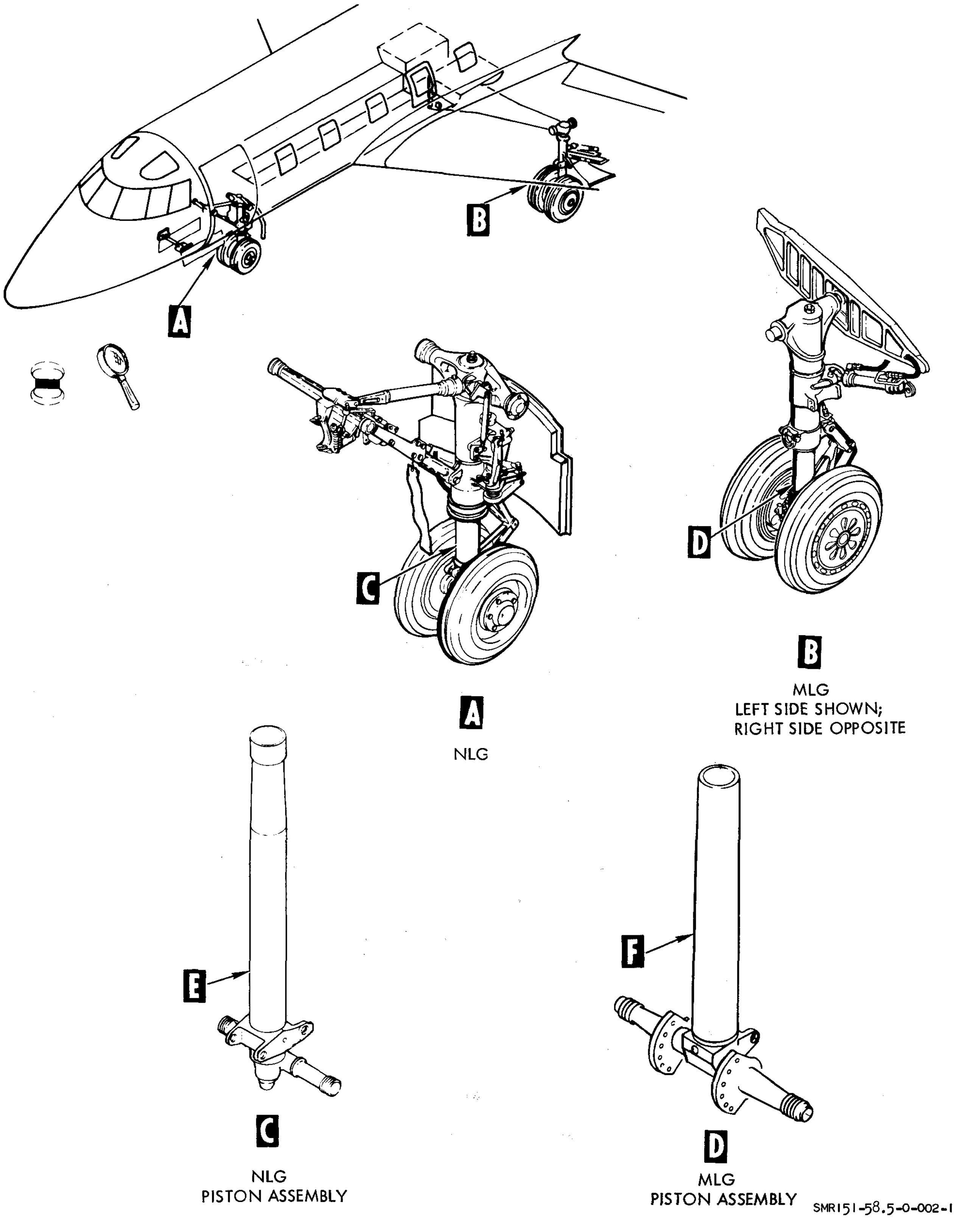
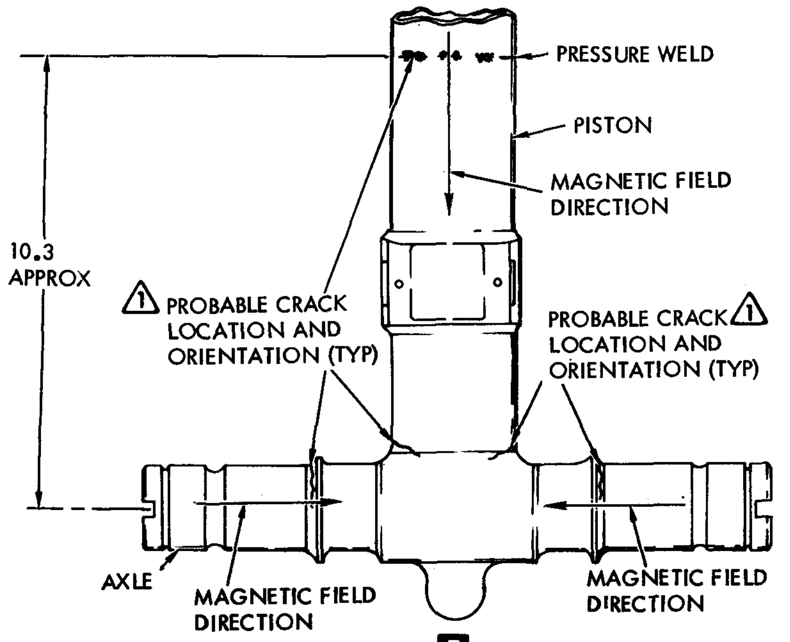


Figure 58.5-2-3/0. NLG and MLG Piston Assemblies (Sheet 1 of 2)

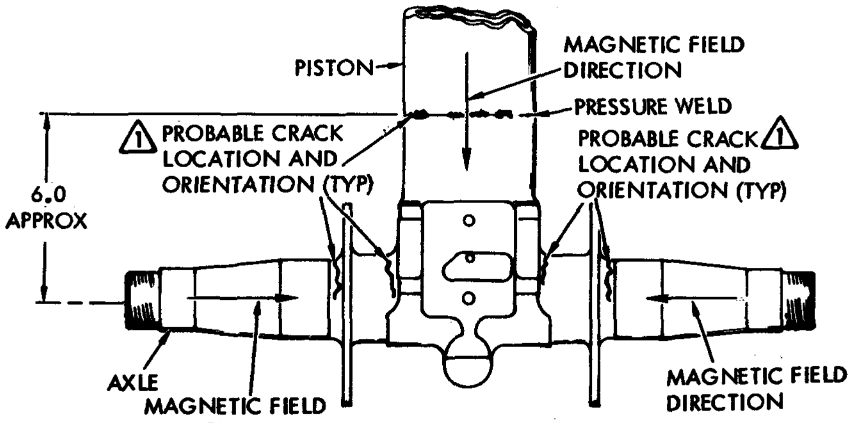


E

NLG PISTON ASSEMBLY
INSPECTION AREAS

NOTE

1 PROBABLE CRACK LOCATION EXTENDS COMPLETELY AROUND THE PISTON AND AXLE.



F

MLG PISTON ASSEMBLY
INSPECTION AREAS

SMR151-58.5-0-002-2

Figure 58.5-2-3/0. NLG and MLG Piston Assemblies (Sheet 2 of 2)

58.5.4 NLG AND MLG STRUT CYLINDERS.

58.5.4.1 DESCRIPTION. (See figure 58.5-3-4/2.) The aircraft has two main strut assemblies and one nose strut assembly. The struts are fabricated from either 7079-T6 or 7049-T73 aluminum alloy. The bore surfaces of the struts are hard anodized and the exterior surfaces are finished with epoxy primer, epoxy enamel, and Corogard 14.

58.5.4.2 DEFECTS. Stress corrosion cracks may occur along the entire length of the bore. Also cracks may occur in the strut boss hole, or on the interior and exterior surfaces around the boss hole.

Note

Both Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.5.4.3 PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1.252-inch diameter, Ideal Specialty Co. P/N 6200-1.252-BH-BNC or equivalent.

(3) Bore inspection assembly equipment listed in the Lockheed Aeronautical Systems Company JSK-7-INST/M instruction manual.

(4) Nose landing gear probe holder assembly, Lockheed P/N JSK-7-201.

(5) Main landing gear probe holder assembly, Lockheed P/N JSK-7-702.

(6) Cable, 8-foot, BNC/right angle 3 conductor phone plug, Lockheed P/N JSK-7-2001.

(7) Calibration standard, aluminum. (See figure 58.1-9.)

(8) Allen wrench, 3/16-inch size, local purchase.

(9) Allen wrench, 1/8-inch size, local purchase.

(10) Carrying case, Lockheed P/N JSK-7-600.

(11) Probe Guide Assembly - Bolt Hole, Lockheed P/N JSK-7-501 through -504.

(12) Probe, Surface, 1/4-inch diameter x 1 inch long, Ideal Specialties P/N 6100-1/4-S x 1.00 long with 8-foot cable.

(13) Probe Guide Assembly - Surface, Lockheed P/N JSK-7-506 through -511.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Disassemble the MLG and NLG shock strut assemblies, removing the piston assemblies from the cylinders in accordance with Section 32.0 of this manual. The cylinders may be allowed to remain attached with the aircraft on jacks for this inspection.

c. Access: Access to the inspection areas requires removal of the MLG and NLG piston assemblies from the struts.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the interior surface of the cylinder bore, the strut boss hole and the area around the strut boss hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

f. Inspection:

(1) NLG Bore:

Note

To prevent probe damage, do not allow the probe to extend beyond the rubber bumper on the probe holder.

(a) Insert the 1.252-inch diameter bolt hole probe into the nose landing gear probe holder P/N JSK-7-201 and attach the assembly to the lead screw on the bore inspection drive assembly. The probe should be pointing toward the boss hole when inserted in the bore and slightly shorter than the rubber bumper.

(b) Position the strut adapter screw on the lead screw to permit the inspection scan to begin near the top of the cylinder bore.

(c) Insert the lead screw with the probe holder assembly into the cylinder and screw the strut adapter screw to attach the drive assembly to the strut.

(d) Index the bore drive assembly to the top of the cylinder.

(e) Scan the inner surface of the bore from the top down approximately 8 inches.

(f) Remove the bore drive assembly and reverse the bolt hole probe direction in the holder and repeat steps (a) through (d).

58.5.4.3 (Cont)

(g) Scan the inner surface of the bore down to the keyway.

(2) MLG Bore: Using the MLG probe holder P/N JSK-7-202 repeat steps f.(1)(a) through f.(1)(g) to inspect the MLG bore from the top down to where the cylinder diameter increases.

(3) NLG and MLG Boss:

(a) Insert the 1.252 inch diameter bolt hole probe into the probe guide assembly P/N JSK-7-501 through -504.

(b) Using the probe guide assembly, scan the entire inner surface of the NLG and MLG boss hole.

(4) NLG and MLG surfaces around Boss Hole:

Note

The wear face of the probe should extend approximately 0.010 inch beyond the face of the probe holder to provide good contact between the part and probe.

(a) Insert the 1/4 inch diameter surface probe into the Probe Guide Assembly P/N JSK-7-506 through -511.

Note

The extension handle may be removed for inspection of the boss hole exterior surface.

(b) Using the probe guide assembly, scan the interior and exterior surfaces around the NLG and MLG boss hole.

(5) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report suspected defects.

h. To confirm suspected defects, perform the Backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.5.4.4 PRIMARY NDE PROCEDURE NO. 2 - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. The hard anodized surface on the interior of the cylinder is not to be stripped.

(1) NLG - Clean the interior surface of the cylinder from the top of the keyway to the open end.

(2) MLG - Clean the interior surface of the cylinder from the change in diameter to the open end.

e. Instrument settings/calibration: Not applicable.

Note

Hard anodized surfaces of the cylinder may produce spider web indications with the penetrant inspection. Such indications should not be interpreted as cracks.

f. Inspection: Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

(1) NLG - Inspect the cylinder from the top of the keyway to the open end.

(2) MLG - Inspect the cylinder from the change in diameter to the open end.

g. Mark and report indicated defects. If no defects are indicated, proceed to System Securing.

58.5.4.5 BACKUP NDE PROCEDURE - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment from approved list in figure 58.0-4.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: Same as Primary NDE Procedure No. 1.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. The hard anodized surface on the interior of the cylinder is not to be stripped. Clean surface of the suspected defective area.

e. Instrument settings/calibration: Not applicable.

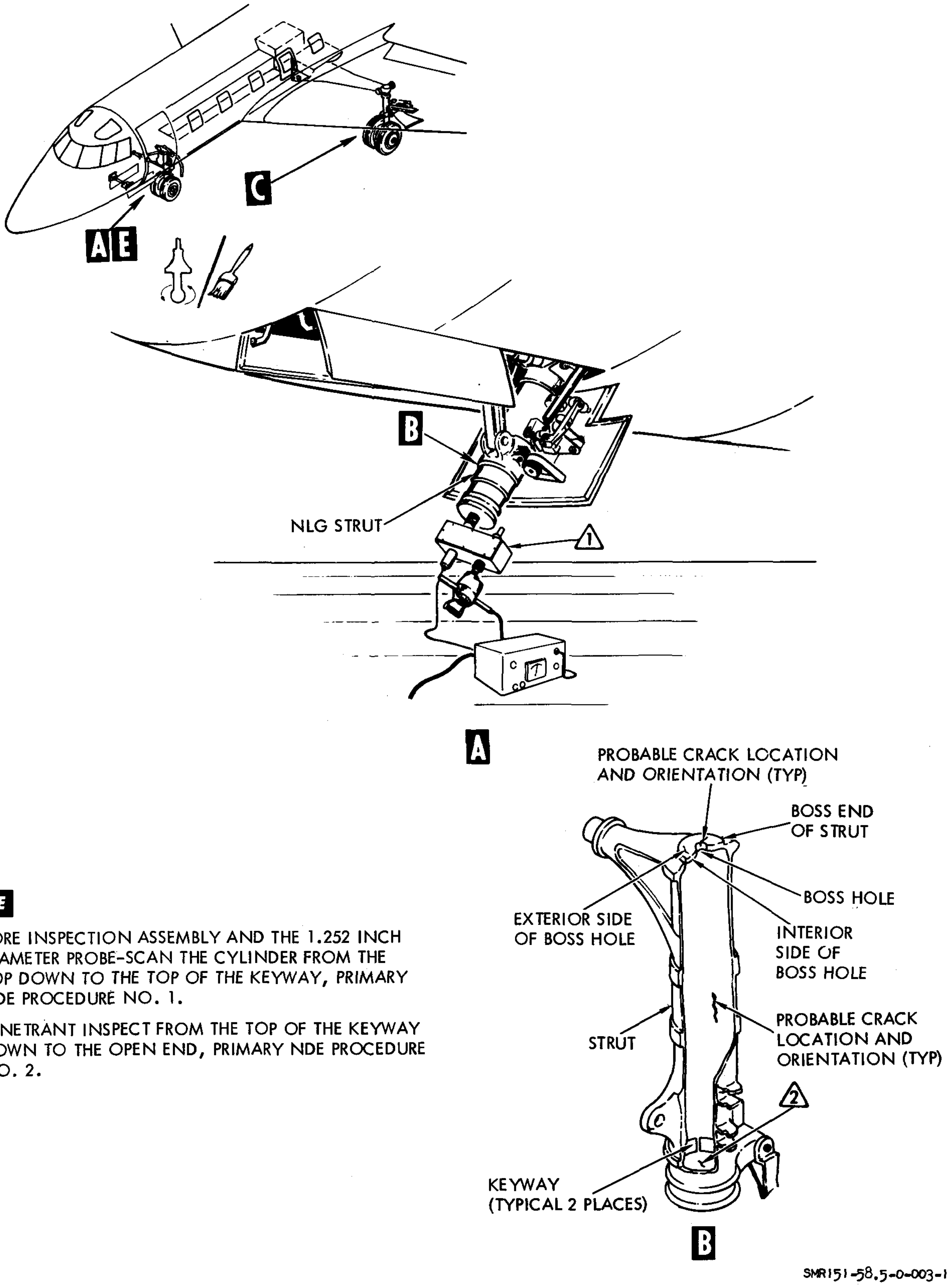
Note

Hard anodized surfaces of the cylinder may produce spider web indications with the penetrant inspection. Such indications should not be interpreted as cracks.

f. Inspection: Inspect only the suspected defective areas. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects.

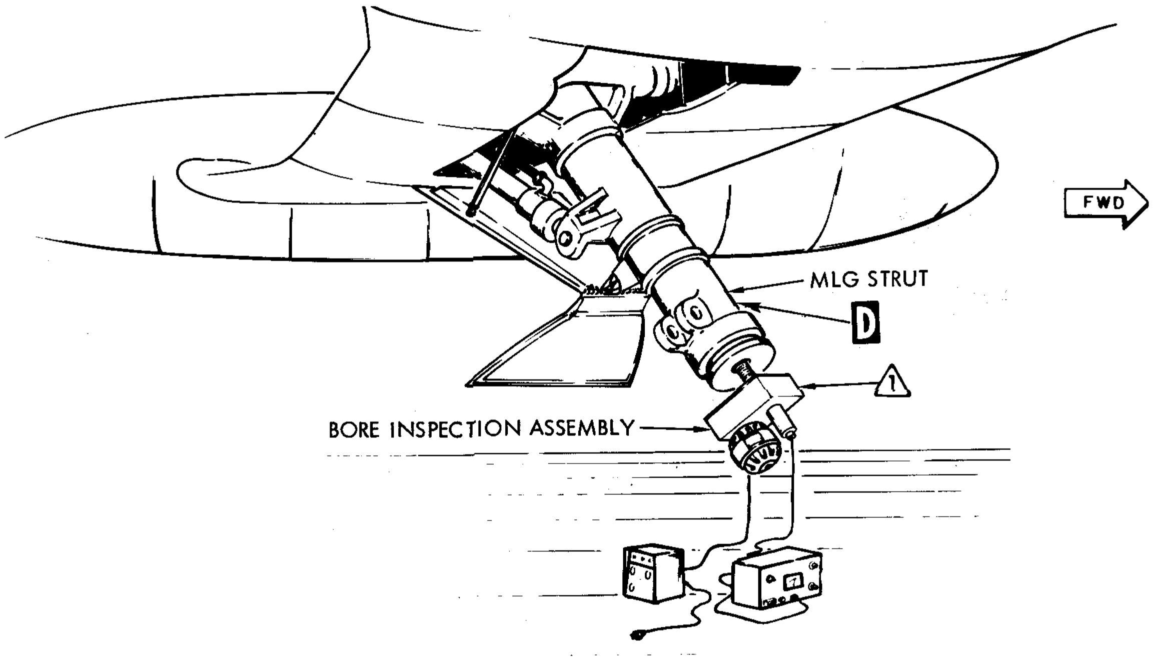
58.5.4.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


C
A E
B
NLG STRUT
1
A
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
BOSS END OF STRUT
BOSS HOLE
EXTERIOR SIDE OF BOSS HOLE
INTERIOR SIDE OF BOSS HOLE
STRUT
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
2
KEYWAY (TYPICAL 2 PLACES)
B
SMR151-58.5-0-003-1

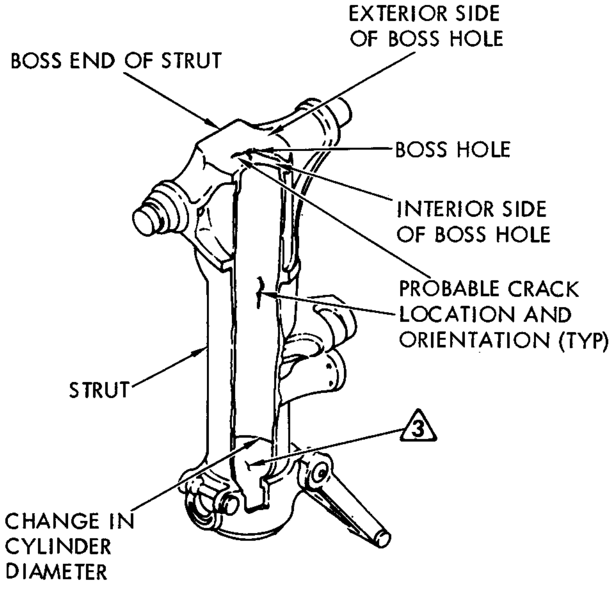
NOTE

1. BORE INSPECTION ASSEMBLY AND THE 1.252 INCH DIAMETER PROBE-SCAN THE CYLINDER FROM THE TOP DOWN TO THE TOP OF THE KEYWAY, PRIMARY NDE PROCEDURE NO. 1.
2. PENETRANT INSPECT FROM THE TOP OF THE KEYWAY DOWN TO THE OPEN END, PRIMARY NDE PROCEDURE NO. 2.

Figure 58.5-3-4/2. NLG and MLG Strut Cylinders (Sheet 1 of 5)



C


EXTERIOR SIDE OF BOSS HOLE
BOSS END OF STRUT
BOSS HOLE
INTERIOR SIDE OF BOSS HOLE
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
STRUT
3
CHANGE IN CYLINDER DIAMETER

D

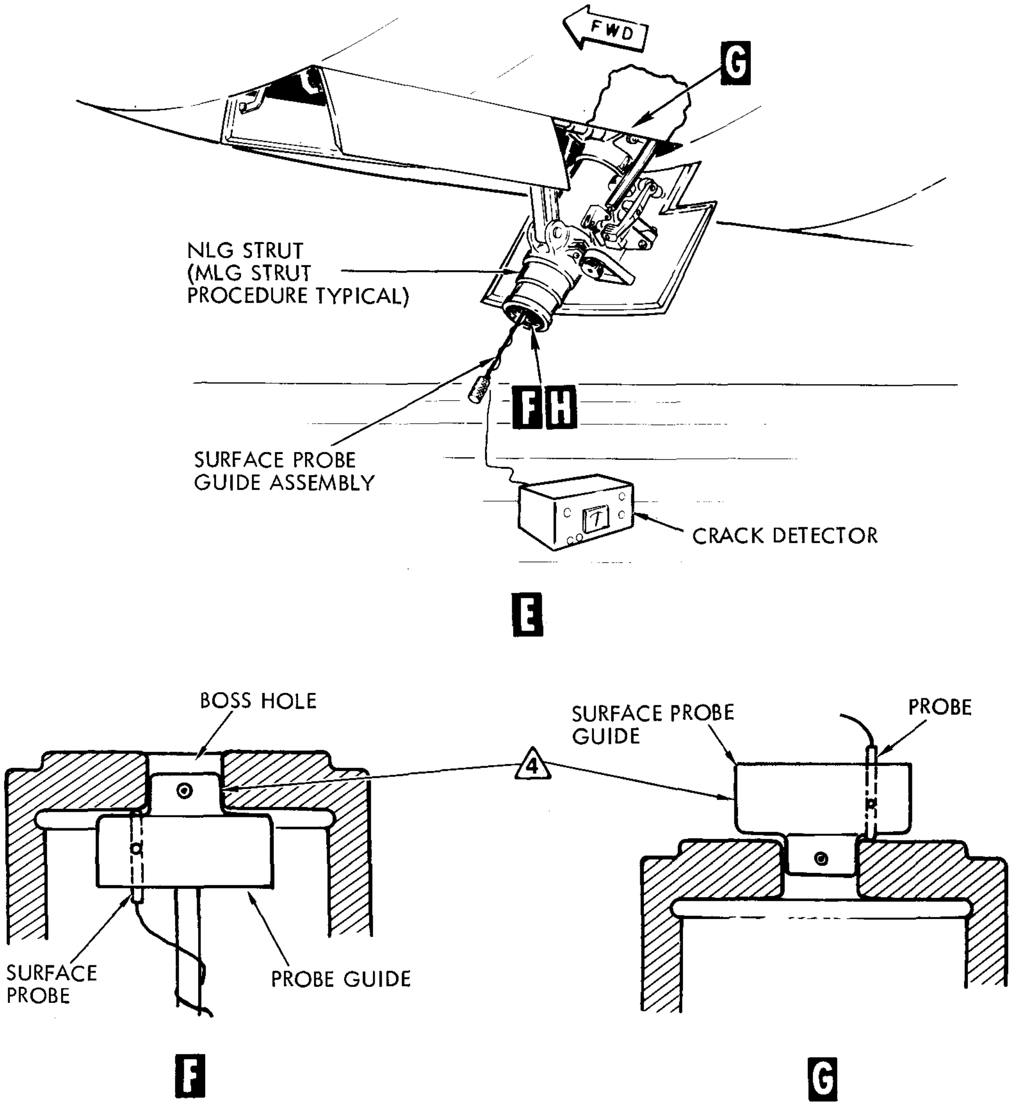
NOTE

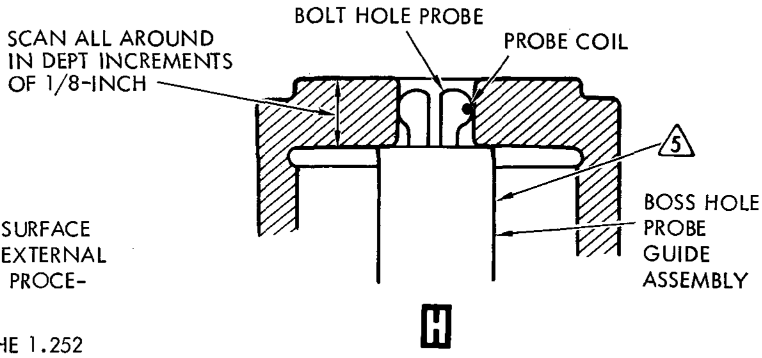
⚠1 BORE INSPECTION ASSEMBLY AND THE 1.252 INCH DIAMETER PROBE-SCAN THE CYLINDER FROM THE TOP DOWN TO THE TOP OF THE KEYWAY, PRIMARY NDE PROCEDURE NO. 1.

⚠3 PENETRANT INSPECT FROM THE CHANGE IN CYLINDER DIAMETER TO THE OPEN END, PRIMARY NDE PROCEDURE NO. 2.

SMR151-58.5-0-003-2

Figure 58.5-3-4/2. NLG and MLG Strut Cylinders (Sheet 2 of 5)





NOTE

4 SURFACE PROBE GUIDE ASSEMBLY AND THE SURFACE PROBE-SCAN AROUND THE INTERNAL AND EXTERNAL SURFACE OF THE BOSS HOLE, PRIMARY NDE PROCEDURE NO. 1.

5 BOLT HOLE PROBE GUIDE ASSEMBLY AND THE 1.252 BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF BOSS HOLE, PRIMARY NDE PROCEDURE NO. 1.

SMR151-58.5-0-003-3

Figure 58.5-3-4/2. NLG and MLG Strut Cylinders (Sheet 3 of 5)

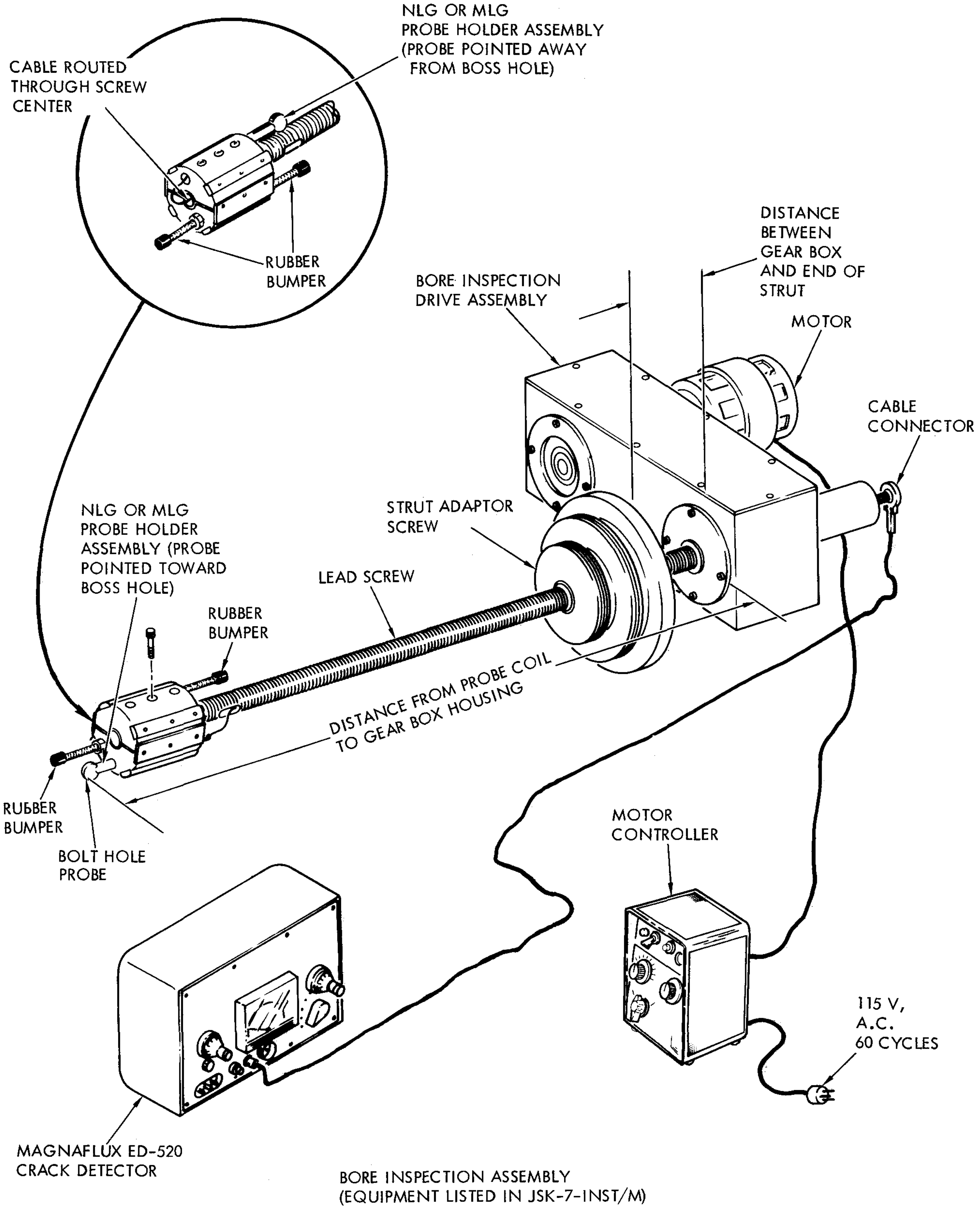
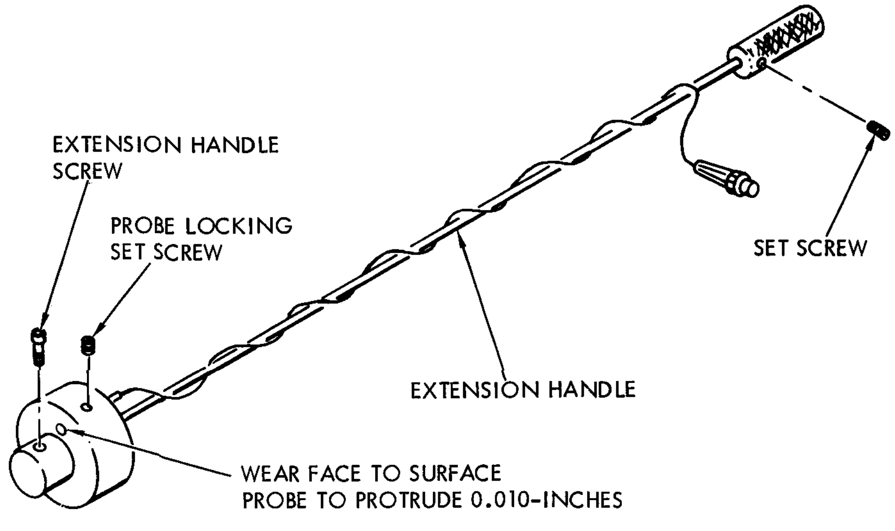
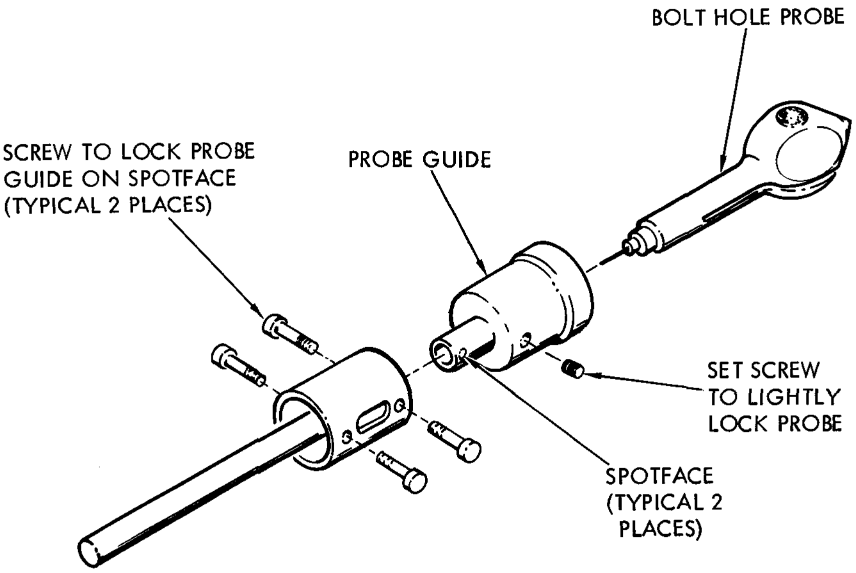

NLG OR MLG
PROBE HOLDER ASSEMBLY
(PROBE POINTED AWAY FROM BOSS HOLE)
CABLE ROUTED THROUGH SCREW CENTER
RUBBER BUMPER
DISTANCE BETWEEN GEAR BOX AND END OF STRUT
BORE INSPECTION DRIVE ASSEMBLY
MOTOR
CABLE CONNECTOR
NLG OR MLG PROBE HOLDER ASSEMBLY (PROBE POINTED TOWARD BOSS HOLE)
STRUT ADAPTOR SCREW
LEAD SCREW
RUBBER BUMPER
DISTANCE FROM PROBE COIL TO GEAR BOX HOUSING
RUBBER BUMPER
BOLT HOLE PROBE
MOTOR CONTROLLER
115 V, A.C. 60 CYCLES
MAGNAFLUX ED-520 CRACK DETECTOR
BORE INSPECTION ASSEMBLY (EQUIPMENT LISTED IN JSK-7-INST/M)
SMR151-58.5-0-003-4

Figure 58.5-3-4/2. NLG and MLG Strut Cylinders (Sheet 4 of 5)



SURFACE PROBE GUIDE ASSEMBLY
(P/N JSK-7-506 THROUGH -511)


BOLT HOLE PROBE
SCREW TO LOCK PROBE GUIDE ON SPOTFACE (TYPICAL 2 PLACES)
PROBE GUIDE
SET SCREW TO LIGHTLY LOCK PROBE
SPOTFACE (TYPICAL 2 PLACES)

BOLT HOLE PROBE GUIDE ASSEMBLY
(P/N JSK-7-501 THROUGH -504)

SMR151-58.5-0-003-5

Figure 58.5-3-4/2. NLG and MLG Strut Cylinders (Sheet 5 of 5)

58.5.5 NLG AND MLG WHEELS.

58.5.5.1 DESCRIPTION. (See figure 58.5-4-2/0.) The landing gear has two main strut assemblies and one nose strut assembly. Each strut has dual wheels with tubeless tires. Each wheel has two wheel halves which are bolted together with steel fasteners. The wheel halves are machined from aluminum forgings and are finished with zinc chromate primer and aluminum lacquer. The bolts are fabricated from high strength alloy steel and are cadmium plated.

58.5.5.2 DEFECTS. Cracks may occur in the inner and/or outer flange lip or any of the holes in the wheel half. Circumferential and longitudinal cracks may occur in the thread root areas, in the shank sections and in the radii beneath the heads and/or in the heads of the attachment bolts.

Note

Either Primary NDE Procedure No. 1 or Alternate Primary NDE Procedure No. 1 must be performed.

58.5.5.3 PRIMARY NDE PROCEDURE NO. 1 - FLUORESCENT PENETRANT.

a. NDE equipment: Fluorescent penetrant materials and equipment selected from approved list in figure 58.0-4.

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

WARNING

Deflate the tire before loosening tie bolts. Failure to comply may result in serious injury to personnel.

(2) Dismount and disassemble the wheels in accordance with Chapter 32 of this manual and applicable B.F. Goodrich overhaul manual.

c. Access: The inspection areas are accessible when the wheels are disassembled.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Strip finish and clean surfaces of both sides of the wheel halves.

e. Instrument settings/calibration: Not applicable.

f. Inspection: Inspect the entire inner and outer surface of the wheel halves on main and nose landing gear. Refer to Section 58.1 of this manual and perform fluorescent penetrant inspection in accordance with instructions in figure 58.1-2.

g. Mark and report indicated defects. Proceed to Primary NDE Procedure No. 2.

58.5.5.4. ALTERNATE PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Shielded, V.M. Products P/N VM202-1/8 or equivalent.

(3) Calibration Standard, aluminum. (See figure 58.1-9.)

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

c. Access: The inspection areas are accessible when the wheels are disassembled.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions prior to surface eddy current scanning.

e. Instrument settings/calibration: Refer to Section 58.1 for calibration of the crack detector.

f. Inspection: Surface scan the entire inner and outer surface of the wheel halves on the main and nose landing gear. Surface scan around all holes. Scan all radii in both transverse and longitudinal direction. Refer to Section 58.1 for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

g. Mark and report indicated defects.

h. To confirm suspected defects, perform the backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.5.5.5. PRIMARY NDE PROCEDURE NO. 2 - MAGNETIC PARTICLE.

a. NDE equipment.

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as for Primary NDE Procedure No. 1.

58.5.5.5. (Cont)

c. Access: The tie bolts are accessible when the wheels are disassembled.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-base cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

Nose landing gear wheel tie bolts - 8 per wheel.

Circular Field	350 amperes
Longitudinal Field	1000 amperes

Main landing gear wheel tie bolts - 12 per wheel.

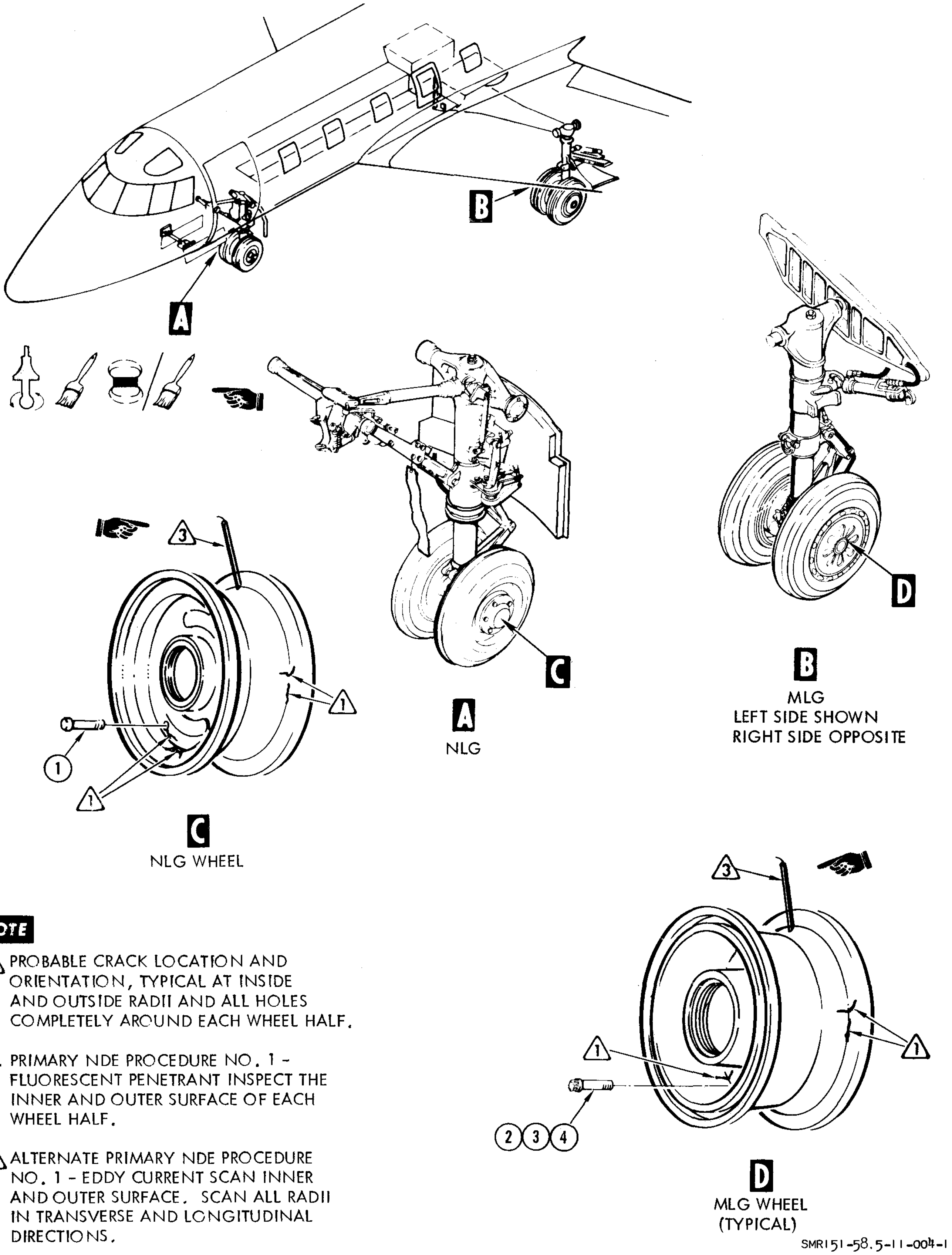
Circular Field	500 amperes
Longitudinal Field	2000 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Inspect all of the tie bolts on all main and nose landing gear wheels. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. Proceed to System Securing.

58.5.5.6. BACKUP NDE PROCEDURE. If Alternate Primary NDE Procedure No. 1, Eddy Current, is performed then Primary NDE Procedure No. 1, Fluorescent Penetrant, must be performed as a backup procedure. If Primary NDE Procedure No. 1, Fluorescent Penetrant, is performed no backup procedure is required. A backup procedure is not required for Primary NDE Procedure No. 2, Magnetic Particle.

58.5.5.7. SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.



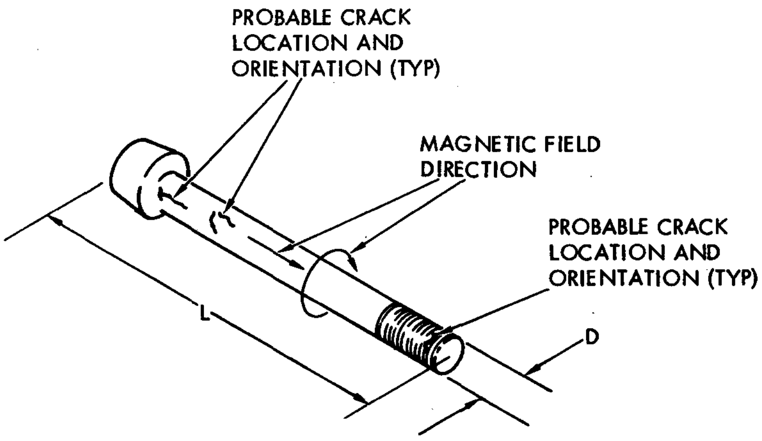
NOTE

△1 PROBABLE CRACK LOCATION AND ORIENTATION, TYPICAL AT INSIDE AND OUTSIDE RADII AND ALL HOLES COMPLETELY AROUND EACH WHEEL HALF.

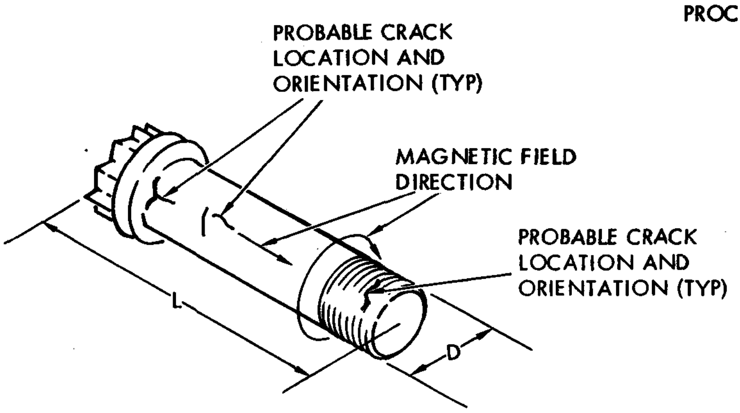
2. PRIMARY NDE PROCEDURE NO. 1 - FLUORESCENT PENETRANT INSPECT THE INNER AND OUTER SURFACE OF EACH WHEEL HALF.

△3 ALTERNATE PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT SCAN INNER AND OUTER SURFACE. SCAN ALL RADII IN TRANSVERSE AND LONGITUDINAL DIRECTIONS.

Figure 58.5-4-2/0. NLG and MLG Wheels (Sheet 1 of 2)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D

①
PRIMARY NDE
PROCEDURE NO. 2


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D

②③④
PRIMARY NDE
PROCEDURE NO. 2

WHEEL	BOLT	D (IN)	L (IN)	MATERIAL	FINISH
NLG, 3-999	① NAS 145-41	0.31	2.83	△3	△4
MLG, 3-1003	② NAS 626-20	0.37	2.26	△5	△4
MLG, 3-1184	③ 43-626	0.43	2.34	△5	△4
MLG, 3-1274	④ 43-715	0.50	2.30	△5	△4

NOTE

△3 ALLOY STEEL, INTERNAL WRENCHING

△4 CADMIUM PLATED

△5 ALLOY STEEL, 12-POINT EXTERNAL WRENCHING

SMR151-58.5-0-004-2

Figure 58.5-4-2/0. NLG and MLG Wheels (Sheet 2 of 2)

SECTION 58.6

NACELLE GROUP

Contents Page

GENERAL DESCRIPTION 58-181

Inspection Procedures 58-181

58.6.1 GENERAL DESCRIPTION.

The nacelle group consists of a nose cowl, forward nacelle, box beam frames, aft nacelle, longitudinal firewall, thrust reverser, and stang assemblies. See Chapter 54 for complete description of the nacelle structure and detail components.

58.6.1.1 INSPECTION PROCEDURES. Nacelle Group areas and components to be inspected are tabulated in the Nacelle Group Inspections Index (See Table 58.6-1). Each inspection procedure has been assigned an inspection index number which is cross referenced to the relevant paragraph and page number.

Table 58.6-1. Nacelle Group Inspection Index

INDEX NO.	INSPECTION TITLE	PARAGRAPH NO.	PAGE NO.
N-1	Engine Nacelle Attachment Fittings and Bolts.	58.6.2	58-182
N-2	Engine Mount Fittings and Bolts.	58.6.3	58-185

58.6.2 ENGINE NACELLE ATTACHMENT FITTINGS AND BOLTS.

58.6.2.1 DESCRIPTION. (See figure 58.6-1-4/0.) The nacelle structure includes two box beam frames that provide attachment of the nacelle to the fuselage. Four high strength steel bolts secure each frame to engine mount fittings attached to the fuselage bulkheads at fuselage stations 589 and 608. The box beam frames are fabricated from 4340 steel forgings, are cadmium plated, and finished with epoxy primer. The mount fittings are fabricated from 2014-T6 aluminum alloy forgings, are sulfuric acid anodized, and finished with zinc chromate primer. The attach bolts are 12-point external wrenching type and are cadmium plated.

58.6.2.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the box beam frames, mount fittings, and attach bolts. Cracks will initiate in the fastener holes of the frames and fittings and propagate in a random direction. Cracks in the bolts will propagate in a circumferential and longitudinal direction.

Note

Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.6.2.3 PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-250 or equivalent.

(2) Probe, Shielded, Bolt Hole, 5/8-inch diameter, V.M. Products P/N VM101BS-5/8 or equivalent.

(3) Calibration Standard, steel (See figure 58.1-9).

(4) Calibration Standard, aluminum (See figure 58.1-9).

b. Preparation of airplane:

WARNING

Ensure that power is isolated from all systems in the inspection area prior to approaching the inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove upper box beam fillet left and right, and the lower box beam fairing left and right. Refer to Section 52.3 of this manual for fillet and fairing location.

c. Access: The attachment bolt holes are accessible from outside the fuselage above and below the nacelle with the use of appropriate AGE or maintenance stands. Access to the interior of the fuselage is through the lower aft access door.

d. Preparation of part:

CAUTION

Do not remove more than one attachment bolt from each side of the airplane at any time. Failure to comply could result in difficulty in reinstalling bolts and/or damage to structure.

(1) Remove only one nacelle attach bolt and barrel nut each from the left and right sides. After the inspection of the bolt and the fastener hole, reinstall or replace the bolt and barrel nut prior to removing the next bolt and barrel nut.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

(2) Refer to Section 58.0 for cleaning instructions. Clean the fastener hole surface areas on the nacelle box beam frames and nacelle mount fittings area as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 of this manual for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection:

(1) Using the bolt hole probe and the instrument calibrated for steel, scan the entire inner surface of the bolt hole in the nacelle box beam frame from the outer edge to the first interface only.

Note

The inboard end of the fastener holes in the engine mount fittings are curved at the barrel nut cavity. The upper and lower surfaces extend farther inboard than the fore and aft surfaces. A downward deflection of the instrument needle will occur when scanning at the fore and aft quadrants of the hole. This will not indicate a defect.

(2) Recalibrate the instrument for aluminum and scan the entire inner surface of the bolt hole from the first interface to the inboard end of the hole in each engine mount fitting.

(3) Refer to Section 58.1 of this manual for eddy current scanning techniques. A sharp meter deflection, as noted during calibration, will indicate a probable crack in the part being inspected.

(4) Inspect the bolt removed in step d. (1) in accordance with Primary NDE Procedure No. 2.

(5) Mark and report indicated defects.

(6) If no defects are indicated, reinstall or replace the attach bolt and repeat steps (1) through (5) at remaining attach bolt locations.

58.6.2.4 PRIMARY NDE PROCEDURE NO. 2 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The attachment bolts are accessible when removed from the aircraft.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field - 700 amperes

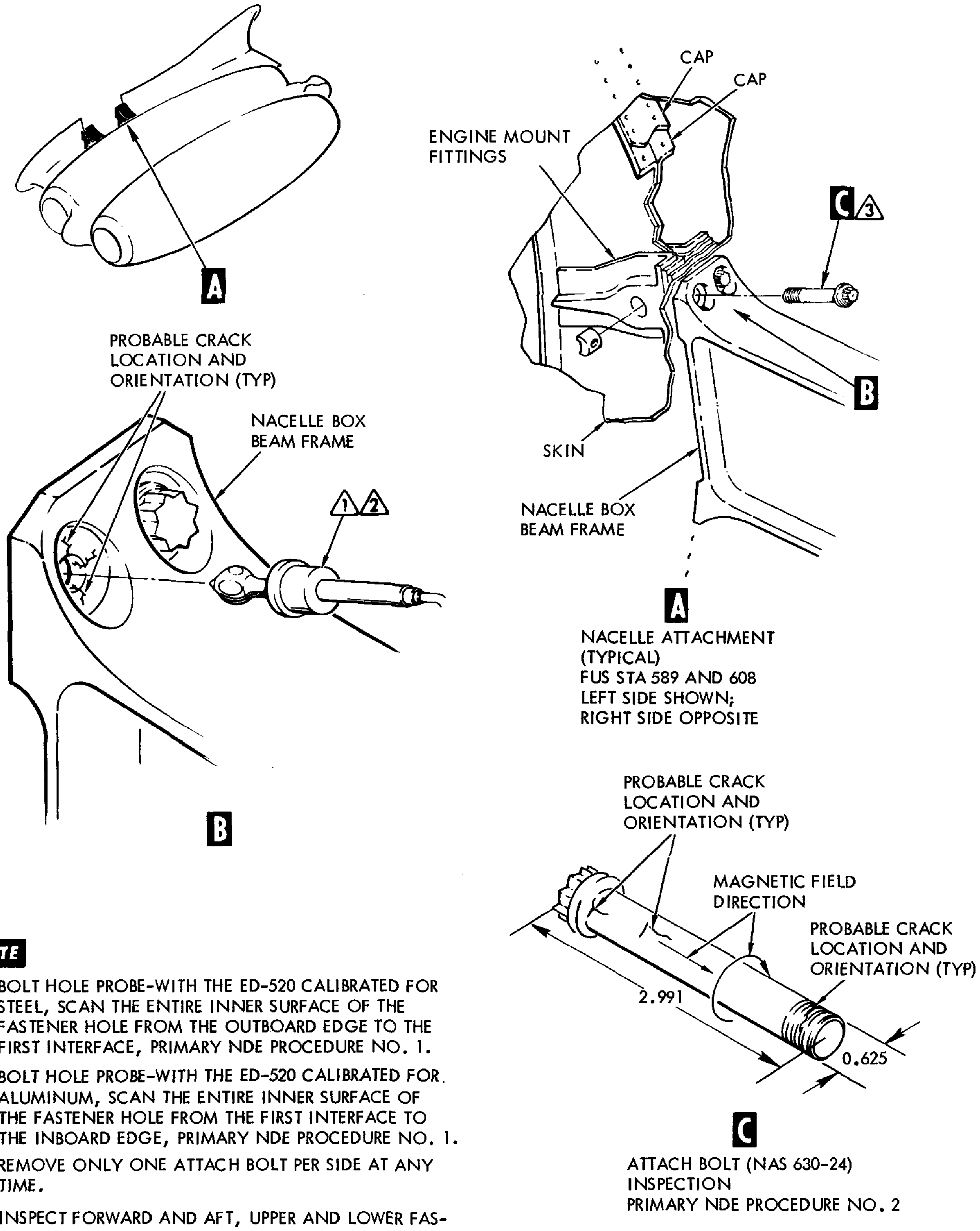
(2) Longitudinal field - 2000 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Apply the circular magnetic field first. Demagnetize the bolt upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects.

58.6.2.5 BACKUP NDE PROCEDURE: None required.

58.6.2.6 SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


A
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
NACELLE BOX BEAM FRAME
1 2
B
CAP
CAP
ENGINE MOUNT FITTINGS
C 3
B
SKIN
NACELLE BOX BEAM FRAME
A
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
2.991
0.625
C

NACELLE ATTACHMENT (TYPICAL)
FUS STA 589 AND 608
LEFT SIDE SHOWN;
RIGHT SIDE OPPOSITE

ATTACH BOLT (NAS 630-24)
INSPECTION
PRIMARY NDE PROCEDURE NO. 2

NOTE

△1 BOLT HOLE PROBE-WITH THE ED-520 CALIBRATED FOR STEEL, SCAN THE ENTIRE INNER SURFACE OF THE FASTENER HOLE FROM THE OUTBOARD EDGE TO THE FIRST INTERFACE, PRIMARY NDE PROCEDURE NO. 1.

△2 BOLT HOLE PROBE-WITH THE ED-520 CALIBRATED FOR ALUMINUM, SCAN THE ENTIRE INNER SURFACE OF THE FASTENER HOLE FROM THE FIRST INTERFACE TO THE INBOARD EDGE, PRIMARY NDE PROCEDURE NO. 1.

△3 REMOVE ONLY ONE ATTACH BOLT PER SIDE AT ANY TIME.

4. INSPECT FORWARD AND AFT, UPPER AND LOWER FASTENER HOLES AND BOLTS AT FUSELAGE STATIONS 589 AND 608.

SMR151-58.6-0-001

Figure 58.6-1-4/0. Engine Nacelle Attachment Fittings and Bolts.

58.6.3 ENGINE MOUNT FITTINGS AND BOLTS.

58.6.3.1 DESCRIPTION. (See figure 58.6-2-4/0.) Each engine is secured to the upper nacelle structure at three attachment points; two at the aft engine mount and one at the forward engine mount. The aft attachment points consist of two aft engine mount fittings attached to the nacelle box beam frame at nacelle station 100.00. Swing foot fittings which cradle the aft engine mounting pads are secured to the mount fittings by swing foot bars and bolts. The forward attachment point consists of the forward engine mount fitting attached to the nacelle box beam frame at nacelle station 81.50 which is secured to the forward engine mount assembly by one bolt. The fittings and bolts are fabricated from high strength alloy steel and are cadmium plated.

58.6.3.2 DEFECTS. Stress corrosion or fatigue cracks may occur in the engine mount fittings and attachment bolts. Cracks in the fittings will initiate in the fastener holes and propagate radially. Cracks in the bolts will propagate in a circumferential and longitudinal direction.

Note

Both Primary NDE Procedures No. 1 and No. 2 must be performed to complete this inspection.

58.6.3.3. PRIMARY NDE PROCEDURE NO. 1 - EDDY CURRENT.

a. NDE equipment:

(1) Crack Detector, Magnaflux P/N ED-520 or equivalent.

(2) Probe, Bolt Hole, 1/4-inch diameter, Ideal Specialty P/N 6200-1/4-BH-BNC or equivalent.

(3) Probe, Bolt Hole, 5/16-inch diameter, Ideal Specialty P/N 6200-5/16-BH-BNC or equivalent.

(4) Calibration Standard, steel. (See figure 58.1-9.)

b. Preparation of airplane:

WARNING

Ensure that power is isolated from aircraft prior to approaching inspection area. Failure to comply may result in serious injury to personnel.

(1) Isolate power from all systems in the inspection area in accordance with appropriate sections of this manual.

(2) Remove the engines from the nacelles per instructions in Chapter 72 of this manual.

(3) Remove the aft engine mount swing foot fittings and the eye bolts from the aft engine mount fittings.

c. Access: The fittings are accessible when above removals have been accomplished.

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part: Refer to Section 58.0 for cleaning instructions. Clean the fastener hole as required to permit good contact between part and probe.

e. Instrument settings/calibration: Refer to Section 58.1 for calibration of the crack detector.

Note

The eddy current instrument must be recalibrated when inspection probes are changed.

f. Inspection: Surface scan the entire inner surface of the fastener hole in the JP311 and JP314 aft engine mount fittings, the JP313 swing foot fittings, and the JP315 forward engine mount fittings. Refer to Section 58.1 for bolt hole scanning of straight wall fastener holes. A sharp meter deflection, as noted during calibration, will indicate a probable crack in part being inspected.

g. Mark and report indicated defects.

h. To confirm suspected defects, perform the backup NDE Procedure. If no defect is suspected, proceed to Primary NDE Procedure No. 2.

58.6.3.4. PRIMARY NDE PROCEDURE NO. 2 - MAGNETIC PARTICLE.

a. NDE equipment:

(1) Magnetic Particle Unit with 16-inch coil, Magnaflux P/N H720 or equivalent.

(2) Magnetic Particle Material, Magnaflux P/N 14 A Magnaglo or equivalent.

(3) Portable Black Light, Magnaflux P/N ZB-26 or equivalent.

(4) Demagnetizer, Magnaflux P/N SB 1619 or equivalent.

(5) Field Indicator, Magnaflux P/N 2480 or equivalent.

b. Preparation of airplane: Same as Primary NDE Procedure No. 1.

c. Access: The bolts are accessible when removed from the mount fittings.

58.6.3.4. (Cont)

CAUTION

High strength steels are extremely susceptible to hydrogen embrittlement caused by acid-based cleaners and solvents. Use only cleaners and solvents that are certified safe for use on high strength steel parts. Failure to comply may cause cracks.

d. Preparation of part. Refer to Section 58.0 for cleaning instructions. Clean the bolts of contaminants that might interfere with concentration of magnetic particles at discontinuities or cause improper accumulation of particles.

e. Instrument settings/calibration: Calibration not applicable. Set selector switch on unit for direct current power supply. Adjust current as follows:

(1) Circular field:

JP 316-1	-	500 amperes
NAS 1104-7	-	600 amperes
NAS 1105-20D	-	600 amperes
NAS 1105-22D	-	600 amperes

(2) Longitudinal field:

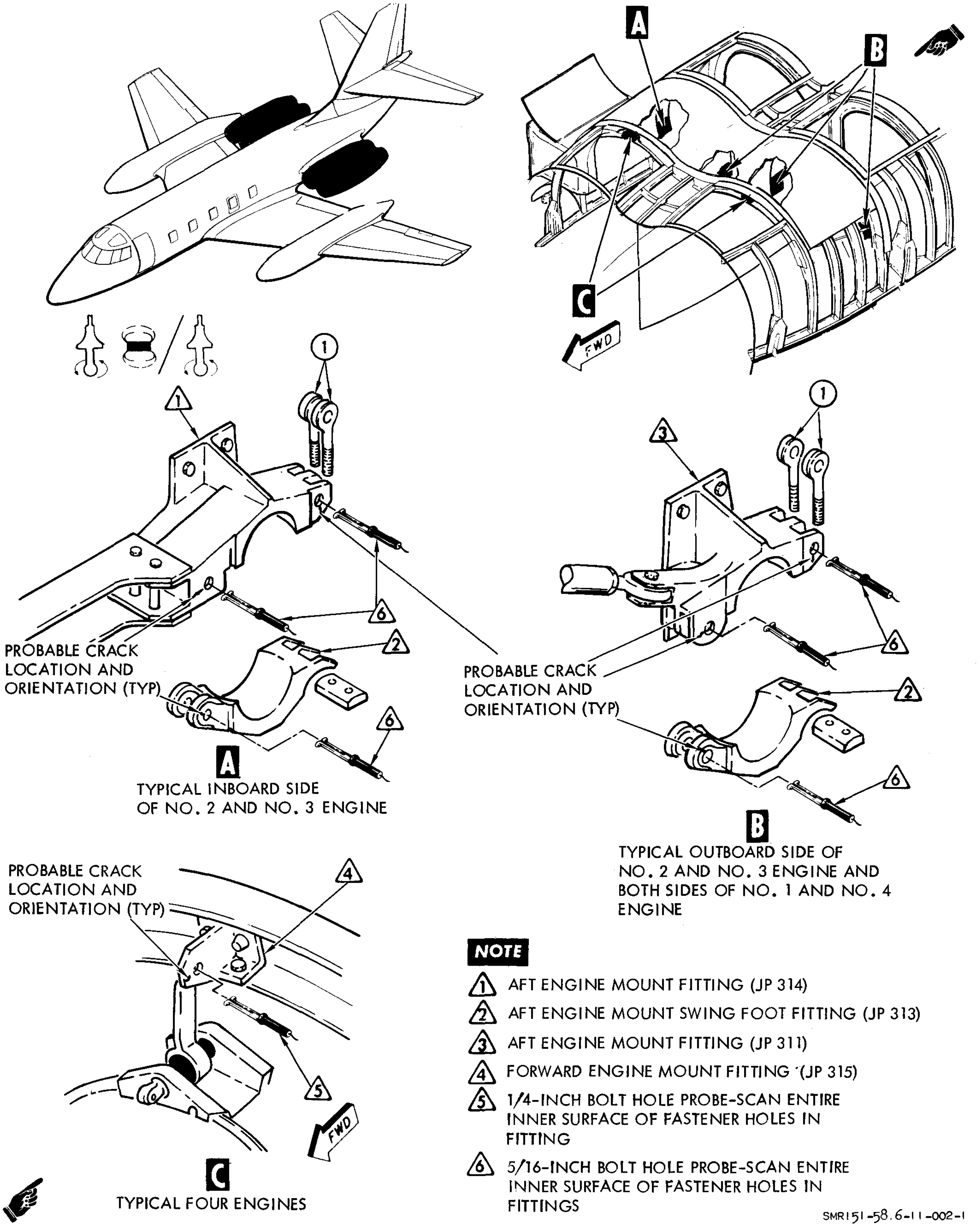
JP 316-1	-	1000 amperes
NAS 1104-7	-	2800 amperes
NAS 1105-20D	-	2800 amperes
NAS 1105-22D	-	2800 amperes

f. Inspection: Inspect the bolts by the magnetic particle wet continuous method using both circular and longitudinal magnetic fields. Inspect the bolts on all four engines as illustrated. Apply the circular magnetic field first. Demagnetize the bolts upon completion of inspection. Refer to Section 58.1 of this manual for instructions on application of magnetic fields and demagnetization.

g. Mark and report indicated defects. Proceed to System Securing.

58.6.3.5. BACKUP NDE PROCEDURE. In order to verify suspected defects, have a different inspector repeat the Primary NDE Procedure No. 1, inspecting only the suspected areas. A backup procedure is not required for Primary NDE Procedure No. 2, Magnetic Particle.

58.6.3.6. SYSTEM SECURING. Clean areas inspected, restore finishes, reinstall removed components, and perform operational checkouts as required in accordance with applicable sections of this manual.


A
B
C
FWD
①
△1
△3
△6
△2
△4
△5
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
FWD

A TYPICAL INBOARD SIDE OF NO. 2 AND NO. 3 ENGINE

B TYPICAL OUTBOARD SIDE OF NO. 2 AND NO. 3 ENGINE AND BOTH SIDES OF NO. 1 AND NO. 4 ENGINE

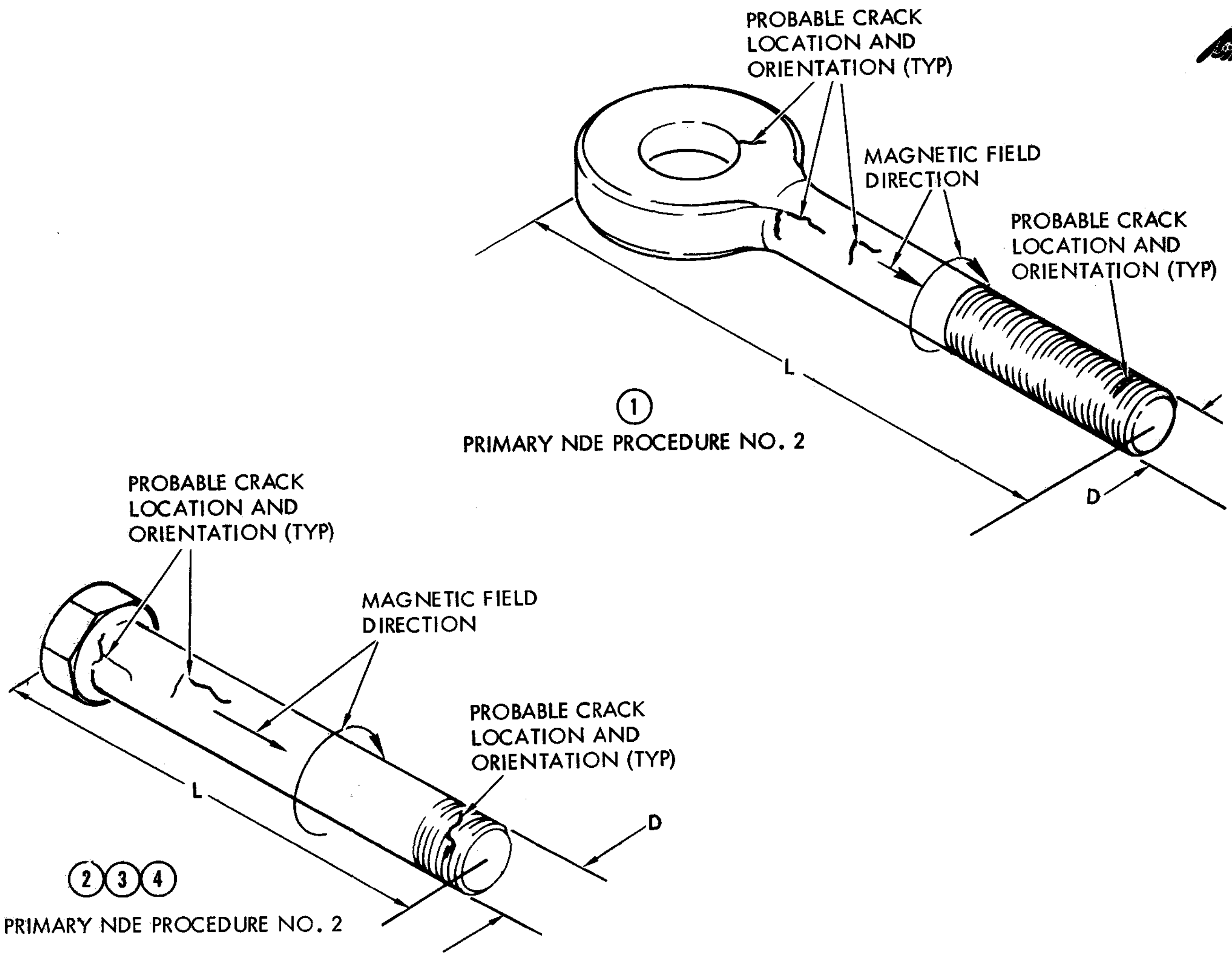
C TYPICAL FOUR ENGINES

NOTE

△1 AFT ENGINE MOUNT FITTING (JP 314)
△2 AFT ENGINE MOUNT SWING FOOT FITTING (JP 313)
△3 AFT ENGINE MOUNT FITTING (JP 311)
△4 FORWARD ENGINE MOUNT FITTING (JP 315)
△5 1/4-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF FASTENER HOLES IN FITTING
△6 5/16-INCH BOLT HOLE PROBE-SCAN ENTIRE INNER SURFACE OF FASTENER HOLES IN FITTINGS

SMR151-58.6-11-002-1

Figure 58.6-2-4/0. Engine Mount Fittings and Bolts (Sheet 1 of 2)


PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
①
PRIMARY NDE PROCEDURE NO. 2
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
MAGNETIC FIELD DIRECTION
PROBABLE CRACK LOCATION AND ORIENTATION (TYP)
L
D
②③④
PRIMARY NDE PROCEDURE NO. 2

	BOLT	D (IN)	L (IN)	MATERIAL	FINISH
①	JP316-1 (2)	0.25	2.31	△7	△9
②	NAS1105-20D (1)	0.31	1.78	△8	△9
③	NAS1105-22D (1)	0.31	1.91	△8	△9
④	NAS1104-7 (1)	0.25	0.88	△8	△9

NOTE

△7 ALLOY STEEL, EYE BOLT

△8 ALLOY STEEL, HEX HEAD

△9 CADMIUM PLATED

SMR151-58.6-11-002-2

Figure 58.6-2-4/0. Engine Mount Fittings and Bolts (Sheet 2 of 2)

CHAPTER 59

STRUCTURAL REPAIR INSTRUCTIONS

Section		Page
59.0	INTRODUCTION	59-3
59.1	GENERAL	59-7
59.2	WING GROUP	59-59
59.3	EMPENNAGE GROUP	59-71
59.4	FUSELAGE GROUP	59-79
59.5	LANDING GEAR GROUP	59-97
59.6	NACELLES	59-99
59.7	DAMAGE DUE TO LANDING GEAR FAILURE	59-103
59.8	TYPICAL REPAIRS AND APPLICATIONS	59-105
59.9	10,000 HOUR INSPECTION	59-151

SECTION 59.0

INTRODUCTION

Contents Page

GENERAL 59-3

COMPOSITION OF THE STRUCTURAL REPAIR CHAPTER 59-3

TERMINOLOGY 59-3

STRUCTURAL REPAIR CODE 59-3

FIGURES

59-1 Major Assembly Index 59-5
59-2 Structural Repair Code 59-6

59.0.1 GENERAL

a. The data in this chapter have been developed for maintenance personnel who are responsible for restoring the airworthiness of damaged JetStar airframes.

b. Repairs illustrated in the following sections apply to the primary and secondary structure. These repairs have been designed to provide strength and stiffness equivalent to the original design requirements of the airframe. The limits of damage which can be permitted, or which can be corrected by a simple procedure without placing restrictions on flight, also are defined in these sections. Most of the repairs shown can be made with tools and materials normally available. However, certain repairs to the primary structure can be made only in well equipped shops and should not be attempted unless the proper tools and materials are available.

59.0.2 COMPOSITION OF THE STRUCTURAL REPAIR CHAPTER. The JetStar structure comprises five basic units: Wing, Empennage, Fuselage, Landing Gear and Nacelle Groups (see figure 59-1 Major Assembly Index). The structural repair instructions are arranged in 9 sections. Section 59.1 contains general information and Sections 59.2 through 59.6 (Wing Group, Empennage Group, Fuselage Group, Landing Gear and Nacelle) describe the major airframe components and contain detailed repair information. Section 59.7 Damage Due to Landing Gear Failure, and Section 59.8 show typical repairs and applications. Section 59.9 gives the recommended 10,000 hour inspection.

59.0.3 TERMINOLOGY

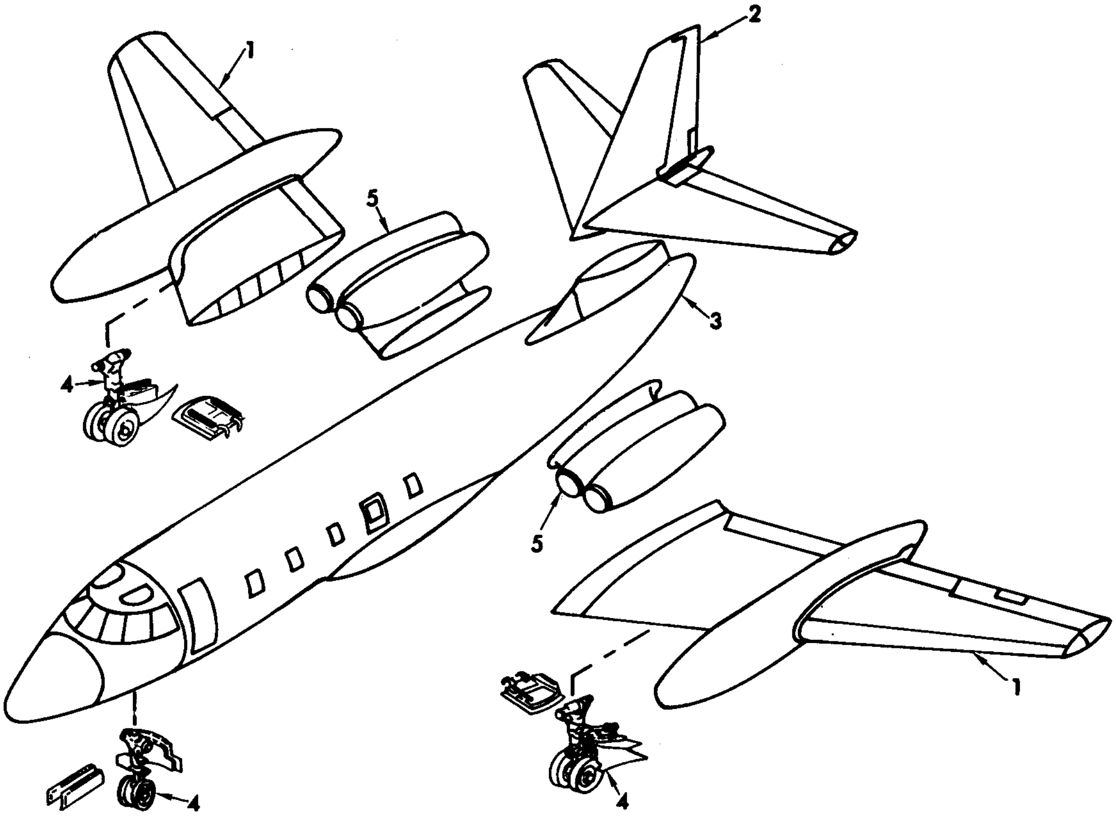
Such terms as "right and left", "upper and lower", "front and rear", "forward and aft", and "clockwise" and "counterclockwise" refer to the airplane as viewed from the rear, on the centerline, looking forward. All dimensions, thicknesses, and distances are given in inches unless otherwise noted.

59.0.4 STRUCTURAL REPAIR CODE

To clarify repair illustrations, a "Structural Repair Code" illustration (figure 59-2) is located in the Introduction. By means of this repair code, the methods of showing the existing structure, repair members, damaged area, fillers, existing rivets, new rivets in existing holes, new rivets in new holes, existing rivet holes, and new rivet holes in all repair figures have been standardized. Use the repair code with all repairs. The skin, web, or floor covering has been omitted from most to show repair of structure underneath.

CAUTION

Fasteners, as shown in the repair illustrations, are symbolic only. Do not countersink for, or drive, fasteners until the repair is completely assembled as shown on the repair illustrations. This includes existing or applicable skin, web, or floor covering.



ITEM	NOMENCLATURE	SECTION
1	WING GROUP	59.2
2	EMPENNAGE GROUP	59.3
3	FUSELAGE GROUP	59.4
4	LANDING GEAR GROUP	59.5
5	NACELLE GROUP	59.6

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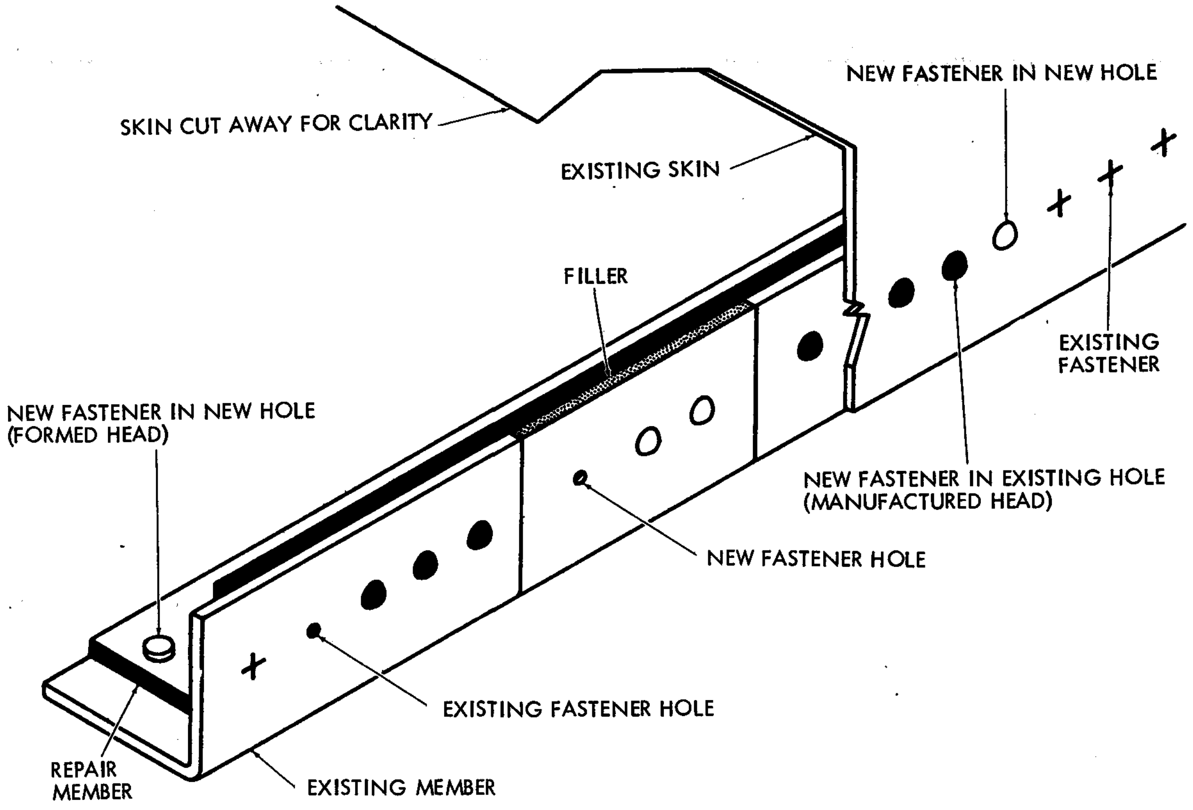
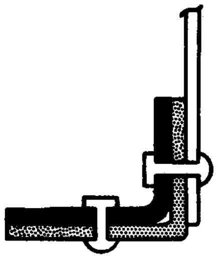
Figure 59-1. Major Assembly Index





SECTION THROUGH REPAIRED AREA

NOTE

1. FASTENER SPACING IS GIVEN FROM CENTER TO CENTER OF FASTENER HOLES.
2. SYMBOLS FOR MANUFACTURED AND FORMED RIVET HEADS ARE USED AS EXAMPLES IN THIS CODE. THIS CODE IS TYPICAL FOR ALL PROTRUDING OR FLUSH HEAD FASTENERS.
3. IN MOST REPAIR ILLUSTRATIONS, SKIN, WEB, OR FLOOR COVERING HAS BEEN OMITTED TO SHOW REPAIR OF STRUCTURE UNDERNEATH, AND FASTENER SYMBOLS ARE SHOWN ON THE STRUCTURE. STUDY CROSS-SECTIONAL VIEWS BEFORE INSTALLING FASTENERS.

Figure 59-2. Structural Repair Code

SECTION 59.1

GENERAL

Contents Page

REPAIR CAPABILITIES 59-9

DAMAGE EVALUATION. 59-9

Preliminary External Visual Inspection 59-9
Access Provisions 59-9
Detailed Visual Inspection 59-9
Structural Failure 59-10
Deformation in Secondary Structure 59-10
Internal Corrosion 59-10
Chemical Identification of Materials 59-10
Nondestructive Evaluation Methods 59-10
Fire Damage Evaluation 59-10
Hardness Testing. 59-10
Riehle Portable Hardness Tester. 59-10
RAR/RBR Portable Hardness Tester. 59-10A
Relative Conductivity Method of Hardness Testing 59-11
Leveling the Aircraft 59-11

CLASSIFICATION OF DAMAGE 59-11

Preparation of Damaged Area for Classification of Damage 59-11

TYPES OF REPAIRS 59-11

Negligible Damage 59-12
Damage Repairable by Patching 59-12
Damage Repairable by Insertion 59-12
Repair of Built-up Structural Members 59-12
Damage Necessitating Replacement of Part 59-13
Specific Repairs 59-13
Typical Repairs 59-13
One-Time Ferry Flight Repairs 59-13
Temporary Repairs 59-13
Permanent Repairs 59-13
Partial Repairs 59-13
Major Repairs 59-13

AERODYNAMIC SMOOTHNESS 59-13

CLEANING EXTERIOR SURFACES 59-13

CORROSION CONTROL 59-14
Corrosion Prevention for Repairs 59-14

AIRFRAME SEALING 59-14

PRESSURE TESTING 59-14

CONTROL SURFACE REBALANCING 59-14

Contents Page

GENERAL SHOP PRACTICES 59-14

Sheet Metal Forming 59-14
Solid Rivet Hole Sizes 59-14
Locating Blind Holes 59-14
Drilling Titanium and Steel 59-14
Holes for Flush Fasteners 59-14
Countersink Limits 59-15

FASTENERS 59-15

NAS 1097 Rivets 59-15
Blind Rivets 59-15
Blind Bolts 59-15
Hi-Loks 59-15
Lockbolts 59-15

INSPECTION OF INSTALLED FASTENERS 59-15

INSTALLATION OF FASTENERS 59-15

Blind Rivets 59-15
Blind Bolts 59-16
Hi-Loks 59-16
Hi-Lok Installation 59-17
Lockbolt Removal 59-17
Installation of Pull Type Lockbolts 59-17

FASTENER SUBSTITUTION 59-18

FILLING MACHINE COUNTERSUNK HOLES 59-18

CRASH HANDLING 59-19

Safety Precautions 59-19
Ground Handling 59-19
Critical Jacking and Towing Conditions 59-19
Hoisting, Lifting, Jacking and Towing Procedures 59-19
Protection of Crash Damaged Aircraft from Corrosion 59-19

SUPPORT OF STRUCTURE DURING REPAIRS 59-19

Fuselage Support Cradles 59-19
Wing Support Cradles 59-20

SERVICE LIFE CRITERIA 59-20

Permissible Wear and Deviations 59-20
Wear Indication 59-20
Reaming and Bushing 59-20
Reaming Holes 59-20
Bushing Removal 59-20
Bushing Installation 59-20

REPLACEMENT ITEMS 59-21

EPOXY INSTALLATION OF BEARINGS 59-21

Contents Page

FIGURES

59-3 Rivet and Joint Inspection 59-23
59-4 Rockwell Hardness Scale 59-24
59-5 Portable Hardness Testers 59-24B
59-6 Magnetic Hardness Testers 59-26
59-7 Classification of Damage 59-27
59-8 Aerodynamic Smoothness Requirements 59-31
59-9 Minimum Bend Radii for Repair Parts 59-33
59-10 Standard Joggles for Repair Parts 59-34
59-11 Attachment Holes, Edge Distances, and Countersinks 59-35
59-12 Locating Blind Holes 59-36
59-13 Comparison between NAS1097 and MS20426 Rivets 59-36
59-14 Blind Rivet Installations and Inspections 59-37
59-15 Blind Bolt Installation, Inspection and Removal 59-40
59-16 Hi-Lok Installation, Inspection and Removal 59-44
59-17 Hand Tools for Hi-Lok Fastener Installation 59-48
59-18 Lockbolt Collar Inspection 59-49
59-19 Lockbolt Installation and Removal 59-52
59-20 Fuselage Support Cradles 59-56
59-21 Bushing Removal and Installation 59-57
59-22 Installation of Bearing Using Adhesive 59-57

59.1.1 REPAIR CAPABILITIES. The repairs shown in the following sections are designed to be as simple as possible to install. They are also designed to the ultimate capabilities of the original sections rather than to the anticipated structural loading.

a. In most cases the repairs are within the capabilities of the average certificated airframe mechanic with the normal supply of hand tools. However, many repairs may require a large amount of machining and other specialized professional and technical capabilities better supplied by aircraft modification and interior installers.

b. Generally, if the repair is extensive and complex, if structural support is required, and if alignment is critically altered, the modification center with its larger stock of repair materials and its professional and technical manpower pool should be considered to rebuild the aircraft to an airworthy condition.

59.1.2 DAMAGE EVALUATION.

59.1.2.1 PRELIMINARY EXTERNAL VISUAL INSPECTION. It is important in this inspection to determine the more obvious damage to the aircraft and the areas involved. Examine the entire external surface of the aircraft and note damage. Damage to the skin such as scratches, scores, dents, buckles, and holes can usually be expected. One of the best indications of internal damage is buckled or wrinkled skin. During the inspection, note areas of the aircraft that are suspected of having internal damage.

59.1.2.2 ACCESS PROVISIONS. For access to the internal structure for inspection and repair, remove access doors and panels as necessary. Refer to Section 52.3 for access provisions.

59.1.2.3 DETAILED VISUAL INSPECTION. Look for buckles or wrinkles in skin, web, fairing, or between rivets. Whenever buckles and wrinkles appear, the rivets within and adjacent to the wrinkled area should be inspected very closely. Wrinkles that extend across the internal structure indicate failure of attachments. Check for rivets that are sheared, loose, or have tipped heads. The rivets still may be in position but may be loose or sheared. One way to inspect rivets is to use a 0.003 feeler gage (see figure 59-3). If a bolt head is tipped or loose, remove and check for a bent bolt and an elongated hole.

a. If the surface is painted, loose rivets may be detected by looking for cracks in the paint around the rivets. If the skin of the airplane has been subjected to severe stresses or strains, the surface paint will usually be cracked or chipped. All hinges, fasteners, and locking mechanisms should be carefully checked to determine efficiency of operation. If they are not functioning properly, it is an indication that the structural parts have been strained.

b. Collision damage is readily identified by dented or broken areas of skin and internal structure. Clean up the damage by cutting away broken, bent, or otherwise defective skin and internal structure. Inspect the area adjacent to the obvious damage and the attaching structure to which loads may have been transmitted.

c. Exceeding design loads by improper operation of the airplane such as extremely hard landings or high-speed taxi turns may cause damage indicated by permanently wrinkled skin and pulled rivets. A thorough investigation of the internal structure in

59.1.2.3 (Cont).

the vicinity of the wrinkled skin and pulled rivets must be made. A part of the structure might be distorted or weakened without an actual rupture of the part. When any doubt exists regarding distortion of the airplane, perform an alignment check in accordance with Section 51.1.

d. Defects which would escape detection by the naked eye may be seen when magnified. Optical devices for visual inspections are borescopes and magnifying glasses. The use of optical devices has two purposes:

   (1) To magnify defects which could not otherwise be seen by the naked eye.

   (2) To permit visual inspection of areas which are not accessible to the unaided eye.

Note

All visual inspections are to be performed with a 10X magnifying glass.

59.1.2.4 STRUCTURAL FAILURE. Repeated loads and vibration will cause cracking. The JetStar has been designed for long service life. Prevention of fatigue failure was an important factor in the design of the airplane. However, nicks, scratches, corrosion, or other damage to the surface of the metals will cause fatigue cracking sooner than predicted. Give particular attention to the skin and structure in the pressurized area of the fuselage. Inspect the fuselage carefully for cracks, particularly around the windows and openings.

59.1.2.5 DEFORMATION IN SECONDARY STRUCTURE. Permanent wrinkles or buckles in the secondary structure caused by deflections of the primary structure are not detrimental to structural integrity; however, these wrinkles or buckles should be inspected for fatigue cracking.

59.1.2.6 INTERNAL CORROSION. Watch for evidence of corrosion inside of skin. This is most likely to occur in pockets and corners where moisture and salt spray may accumulate. Drain holes must always be kept clean. Corrosion may be recognized by dulling and pitting of surface accompanied by a white or reddish powdery deposit. Sometimes corrosion is found under paint where paint flakes off when pressed. See Section 28.2 and Section 51.7 for corrosion control.

59.1.2.7 CHEMICAL IDENTIFICATION OF MATERIALS. Before any member can be repaired, its material must be positively identified. See Section 51.6 for a detailed outline for the chemical identification of the listed materials. This procedure will give positive identification if the material in question is known to be in the following group:

Aluminum or aluminum alloy
Magnesium or magnesium alloy
Titanium
17-7PH steel
Type 301 or 302 stainless steel
Carbon steel
Low alloy steel

59.1.2.8 NONDESTRUCTIVE EVALUATION METHODS. See Section 58.1 for detailed instructions for the various inspection methods. Section 58.0 provides the information required for cleaning prior to these inspections. Sometimes crack removal and other repairs can produce surface finishes that can mask defects on post-rework inspections. To reduce this possibility, reworked aluminum surfaces should be polished with 200 grit cloth or equivalent polishing compound after surface rework.

59.1.2.9 FIRE DAMAGE EVALUATION. Discolored or scorched paint is usually the primary evidence of overexposure to heat. Perform a hardness test on all parts which have been (or are suspected of having been) subjected to temperatures above the normal operating ranges. High temperatures will change the heat treat, and hence the load bearing ability of the affected parts. Compare the results of the hardness test with the properties which the part should have, according to the material and heat treat called out in the appropriate structure diagram and key. Repair or replace any part which is found to have damage as a result of excessive heat. A hardness test, even though it shows no change in strength resulting from excessive heat, does not mean there is no hidden damage. Any part which has been (or is suspected of having been) subjected to temperatures above the normal operating range should be checked frequently for signs of corrosion. Any evidence of corrosion is reason for repair or replacement of the part. Especially critical in this respect is 7075-T6 aluminum alloy.

59.1.2.10 HARDNESS TESTING. There is a direct relationship between material hardness and strength. This relationship makes it possible to determine the strength and heat-treat condition of a known material with reasonable accuracy by using a hardness test. There are two methods presented in this manual that may be used to determine the hardness of a material. One method (Rockwell) relates hardness to the penetration depth of a small ball or diamond point penetrator pressed against the material at a given load. Refer to figure 59-4 for correlation of Rockwell Scale readings with selected heat treat conditions of aluminum alloys and steels. The other method utilizes the measured electrical conductivity of a material to determine the relative hardness of a suspected defective specimen with that of a known-to-be acceptable specimen of the same material.

59.1.2.10.1 RIEHLE PORTABLE HARDNESS TESTER. (See figure 59-5.) The Riehle hardness tester provides Rockwell readings that may be related to the material strength values given in figure 59-4. The Riehle tester is limited to test specimen thickness and/or flange

59.1.2.10.1 (Cont).

depth that can be accommodated by its clamp dimensions. The procedure for its use is as follows:

Note

> Prior to correlating hardness readings obtained in the following procedure with the values in figure 59-4, the readings should be verified on the calibration test block (8) provided with the tester.

a. Select penetrator (2) suitable for the material being tested; install and secure with clamp screw (3).

b. Select proper anvil (9) to suit configuration of test specimen.

c. Make sure that loading screw (5) is backed off sufficiently so that penetrator (2) does not project beyond upper clamp.

d. Clamp work firmly between upper clamp and anvil by means of adjusting knob (1) on lower clamp. Do not force.

e. Check zero setting of load indicator (6). Rotate bezel to bring pointer over small black dot for zero load.

f. Apply 10Kg minor load (pointer to "set") by means of loading screw (5). Hold instrument steady to avoid twisting motion of penetrator on the test specimen.

g. Check zero setting of penetration indicator (7). Rotate bezel to bring pointer to "0" on the black scale.

h. Apply major load specified in illustration for Rockwell scale being employed by means of loading screw (5).

i. Take off major load by reducing to minor load (pointer back to "set").

j. Hardness is indicated on penetration indicator (7).

k. Release load completely by backing off loading screw (5).

l. Turn adjusting knob (1) to left to loosen lower clamp.

59.1.2.10.2 RAR/RBR PORTABLE HARDNESS TESTER. (See figure 59-5.) The RAR/RBR hardness tester provides Rockwell readings on the A or B scale; however, due to variations in operating technique, the results obtained from this instrument are not considered as reliable as the results obtained from a clamp type instrument. This instrument is useful in obtaining hardness values from specimens in areas where a clamp type instrument cannot be installed. For operation and use of the RAR/RBR hardness tester, proceed as follows:

Note

> Hardness readings taken on the surface of a clad sheet of 0.090 inch thickness or greater will be in error due to the effect of the relatively thick coating of soft aluminum cladding.

a. Remove the finish from the surface to be tested. Remove all scratches and variations that might affect the reading.

Note

> The indenter must be perpendicular to the surface.

b. Adjust the dial to 0.

Note

> A specimen being tested by the indentation method must be firmly supported to prevent deflection of the specimen. Erroneous readings will be obtained if the specimen deflects.

CAUTION

> Take care to avoid side thrust on the point of the instrument when applying it to the surface. Failure to comply may result in damage to the material being tested.

59.1.2.10.2 (Cont).

c. Apply pressure to the tester as shown in figure 59-5.

d. Take a reading on the structure that has not been damaged by heat for comparison.

e. Now take readings, working toward the damaged area, to determine the extent of damage.

f. Compare the readings in the suspected damaged area with the undamaged area.

g. Repair or replace any skin and/or structure found to have heat damage.

59.1.2.10.3 RELATIVE CONDUCTIVITY METHOD OF HARDNESS TESTING. (See figure 59-6.) The conductivity method of testing is accomplished by the Magnatest FM-110 or Magnatest FM-120 unit. (See figure 59-6.) This method of testing is based on the principle that the conductivity of a material will vary as the hardness or heat-treat varies. The instrument operates by using a small coil which is energized with a 60-cycle signal. When the probe is placed against the specimen (requiring a flat surface about the size of a nickle), the electromagnetic field is altered and unbalances a bridge circuit. The bridge is restored to balance by a variable capacitor connected to the dial calibrated in conductivity units. The conductivity is measured in a circle in the plane of contact and takes into account only the metal in the immediate vicinity (0.040 to 0.060 inch deep and 5/8 inch in diameter). Use the following procedure to perform the conductivity test:

CAUTION

> It is not recommended that this method be used for testing of 2024-T6, -T36, -T81, or -T86 alloys since the conductivity changes of these tempers are not sufficient for reliable determination of a loss in physical properties.

a. Survey the suspected damaged area visually using paint discoloration, smut, or distortion as a guide.

b. Determine the material's composition and temper by referring to the appropriate section of this manual.

c. Prepare the working area by removing any components which could impair the heat damage survey. The area to be inspected should be free of any heavy contamination which would prevent the probe from making good contact with the surface.

d. Balance the conductivity meter.

e. Take reference conductivity readings of like material and temper from unaffected area. (This could be obtained from another airplane if feasible.)

f. Take systematic grid-type (horizontal and vertical) conductivity readings at about 6-inch intervals until the reference conductivity value obtained in step e is obtained in all directions. (Do not skip from place to place.)

g. Make each reading on the material using a suitable pen or pencil. See paragraph titled Temporary Marking contained in Section 58.0 for approved materials.

59.1.2.11 *LEVELING THE AIRCRAFT. Aircraft* symmetry and alignment checks require the aircraft to be jacked and precisely leveled. In leveling the aircraft for symmetry and alignment checks, jack the aircraft at the wing jack points and the nose jack point only. Use no support medium in contact with the wing, fuselage, or the empennage. Refer to Chapter 7 for jacking procedure, and use the external leveling method specified in Chapter 8.

59.1.3 CLASSIFICATION OF DAMAGE. Damage is defined as any reduction in cross-sectional area or any deformation of a structural member or skin of the aircraft. Generally, damage consists of cuts, holes, wrinkles, twists, nicks, dents, cracks, scratches, corrosion, or burned areas. Damage is classified as negligible, reparable by patching, reparable by insertion, or necessitating replacement of parts. Use the following paragraphs to aid in determining the proper classification of the damage.

59.1.3.1 PREPARATION OF DAMAGED AREA FOR CLASSIFICATION OF DAMAGE.

a. Remove damaged portions of the members.

b. Smooth the edges of the holes and smooth the cut or nicked portions of members to eliminate stress concentrations.

c. Clean or remove corroded or burned areas, depending on the amount of damage. Use generous radii when cutting out damage or when smoothing members that have been damaged.

d. Paint the bare surfaces of the reworked areas with zinc chromate primer (TT-P-1757). Pre-treat the surfaces with anti-corrosion treatment per Section 51.7 before applying the primer.

59.1.4 TYPES OF REPAIRS. The repairs given in this Section are of two general types: specific or typical. These repairs are further defined, depending upon the load-carrying capability as one-time ferry flight, temporary, or permanent. The repairs in this manual are illustrated for the maximum cross sectional damage to a member. Repair of less than maximum cross sectional damage is considered partial.

59.1.4.1 NEGLIGIBLE DAMAGE. Certain damage to the aircraft which does not affect the strength, rigidity, or function of the part involved is classified as negligible. This damage either requires no repair or can be corrected by removing dents, stop-drilling cracks, or smoothing nicks or holes. For the limits and treatment of negligible damage, see the applicable section text and typical negligible damage illustrations in Section 59-8.

59.1.4.2 DAMAGE REPAIRABLE BY PATCHING. (See figure 59-7.) Damage which can be repaired to give strength equal to that of the original member by installing repair members to bridge the damaged portion of the part is classified as reparable by patching. This applies to skin and structure. The majority of the repairs shown are for damage of this type. Repair members are attached to the undamaged portions of the part on each side of the damaged portion to restore full load-carrying ability and airworthiness to the aircraft. The fillers shown are included to provide proper spacing. In every case, the repair illustration shows the repair of the worst allowable damage to the cross section of the member. The length of the repair member needed for a given repair will depend on two things; the rivet spacing of the desired repair, and the length of the cleaned up damaged area. The two terms "tie-in" and "bridge", used to identify areas of the repair, are explained as follows:

a. The tie-in length is the part of the repair member that overlaps the sound original member. Rivet spacing, as shown in the repair illustration, will determine the length of material needed for the tie-in. The tie-in length is found by this formula: number of rivets in the longest single row less one, multiplied by center-to-center spacing, plus twice the edge distance.

   EXAMPLE: With 8 rivets at 0.75 inch center-to-center spacing with 0.37 inch edge distance at each end:

   (8-1) x (0.75) + (2x0.37) =
   (7 x 0.75) + (2x0.37) =
   (5.25) + (0.74) = 5.99

   or 6 inches of tie-in length of the repair member on each side of the cleaned up damaged area.

b. The bridge length is the length of the cleaned up area. Then the total length of the splice or repair member is equal to twice the tie-in length plus the bridge length.
   EXAMPLE: Assume the bridge length is equal to 3 inches. Use the tie-in length computed in the previous example of 6 inches. Compute the repair member length as follows:

   Repair member length = (2x6) +3
   = 15 inches.

   The maximum length of damage (bridge length) which can be repaired by patching should never be more than twice the tie-in length for the designed repair for the particular member, unless otherwise specified on the repair illustration. EXAMPLE: Using previously computed tie-in length of 6 inches, compute maximum bridge length that can be repaired by patching:

   Maximum bridge length = 2 x 6
   = 12 inches.

c. For repair of more extensive damage, use insertion repair or replacement of the member (see Section 59.1). Typical skin repairs are located in Section 59.8, and specific skin repairs are located in the applicable section.

59.1.4.3 DAMAGE REPAIRABLE BY INSERTION. (See figure 59-7). Damage which can be repaired to restore original strength by inserting a length of repair member of the same shape and material is classified as repairable by insertion. Repair-by-insertion differs from repair-by-patching shown in this manual in three ways: (1) insertion repairs are used to simplify repair when the damaged part of a member is longer than allowed by repair patching, (see Section 59.1), (2) when interference with other structural members is to be avoided, and (3) an insertion member, made of the same material and of the same cross-sectional properties as the original member, is substituted for and replaces the filler shown in the repair illustrations. The insertion member is fitted into the space left by the removal of the damaged part. Allow a clearance of not over 0.06 inch between each end of the insertion member and the original member. Repair members, as shown in the sections are used at each end of the insertion member to attach it to the original member, thus providing an even load distribution between the insertion and the original existing members. These insertion repairs are made exactly as instructed in the repair illustrations shown in the manual, except that no filler is used and the insertion member replaces a part of the original member. For example, assume that repair members installed to patch a stringer would interfere with a rib, making it necessary to rework the rib. An easier repair can be made by cutting out the part of the existing stringer passing through the rib and installing an insertion member for the removed part of the stringer. This example will also work for longerons and frames.

59.1.4.4 REPAIR OF BUILT-UP STRUCTURAL MEMBERS. When it becomes necessary to repair any of these built-up members in the reinforced area, or

59.1.4.4 (Cont.)

within a "tie-in" length of the reinforced area (refer to Section 59.1), classify and repair as follows:

a. If the damage is to reinforcing member(s) (such as straps or angles) only, replace the member with its equivalent, or repair it according to the appropriate repair instructions in Section 59.8.

b. If the damage is to the basic original member, repair it as follows:

   1. Remove all added-on pieces such as straps or angles.

   2. Cut out and remove the part of the basic original member to which the additional pieces were attached plus a working allowance at each end. The working allowance will be equal in length to the "tie-in" length as defined in Section 59.1.

Note

If the remaining amount of the basic original member is not long enough to permit the installation of the repair member according to the appropriate repair illustration, the entire member must be replaced.

   3. Select, fit, and install a suitable insertion member.

   4. Reinstall the reinforcing straps, angles or similar parts. If the original reinforcing members are not usable, replace them with equivalent parts.

59.1.4.4 (Cont).

c. When it becomes necessary to repair any of these members in an area other than the one just mentioned, use the same procedures as for any other simple, non-built-up member.

59.1.4.5 DAMAGE NECESSITATING REPLACEMENT OF PARTS. Damage beyond the negligible classification which cannot be repaired by patching or insertion is classified as necessitating replacement of parts. Unless directed otherwise on the appropriate structure or repair illustrations, replace existing doublers, splices, and clips which have been damaged beyond negligible limits. Damage beyond the negligible classification to parts which are short (less than 30 inches in length) is classified as necessitating replacement of parts. Repair of short members is generally not practical. A saving in repair time usually can be accomplished by replacing short members. Areas of some highly stressed members cannot be repaired, because the repaired members would not have sufficient margin of safety. Members with shapes which cannot be repaired by patching or insertion must be replaced when damaged in excess of negligible limits.

59.1.4.6 SPECIFIC REPAIRS. A specific repair is a repair designed for damage to a particular structural member. The specific repair illustration gives all information that is needed concerning the size of the repair member and the size, type, and number of attachments. When a specific repair is available, a reference is given in the structural key opposite the structural member for which the repair is designed. Specific repair illustrations are given in the applicable section.

59.1.4.7 TYPICAL REPAIRS. A typical repair is a repair designed to be adapted for use on different structural members with the same general cross sectional shape, even though the cross section dimensions may vary. Typical repairs are given in Section 59.8. Repairs are shown for members with various cross sections which are used in many applications throughout the aircraft. With each repair illustration, information is given on how to determine the size of the repair member and the number, size, and type of attachments to be used.

59.1.4.8 ONE-TIME FERRY FLIGHT REPAIRS. In some instances it will be necessary to make repairs to a damaged aircraft when adequate facilities are not available for permanent repairs. One-time ferry flight repairs are made in order to fly the aircraft to a base where adequate facilities are available for permanent ferry repairs. It is not mandatory that one-time ferry flight repairs fully restore the original load-carrying ability or aerodynamic characteristics to the aircraft.

59.1.4.9 TEMPORARY REPAIRS. Temporary repairs are repairs which return the aircraft to its original strength structurally but do not restore it aerodynamically. These repairs are accomplished by the use of external patches and protruding-headfasteners in areas of the aircraft where flush repairs and flush fasteners are required to restore the aircraft to its original aerodynamic configuration. These repairs are called temporary because they will be replaced by permanent repairs at some future date by higher level maintenance.

59.1.4.10 PERMANENT REPAIRS. Permanent repairs are repairs which return the aircraft to its original strength structurally and to its original configuration aerodynamically. Most of the repairs in this manual are of this type.

59.1.4.11 PARTIAL REPAIRS. The repair illustrations show the maximum repair to the cross section of any member. If the cross section of the member is not damaged to the extent shown on the repair illustration, use only as much of the repair as necessary to restore the strength of the original member. If in doubt, use the full repair. Partial repairs apply only to partially damaged internal structure.

59.1.4.12 MAJOR REPAIRS. Repairs to primary structural members involving the strengthening, reinforcing, splicing, and manufacturing of primary structural members or their replacement when replacement is by fabrication such as riveting or welding are major repairs. Refer to Federal Aviation Regulations, Part 43 for specific airframe parts and types of repairs. When a repair involves restoring one or more primary structural members to original load carrying requirements and/or aerodynamic characteristics, it is imperative that adjacent structural members be held in alignment with adequate fixtures while the repair is being made.

59.1.5 AERODYNAMIC SMOOTHNESS. In the interest of operating efficiency, it is important that the exterior surfaces of the airplane are maintained as smooth as possible. See figure 59-8 for limits on surface irregularities which should be adhered to when making repairs or replacements which involve the exterior surface in any way.

59.1.6 CLEANING EXTERIOR SURFACES. In order to conduct an investigation of damage, it is necessary that the exterior surfaces of the airplane be clean. Use the following procedure for normal cleaning of the airplane exterior surfaces.

a. Before beginning cleaning operations, make sure all access doors and windows are closed. Protect lubricated parts (exposed flap and control actuators and bearings, wheel, brake, etc.) louvers, scoops, and other natural entrapments from the cleaning solution by masking. Masking tape and barrier paper can be used effectively for masking purposes.

Note

Areas requiring lubrication should be relubricated after cleaning operations.

b. Prepare the proper mixture of Specification MIL-C-25769 concentrate and water. For painted and unpainted surfaces, dilute one part of concentrate in

59.1.6 (Cont).

seven parts of water. For areas that are difficult to clean, the solution may be increased to one part of concentrate to three parts of water. In some cases, undiluted concentrate may be used to clean areas that are extremely difficult to clean.

CAUTION

Do not use undiluted concentrate or allow diluted concentrate to dry on plastics, plate glass, or painted surfaces.

Note

Conduct cleaning operations in the shade, if possible, during hot weather (above 80° F). Wetting the surfaces to be cleaned with water before applying the cleaning solution will tend to cool the surface and prevent fast drying of the cleaning solution.

c. Apply the cleaning solution until the area to be cleaned is completely covered. Begin cleaning at the lowest surface on the airplane.

d. Agitate the compound by scrubbing the surface with a nonmetallic, soft-bristle brush. Vigorous scrubbing action with a brush is not usually required except in extremely soiled areas. Depending on weather conditions, allow approximately 5 to 10 minutes after applying the cleaning solution before rinsing.

Note

Do not allow the surface to dry before rinsing.

e. Thoroughly rinse all cleaning solution from the airplane using large amounts of water.

f. If streaking appears, reapply the cleaning solution (one part of concentrate to three parts of water) as soon as possible after streaking is noticed, and repeat step (e).

g. If the first cleaning operation does not produce a clean surface, repeat the operation.

h. Dry windows with a clean chamois. A soft, clean cloth or a soft tissue may be used if care is taken not to continue rubbing after the window is dry.

59.1.7 CORROSION CONTROL. Refer to Section 51.7 for corrosion control, removal, inspection, and treatment on the JetStar.

59.1.7.1 CORROSION PREVENTION FOR REPAIRS. See Section 51.7 for the steps to be followed to prevent corrosion to repairs.

59.1.8 AIRFRAME SEALING. Refer to Section 51.2 for airframe sealing and for detailed instructions for sealant removal and sealant replacement in repair areas.

59.1.9 PRESSURE TESTING. Extensive repair to the fuselage pressure shell will require a pressure test to be run on the aircraft. Refer to Section 51.4 for structural proof and leakage tests.

59.1.10 CONTROL SURFACE REBALANCING. The elevators, ailerons, and rudder are mass-balanced control surfaces. Repairs to these surfaces can change their mass balance enough to exceed certain tolerance limits, beyond which the safety of the aircraft is affected. Both static and dynamic balance have been carefully worked out in the design of the control surfaces and they must be within tolerance limits after each repair to any of the control surfaces. Refer to Section 51.5 for detailed instructions for the mass rebalancing of the control surfaces.

59.1.11 GENERAL SHOP PRACTICES.

59.1.11.1 SHEET METAL FORMING. The sheet metal bend radii listed in figure 59-9 are standard and minimum. They should be used whenever it is necessary to fabricate sheet metal for repairs to the aircraft, except when otherwise specified on the repair illustration.

59.1.11.2 SOLID RIVET HOLE SIZES. When drilling new holes for solid rivets, use figure 59-11 to determine the correct hole size. Hole sizes for blind rivets, blind bolts, Hi-Loks, and lockbolts can be found in the applicable installation illustration.

59.1.11.3 LOCATING BLIND HOLES. When patching or replacing sections of airplane structure, the holes in the replacement sheet or patch must be drilled to match existing holes in the structure. Existing holes may be located by using a hole duplicator similar to that shown in figure 59-12. The peg on the bottom leg of the hole duplicator fits into the existing rivet hole. A pilot hole in the new part is drilled in alignment by drilling through the bushing in the top leg. A separate hole duplicator must be used for each diameter of rivet used.

59.1.11.4 DRILLING TITANIUM AND STEEL. Drilling titanium, hard steels, and stainless steels is extremely difficult when using standard high-speed drill bits. It is recommended that a drill bit ground to the dimensions given in Specification NAS907 be used for drilling these materials.

59.1.11.5 HOLES FOR FLUSH FASTENERS. Holes for flush head skin fasteners are to be prepared by countersinking. Dimpling thin sheets for flush fasteners is a severe forming operation and is rarely used on the JetStar. The use of dimples is not recommended for structural repair operations.

59.1.11.6 COUNTERSINK LIMITS. The limits given in figure 59-11 are applicable to all repairs requiring flush fasteners in new fastener holes.

59.1.12 FASTENERS.

Specific data is given in this section for NAS1097 reduced head flush rivets, NAS1398 and NAS1399 blind rivets (with locked spindle) NAS1738 and NAS1739 blind rivets (with locked spindle and bulbed end), NAS1669 and 1670 blind fasteners (Jo-bolts), Hi-loks and NAS pull-type lockbolts.

59.1.12.1 NAS 1097 RIVETS. NAS 1097 rivets are the same as MS 20426 standard flush rivets except for the reduced head size. See figure 59-13 for the comparison between these rivets. The NAS 1097 rivets are used to permit countersinking of thin skins; however, do not enlarge the countersink for an NAS 1097 rivet to accept a MS 20426 rivet. This would violate the countersinking limits given in the table in figure 59-11. When necessary to make a substitution, the same diameter and alloy MS 20426 rivet may be driven into an existing prepared hole for an NAS 1097 rivet, and the head may then be shaved flush.

59.1.12.2 BLIND RIVETS. These rivets, made by Cherry Fasteners and others, are designed for riveting where access is available to only one side of the work. When installed, they have a mechanically locked spindle which prevents the pin from working out of the sleeve under vibration. For fastener substitution, refer to paragraph 59.1.15.

59.1.12.3 BLIND BOLTS. These bolts are high-strength, blind fasteners made by Monogram Aero-Space Fasteners and others. All components of each unit (screw, nut and sleeve) are completely assembled and ready for use.

59.1.12.4 HI-LOKS. Hi-loks are manufactured by the Hi Shear Rivet and Tool Company. Each fastener consists of a threaded pin and threaded collar as shown in figure 59-16. Hi-loks may be procured with aluminum alloy, alloy steel or stainless steel pin materials. Hi-lok pins may be reused with new collars. Replace a hi-lok with another unit of comparable size, type, and material, except as noted in figure 59-16 and paragraph 59.1.15.

59.1.12.5 LOCKBOLTS. (See figure 59-19.) Extensive use is made of pull-type lockbolts made by Huck Manufacturing Company and others and identified by NAS numbers. Each unit consists of a headed pin (bolt) secured by a collar. Replace a lockbolt with another unit of comparable size, type and material. Special tools are required for this operation. Take care not to damage or enlarge the hole during removal.

CAUTION

Do not substitute stump type for pull-type lockbolts; however, pull-type lockbolts may be substituted for stump-type lockbolts if they are installed.

59.1.13 INSPECTION OF INSTALLED FASTENERS.

The inspection criteria for NAS1398, NAS1399, NAS1738 and 1739 blind rivets, blind bolts, Hi-loks, and pull-type lockbolts are as follows:

a. Blind rivet installations are inspected using the data given in figure 59-14. If the pin is too long or too short, remove the rivet and reinstall a rivet of the correct grip length.

b. Blind bolt (Jo-bolts) inspection:

   (1) The stem break-off position of the screw in the head of the nut should be within the limits specified in figure 59-15. If these limits are exceeded, the blind bolt should be removed.

   (2) Check the blind bolt for looseness. The nosepiece of the installation tool may be adapted to a torque wrench. All fasteners must resist the specified torque without turning. The torque must be applied counterclockwise only.

c. Hi-lok inspection is to be accomplished using the data given in figure 59-16. Installed standard Hi-lok pins are to be checked by dimension "B." If dimension "B" is below the minimum or if it exceeds the limits given in figure 59-16, replace the Hi-lok fastener. Investigate the possibility of incorrect grip length as the cause for rejection.

d. Lockbolt inspection by visual means is usually satisfactory. If in doubt or if controversy arises, use the gage shown in figure 59-18.

59.1.14 INSTALLATION OF FASTENERS.

The following paragraphs describe the removal and the installation of blind rivets, blind bolts, Hi-loks, and lockbolts.

59.1.14.1 BLIND RIVETS.

a. Removal of blind rivets is similar to the removal of solid shank rivets and is accomplished as follows:

   (1) File a flat on the exposed end of the pin.

   (2) Center punch the pin.

   (3) Drill through the tapered portion of the pin forming the lock; use a drill one size smaller than the pin diameter.

   (4) Insert a drift punch, and pry the remainder of the locking collar out of the rivet head.

59.1.14.1 (Cont).

Note

It is necessary to remove the locking collar, otherwise the collar will rotate with the drill in the next operation.

(5) Drill nearly through the rivet head, using a drill one size smaller than the rivet sleeve diameter.

Note

Do not damage the skin or drill into the edge of the rivet hole.

(6) Insert a drift punch in the hole drilled in the rivet head, and tilt the punch to break off the weakened head.

(7) Punch out the remainder of the rivet with a drift punch having a diameter equal to or slightly less than the nominal diameter of the rivet.

b. Installation of blind rivets is accomplished using a power pull gun as follows (see figure 59-19):

(1) Insert the rivet assembly into the hole.

(2) Engage the rivet pin with the pulling head of the gun, and press the outer anvil against the rivet head.

(3) Depress the trigger. As the gun pulls the pin into the rivet sleeve, the sleeve expands to fill the hole. As the pull continues, the sleeve is upset. The inner anvil forces the locking collar into the conical space between the recess in the head and the locking groove in the pin, and the pin fractures at the breakneck groove. The broken section of the pin is ejected from the gun.

c. Correct holes for blind rivets are important to obtain a strong, rigid joint. The expanded rivet should completely fill the hole. To assure a proper filling, the holes should be round and within the size limits specified in figure 59-14. The surest and simplest method of automatically obtaining holes of the correct size when drilling in solid metal is as follows:

(1) Pre-drill with a drill slightly smaller in diameter than the desired rivet drill. The usual pilot drill (No. 40) is too small in diameter to serve as a predrill.

(2) Size-drill to the rivet hole diameter. Pre-drill and size-drill sizes for standard and oversize blind rivets are given in figure 59-14.

59.1.14.2 BLIND BOLTS.

a. Removal of installed blind bolts is accomplished as shown in figure 59-15.

b. Installation of blind bolts is as follows (see figure 59-15):

(1) Insert the blind bolt into a properly prepared hole.

Note

The grip length of the blind bolt must not exceed the material thickness by more than 1/16 inch.

(2) The wrenching part of the adapter assembly, whenever using a pneumatic or a hand tool, is inserted over the slabbed portion of the screw, and the nose piece engages the nut.

CAUTION

Tilting of the driver may break the stem before the blind bolt is tight.

The driving tool must be held tightly against the head of the blind bolt and must be perpendicular to the work.

(3) As power is applied, the screw turns while the nut is held in place. This causes the sleeve to expand over the tapered end of the nut until it contacts the material being fastened. As driving is completed, the slabbed portion of the screw is snapped off and rejected.

c. The final hole sizes for blind bolt installation are given in figure 59-15. Follow the same basic steps as outlined in paragraph 59.1.14.1 to assure the correct size hole for blind bolt installation.

59.1.14.3 HI-LOKS

a. The Hi-lok fastener removal procedure is shown in figure 59-16, and is described as follows:

CAUTION

Do not use a chisel to split the collar because of possible damage to the hole, structure, and threads of the Hi-lok pin.

(1) Use a pair of pliers to remove the collar. Prevent the Hi-lok pin from turning with an Allen hex-key wrench.

(2) Carefully drive out the Hi-lok with a pin punch.

(3) Check the hole diameter against limits given in the Hi-lok installation data table, figure 59-16. If tolerances are exceeded, use a 1/64th oversize Hi-lok.

59.1.14.4 HI-LOK INSTALLATION. This may be accomplished by the use of either hand or power driven tools (see figure 59-16). Ratchet-type hand tools and special ratchet-wrench adapter tools may be purchased from the Hi-Shear Rivet and Tool Company, Torrance, California, or equivalent from other sources. The ratchet-wrench adapters can be fitted to high speed power drivers.

a. Install Hi-lok fasteners with hand tools as follows:

(1) Insert the pin into the prepared hole. Check the countersink depth, if applicable.

(2) Manually screw the collar onto the pin a minimum of two threads.

CAUTION

No washers are permitted under the manufactured head of the pin to adjust grip length. A maximum of two AN960D washers may be used under the collar for this purpose.

(3) Insert an Allen hex-key into the hex recess of the Hi-lok pin.

(4) Hold the hex-key firmly to prevent the Hi-lok from turning. Apply a wrench on ratchet-socket assembly to the collar hex, (see figure 59-16). Tighten the collar until the collar hex is broken off.

b. Install Hi-lok fasteners with special ratchet-wrench type power tools as follows:

(1) Insert the pin into the prepared hole. Check the countersink depth, if applicable.

CAUTION

No washers are permitted under the manufactured head of the pin to adjust grip length. A maximum of two AN 960D washers may be used under the collar for this purpose.

(2) Manually screw the collar onto the pin a minimum of two threads.

(3) Insert the hex wrench tip of the power tool into the hex recess of the Hi-lok pin.

(4) Firmly press the power tool against the collar hex, and operate the tool until the collar hex breaks off.

Note

Be sure the power tool hex socket engages the collar hex before operating the tool.

c. The final hole sizes for Hi-lok fasteners are given in figure 59-16. Follow the same basic steps as outlined in paragraph 59.1.14.1 to assure the correct size hole for the Hi-lok pins.

59.1.14.5 LOCKBOLT REMOVAL.

a. Removal of pull-type and stump-type lockbolt fasteners is shown in figure 59-19 and is described as follows:

(1) Remove the collar by splitting it shankwise with a collar splitter (see figure 59-19). Use collar splitter tool 42-6 for removal of 3/16-inch diameter pin collars, 42-8 for removal of 1/4-inch diameter pin collars, 42-10 for removal of 5/16-inch diameter pin collars, and 42-12 for removal of 3/8-inch diameter pin collars. The tools are distributed by the Huck Manufacturing Company, Detroit, Michigan.

(2) Drive out the lockbolt with a pin punch.

(3) Check the hole diameter against the tolerances, as shown in figure 59-19 for the correct fit, "Interference" or "Clearance."

59.1.14.6 INSTALLATION OF PULL-TYPE LOCKBOLTS.

a. Installation of pull-type lockbolts requires special equipment specified in the referenced figure in the table entitled "Installation Tools." The equipment is distributed by the Huck Manufacturing Company, Detroit, Michigan. The mechanical procedure for installing pull-type lockbolts units is shown in figure 59-19. Regular lubricated lockbolts may be used in wing integral fuel tank areas.

Note

When removal of a stump-type lockbolt is required, substitute a pull-type lockbolt or a Hi-lok of equal or higher strength and the same size and fit as a replacement.

b. Special hole sizes are required for pull-type lockbolts. See figure 59-19 for predrill, and interference or clearance fit size drill for both the nominal and 1/64th-inch oversize fasteners.

59.1.15 FASTENER SUBSTITUTION.

Obtain the aircraft manufacturer's approval before an attempt is made to use an oversize fastener of any kind except a rivet. In a series of rivets no more than one in 10 can be oversize, and then only the next larger standard size can be used, except as noted for blind rivets in the following paragraph. No two oversize rivets may be next to each other, and an edge distance of one-and-a-half-times the oversize rivet diameter must be maintained. The size of rivets used to attach fittings cannot be varied without the approval of the aircraft manufacturer.

a. Blind rivets may be substituted for friction lock blind rivets (Cherry rivet or equivalent) on a one-to-one replacement basis. Friction lock blind rivets (Cherry rivet or equivalent) will not be substituted for blind rivets. It is recommended that the 1/64th-inch oversize blind rivet be used when replacing loose friction lock blind rivets.

b. Hi-lok fasteners substitutions are specified in figure 59-16. Use of substitute fasteners must be approved by the aircraft manufacturer, since some Hi-lok fasteners used are higher strength fasteners than the substitutes shown in figure 59-16.

Note

Light drive fits are required for bolts and screws that are used to replace Hi-loks. This may require select-fit application.

c. Substitutions for the pull-type lockbolts are specified in figure 59-19, Sheet 2, Note 1.

59.1.16 FILLING MACHINE COUNTERSUNK HOLES.

On various occasions, it will be necessary to make external skin repairs on aerodynamic surfaces. Before repairing the damaged area, it will be necessary to remove existing fasteners. Removal of existing fasteners will leave machine countersunk holes that must be filled, before attaching the repair patch, in order to eliminate void spots and insure proper sealing. Fill machine countersunk or dimpled holes as follows:

a. Select a fastener of the same type and size as that removed from the damaged area.

b. Clamp the fastener selected in step (a) in a drill vise with the shank pointing down.

c. Pilot drill through the exact center of the fastener. Use a drill press for the drilling operation, if available.

59.1.16 (Cont).

d. Use a larger drill, and increase the pilot hole to full size. The shank should separate from the head after drilling full size. This is the hole filler.

e. Deburr the filler made in steps (a) through (d).

f. Paint the hole to be filled with zinc chromate primer. Allow the primer to become tacky before inserting the hole filler.

g. Insert filler in the hole by pressing firmly with thumb. The head of the filler must be flush with the skin surface.

Note

Holes in the repair patch must be located and drilled before inserting fillers in holes.

59.1.17 CRASH HANDLING.

59.1.17.1 SAFETY PRECAUTIONS. Before any ground handling can be started, the following precautions must be taken.

a. Ground the aircraft.

b. Wash away any fuel or hydraulic fluid leakage.

c. Completely defuel and purge the fuel system as follows:

1. Drain all fuel from the tanks and lines and air-purge the tanks, using low-velocity, high-volume air at a maximum temperature of 120° F.
2. If it is anticipated that the aircraft will be inactive longer than a few days, apply engine oil, Grade 1005 or 1010, to the interior tank surfaces. All parts containing rubber seals must be covered with the oil . Flush the system to accomplish this and then drain the excess oil.

d. Remove the aircraft batteries.

e. Remove all oxygen containers from the aircraft, and bleed any oxygen from the associated lines.

f. Remove all cargo and equipment carried in the aircraft.

59.1.17.2 GROUND HANDLING. Ground handling of the aircraft after a crash landing, on or away from the runway, is critical due to the size of the aircraft and the high weights involved. If the aircraft is not or cannot be made transportable on its main and nose gear, use of a sling Lockheed Part Number 11007-MES) will permit standard hoisting equipment to move the aircraft to a desirable repair location. The use of air bags, fuselage and/or wing support cradles (see figure 59-20) and hydraulic jacks should be considered for raising and for supporting the aircraft either temporarily or during repair. Where possible, standard maintenance procedures and crash handling techniques are to be followed; however, firm procedures cannot be established since each situation must be evaluated individually.

59.1.17.3 CRITICAL JACKING AND TOWING CONDITIONS. Removal of some of the components could cause unsafe lifting or towing conditions. Keep the center of gravity position at fuselage station 484 (approximately) for safe operations. This center of gravity position will give a nose gear reaction of about 2100 pounds. Ballast, in the form of bags of lead shot or other suitable materials, distributed over the center aisle in the passenger cabin will aid in maintaining the correct center of gravity.

59.1.17.4 HOISTING, LIFTING, JACKING AND TOWING PROCEDURES. Jacking of the aircraft and hoisting for the empennage and engines is covered in Chapter 7. Towing procedures are described in Chapter 9. It is recommended that the Lockheed Western Export Company (see page Introduction-3) be contacted for assistance if lifting and hoisting of the complete aircraft or major components of the aircraft is necessary.

59.1.17.5 PROTECTION OF CRASH DAMAGED AIRCRAFT FROM CORROSION. Coat exposed bearing surfaces and critical steel parts with Specification MIL-G-23827 grease. Cover rents and tears in the skin, as well as unprotected openings left by removal of airframe components, to prevent entry of rain water and foreign material. Tape loose ends of electrical connectors. Plug and wrap open ends of all ducts and tubing.

59.1.18 SUPPORT OF STRUCTURE DURING REPAIRS.

Damaged structural parts must be restored to the original shapes and positions during repair. Avoid distortion by constructing welded or bolted steel frames to serve as jigs. If steel is not available, use material as rigid and as dimensionally stable as possible. Before starting repairs, take measurements between reference points on related undamaged parts of the airplane. These measurements must coincide with identical reference points in the damage area. Check frequently during repairs to make sure that there is no distortion.

59.1.18.1 FUSELAGE SUPPORT CRADLES. Make major repairs to the fuselage only after supporting and leveling the airplane. (See figure 59-20). A suggested design for the principal fuselage support cradles is shown on figure 59-20. These cradles are to be used in conjunction with the wing jacks to support the airplane while making major structural repairs to the fuselage. Additional supports may be needed, depending on the nature and extent of damage. If additional supports are needed, they may be made similar to those shown on figure 59-20; contour

59.1.18.1 (Cont).

dimensions will vary however, depending on the locations in which they are to be used.

CAUTION

Care must be taken when locating cradles, to be sure the airplane structure at the point of contact is strong enough to support the load without being damaged.

59.1.18.2 WING SUPPORT CRADLES. Cradles (Lockheed Part No. 11020 MEC) are available to support the wings during removal of the wing panels and during major structural repair to the wings.

59.1.19 SERVICE LIFE CRITERIA.

59.1.19.1 PERMISSIBLE WEAR AND DEVIATIONS. Wear is one of the principal damages to an aircraft in normal service. Allowances for wear have been designed into all components of the airframe. Bushings that can be removed and replaced are installed in fittings that are subject to wear caused by relative motion.

59.1.19.2 WEAR INDICATION. The end of service life is accompanied by excessive wear which is indicated by displaced material, ruts and grooves, misalignment, and malfunctioning equipment. The areas mainly affected are, (1) attaching points of structural members subject to varying loads, and (2) points of relative motion. When inspecting for wear damage, particular attention should be given to bushings, hinges, tracks, fasteners, and fittings. Allowable wear damage is expected, but evidences of rapid or excessive wear should warrant a complete inspection of related and attaching equipment. Evidences of excessive wear damage may be indicated in any of the following ways:

CAUTION

Do not allow wear damage to progress to the extent that it will result in malfunction of a component.

a. Elongated or enlarged holes.

b. Bent, broken, or deformed fasteners.

c. Brinelling or failed material.

d. Misalignment.

e. Binding, sticking, or erratic movement of equipment.

f. Squeaks, rattles, or chatter.

g. Improper seating.

59.1.19.3 REAMING AND BUSHING. During a repair, all holes are to be examined for elongation or enlargement. Fasteners are to be checked for deformity and tightness. If wear damage exists in the joint, determine whether repair is possible or whether parts should be replaced.

59.1.19.4 REAMING HOLES. When replacing worn or damaged bushings, the hole in the assembly normally will not require reaming. However, if upon removal of the bushing it is found that the hole is badly scored, elongated, nicked, or otherwise imperfect, it will be permissible to ream the hole 1/64 inch oversize. Replacement of the part or the aircraft manufacturer's approval will be required if more material than 1/64 inch on the diameter is removed for cleanup of the hole.

a. Where bushing alignment is critical, the inside diameters of bushings should be line-reamed after installation. Ream by machine, if possible, as it is difficult to obtain satisfactory results with a hand reamer. The finished inside diameter of the replacement bushing must be the same as in the original bushing as designed. When hand reaming is necessary, ream as follows:

   1. Clamp the assembly securely in a vise or other holding device, and position so as to provide clearance for operating the reaming tools.

   2. Insert the correct size reamer into a tap wrench or other driving tool.

   3. Keep the reamer directly in line with the sides of the hole in the assembly, and ream to the dimension applicable for the particular bushing.

59.1.19.5 BUSHING REMOVAL. Bushings may be removed with tools as shown in figure 59-21. The use of these tools will depend on the shape of the part or fitting and upon clearance if the part is not removed from the aircraft. Care should be taken so as not to damage structure with the tools.

59.1.19.6 BUSHING INSTALLATION. (See figure 59-21. Do not install oversize bushings except for the 1/64-inch oversize condition.

Note

The aircraft manufacturer's approval will be required for the installation of bushings in holes that did not originally contain bushings.

a. Make replacement bushings from the same material as the original bushings, and heat-treat replacement bushings as necessary to obtain the same hardness and strength as the original

59.1.19.6 (Cont).

bushings. Use instructions in this section as necessary to determine the material and strength of the original bushing. Where bushing alignment is critical, line-ream the bushing blank after installation. Surface finish should be 40 micro-inches at movable or rotating connections. Duplicate shoulders and oil grooves in replacement bushings, when such are provided in original bushings. The ends of a shoulderless bushing must not extend beyond the surface of the fitting; no end thrust on this type bushing is permissible.

b. The three methods for bushing installation are listed according to preference: the arbor press, the drawn-in, and the vise methods. The arbor press method is most desirable because it provides greater control of the driving force with less danger of damaging the bushing or assembly.

Note

Installation of bushings by any of the three methods can be facilitated by cooling the bushing. Do not apply heat to the assembly unless approved by the aircraft manufacturer.

c. The arbor press method for installing bushings is accomplished as follows:

1. Place the assembly on an arbor press, and center the hole which is to receive the bushing over the press bed.
2. Place the chamfered edge of the bushing in the hole in the assembly. Carefully align the bushing with the sides of the hole so that when driving pressure is applied, the bushing will be squarely started into the hole.
3. Place a mandrel or other installing device in the new bushing.
4. Operate the actuating lever of the arbor press to bring the ram down against the mandrel.
5. Hold the assembly firmly in place on the press bed, and with a steady pull on the actuating lever, press the bushing into place.
6. Release pressure and remove the mandrel.

d. The draw-in method for installing bushings is accomplished as follows:

1. Use a puller or pusher arrangement comparable to that shown in figure 59-21.
2. If a puller or pusher is not available, use a combination of stud or bolt, with appropriate size washers and nut, and tighten with a wrench. Align the bushing with the sides of the hole.

e. The vise method for installing bushings is accomplished as follows:

1. Perform steps (1), (2), and (3) of paragraph c, using a machinist's vise instead of an arbor press.

Note

Due to the limited opening of the jaws of most machinists' vises, the mandrel must be relatively shorter or must be eliminated entirely. The shape of the part will regulate the use of the mandrel. To prevent damage to the part or to the bushing, it is preferable to use jaw pads of soft material. Be sure the bushing is started in an aligned position or damage to the bushing and assembly will result.

2. Operate the handle of the vise until the bushing is pressed into place.
3. Open the vise and remove the part.

59.1.20 REPLACEMENT ITEMS. Cotter pins and lockwire are not to be reused; they should be replaced whenever a component is disassembled. All fasteners, pins, and parts damaged beyond repair or rework will be replaced with like components of the same or comparable material and heat-treat specification. When any component is replaced, all clearances and tolerances must be checked for conformance to the original part or to the design value.

Note

Do not reuse lockbolt fasteners. Undamaged Hi-Lock pins may be reused with new collars.

59.1.21 EPOXY INSTALLATION OF BEARINGS.

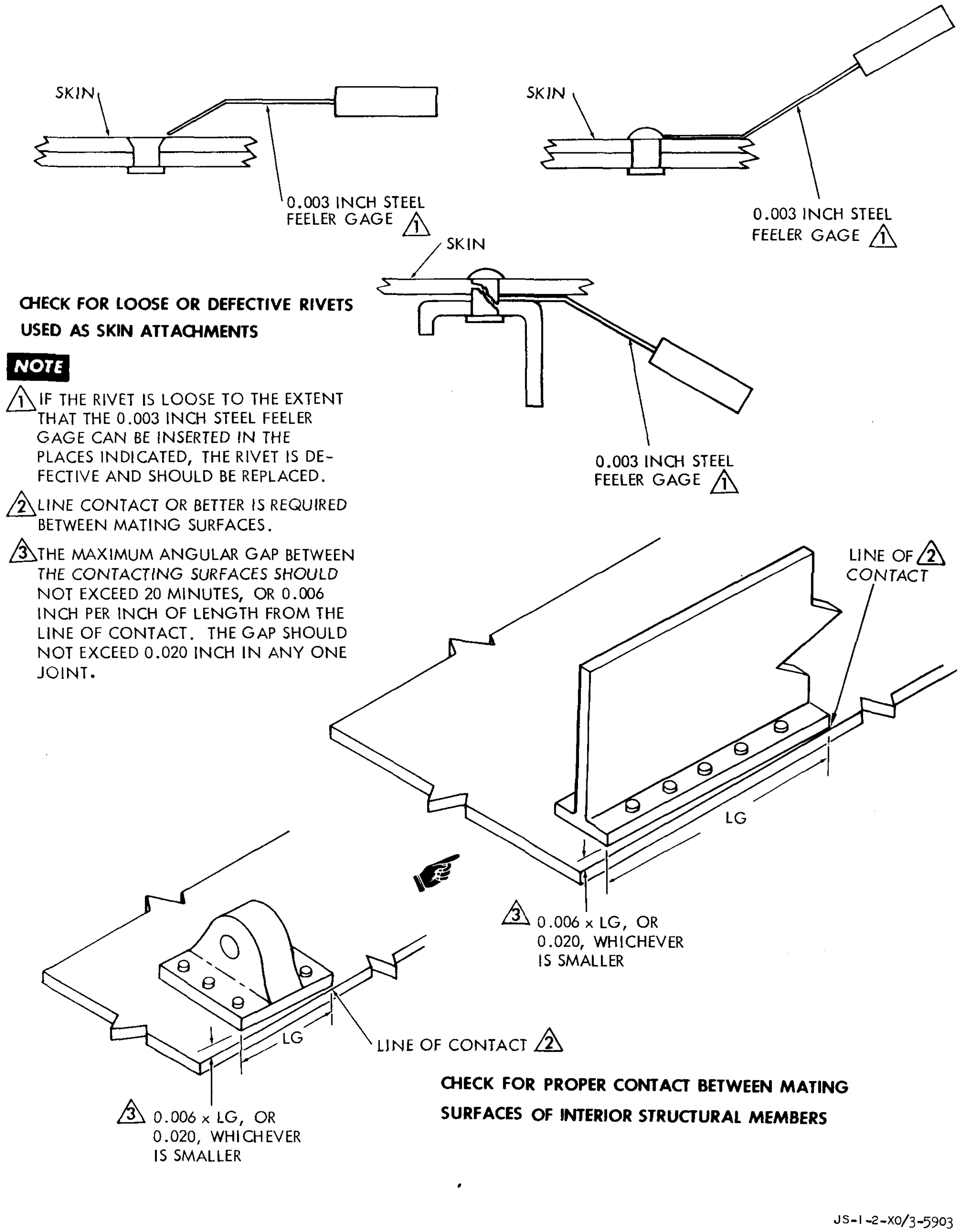
a. The journal ID where the bearing is to be bonded shall be 0.001 to 0.005 inch larger than the OD of the bearing.

b. Surfaces to be bonded will be lightly sanded with 220 to 280 grit sandpaper and cleaned with methylethylketone (MEK). Cleaning of pre-lubricated bearings will be done with care to prevent the cleaner from entering the lubricated area of the bearing. Cleaned areas will be handled with clean, cloth gloves rather than with bare hands.

c. Mix and handle 3-M Corp., EC 3569, epoxy adhesive (Alt BF Goodrich 1177, EA9309.1) in accordance with the manufacturer's instructions, observing limits on work life and storage.

d. Apply adhesive to outer bearing surface and to housing bore as shown in figure 59-22.

e. Allow adhesive to cure in accordance with manufacturer's recommendations.


SKIN
0.003 INCH STEEL FEELER GAGE 1
SKIN
0.003 INCH STEEL FEELER GAGE 1
SKIN
0.003 INCH STEEL FEELER GAGE 1
LINE OF 2 CONTACT
LG
3 0.006 x LG, OR 0.020, WHICHEVER IS SMALLER
LG
LINE OF CONTACT 2
3 0.006 x LG, OR 0.020, WHICHEVER IS SMALLER
JS-1-2-X0/3-5903

CHECK FOR LOOSE OR DEFECTIVE RIVETS USED AS SKIN ATTACHMENTS

NOTE

1. IF THE RIVET IS LOOSE TO THE EXTENT THAT THE 0.003 INCH STEEL FEELER GAGE CAN BE INSERTED IN THE PLACES INDICATED, THE RIVET IS DEFECTIVE AND SHOULD BE REPLACED.
2. LINE CONTACT OR BETTER IS REQUIRED BETWEEN MATING SURFACES.
3. THE MAXIMUM ANGULAR GAP BETWEEN THE CONTACTING SURFACES SHOULD NOT EXCEED 20 MINUTES, OR 0.006 INCH PER INCH OF LENGTH FROM THE LINE OF CONTACT. THE GAP SHOULD NOT EXCEED 0.020 INCH IN ANY ONE JOINT.

CHECK FOR PROPER CONTACT BETWEEN MATING SURFACES OF INTERIOR STRUCTURAL MEMBERS

Figure 59-3. Rivet and Joint Inspection

ALLOY DESIGNATION	PRODUCT	MINIMUM ROCKWELL (EXCEPT AS NOTED)
2014-0	All	E55 △1
2014-T6, T651, T6511	All Bare	B78 or E102
2024-0	All	E55 △1
2024-T3, T351	All Bare	B69 or E95
2024-T3, T42	Clad 0.063 & under	B63 or E91
	Clad 0.064 - 0.099	B63 or E92
	Clad 0.100 - 0.187	B65 or E96
2024-T4, T42	All Bare	B69 or E95
2024-T6	All Bare	B75
2024-T62	Clad 0.062 & under	B60
	Clad over 0.062	B62
2024-T81	All Bare	B78
2219-0	Bare 0.040 - 0.249	E71 △1
2219-T6	Bare 0.040 - 0.249	B62 or E94
2219-T62	Clad 0.019 - 0.039	E87
	Clad 0.040 - 0.099	E90
	Clad 0.100 - 0.249	E95
2219-T81	Clad 0.062 & under	B64
	Clad over 0.062	B63
6061-0	All	H70 △1
6061-T4, T451, T4511	All	E70
6061-T6, T651, T6511	All	B47
7049-T73	Die Forgings	B81
7075-0	All	E67 △1
7075-T6	Die Forgings thru 3.00	B85
	Bare 0.040 - 0.249	B85
	Clad 0.036 & under	B76
	Clad 0.037 - 0.062	B78
	Clad 0.062 - 0.187	B79
7075-T73	Die Forgings thru 3.00	B77
7075-T73, T735, T7351	Sheet 0.040 - 0.249	B78
	Plate 0.025 - 2.00	B80
	Plate 2.001 - 2.50	B77
	Plate 2.501 - 3.00	B76

Figure 59-4. Rockwell Hardness Scale (Sheet 1 of 2)

ALLOY DESIGNATION	PRODUCT	MINIMUM ROCKWELL (EXCEPT AS NOTED)
7075-T651	Plate 0.250 - 2.00	B85
	Plate 2.01 - 2.50	B82
	Plate 2.51 - 3.50	B80
	Plate 3.51 - 4.00	B78
	Bar & Rod thru 4.00	B86
7075-T6511	Extrusion thru 0.250	B87
	Extrusion 0.251 - 5.00	B88
7075-T7351	Bar & Rod thru 4.00	B77
7075-T76511	Extrusions	B82
7079-T6	Forgings	B83
7178-T6, T651, T6511	Extrusions thru 0.249	B86
	Extrusions over 0.250	B87
301 Stainless △2	Annealed Sheet	B85
	1/4 Hard Sheet	C25
321	Sheet	B85
Aged A286	Sheet, Bar, Forging	C24
AM350	Doubled Aged Sheet	C37
	STC 850 Sheet	C40
17-4PH H950	Bar, Forging, Plate	C37
17-4PH H1000	Castings	C38
17-7PH TH1050	Sheet	C39
Low Alloy Steel	125 KSI TS	C26
	130 KSI TS	C28
	140 KSI TS	C31
	145 KSI TS	C32
	150 KSI TS	C34
	160 KSI TS	C36
	170 KSI TS	C38
	180 KSI TS	C39
	200 KSI TS	C43
	220 KSI TS	C46
D6AC	240 KSI TS	C49
4340, H11	260 KSI TS	C50.6
4340, H11, 300M	280 KSI TS	C52.7

NOTE

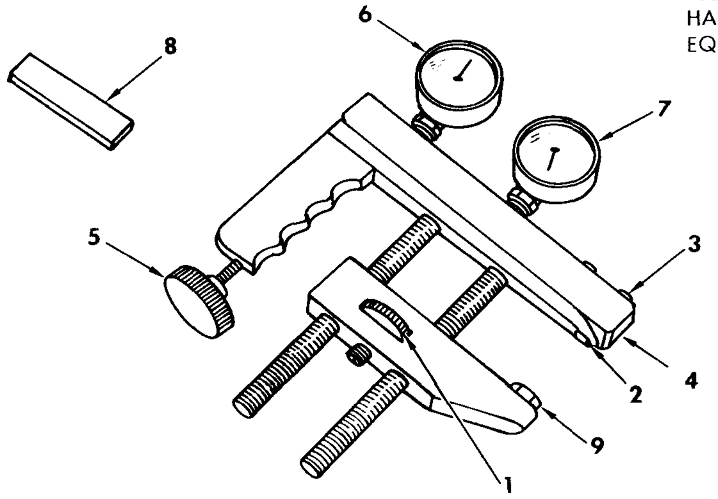
△1 Maximum Value

△2 May also be used as a guide for other 300 series stainless steels

Figure 59-4. Rockwell Hardness Scale (Sheet 2 of 2)

NOTE

SEE TEXT IN THIS SECTION UNDER PARAGRAPH TITLED HARDNESS TESTING FOR OPERATION AND USE OF EQUIPMENT ILLUSTRATED.



1. ADJUSTING KNOB
2. PENETRATOR
3. PENETRATOR CLAMP SCREW
4. WEIGHT BAR
5. LOADING SCREW
6. LOAD INDICATOR
7. PENETRATION INDICATOR
8. CALIBRATED TEST BLOCK
9. ANVIL

RIEHLE HARDNESS TESTER

ROCKWELL SCALE	PENETRATOR	MAJOR LOAD	PENETRATION INDICATOR	APPLICATION (TYPICAL)
A	DIAMOND CONE	60 KGS	BLACK	FOR EXTREMELY HARD MATERIALS SUCH AS SINTERED CARBIDES: FOR HARD STEEL WHICH IS TOO THIN FOR HEAVIER LOADS.
B	1/16 INCH BALL	100 KGS	RED	SOFT STEEL, CAST IRON, TITANIUM AND NON-FERROUS METALS.
C	DIAMOND CONE	150 KGS	BLACK	HARDENED STEEL; OTHER METALS ABOVE B100.
D	DIAMOND CONE	100 KGS	BLACK	THIN STEEL.
E	1/8 INCH BALL	60 KGS	RED	ALUMINUM AND MAGNESIUM
F	1/16 INCH BALL	60 KGS	RED	SOFT BEARING MATERIALS, ANNEALED COPPER; THIN SOFT SHEET
G	1/16 INCH BALL	150 KGS	RED	PHOSPHORBRONZE, BERYLLIUM COPPER; OTHER SOFT METAL BELOW G92.
H	1/8 INCH BALL	100 KGS	RED	SOFT ALUMINUM AND MAGNESIUM; VERY SOFT OR THIN METAL
K	1/8 INCH BALL	150 KGS	RED	

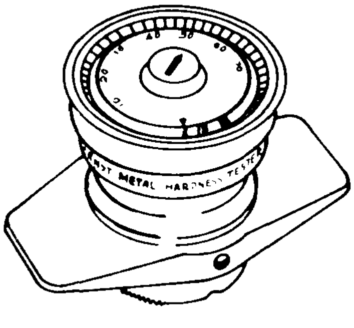
SMR151-59.1-0-0595

Figure 59-5. Portable Hardness Testers (Sheet 1 of 2)

MODEL RAR	ROCKWELL "A" RANGE 60 TO 75
MODEL RBR	ROCKWELL "B" RANGE 5 TO 100

NOTE

SEE TEXT IN THIS SECTION FOR OPERATION AND USE OF THIS TESTER.



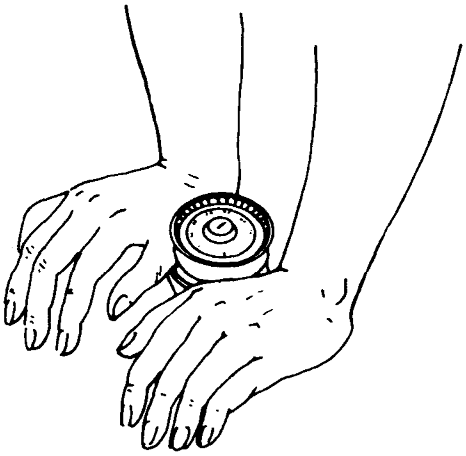
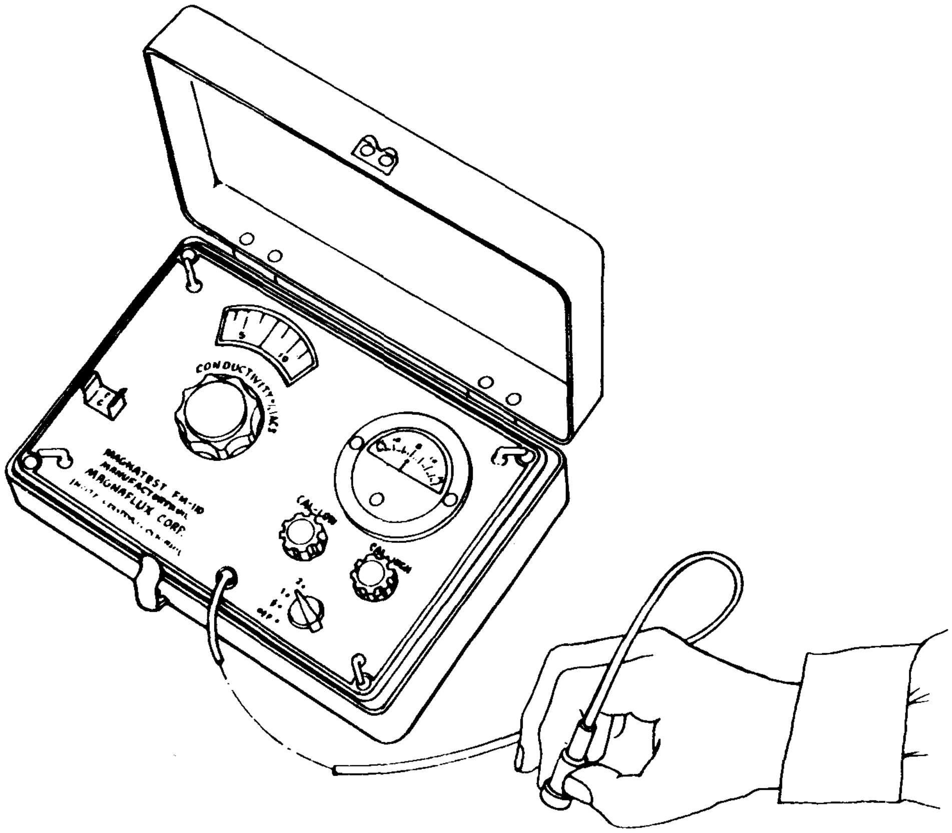
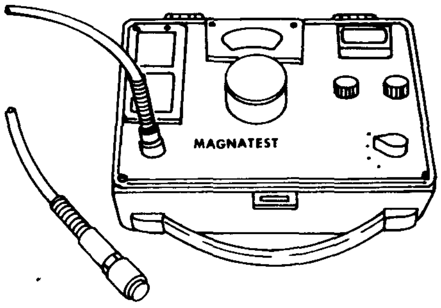


Figure 59-5. Portable Hardness Testers (Sheet 2 of 2)



FM-110 PORTABLE ELECTRICAL CONDUCTIVITY METER
(BATTERY POWERED)

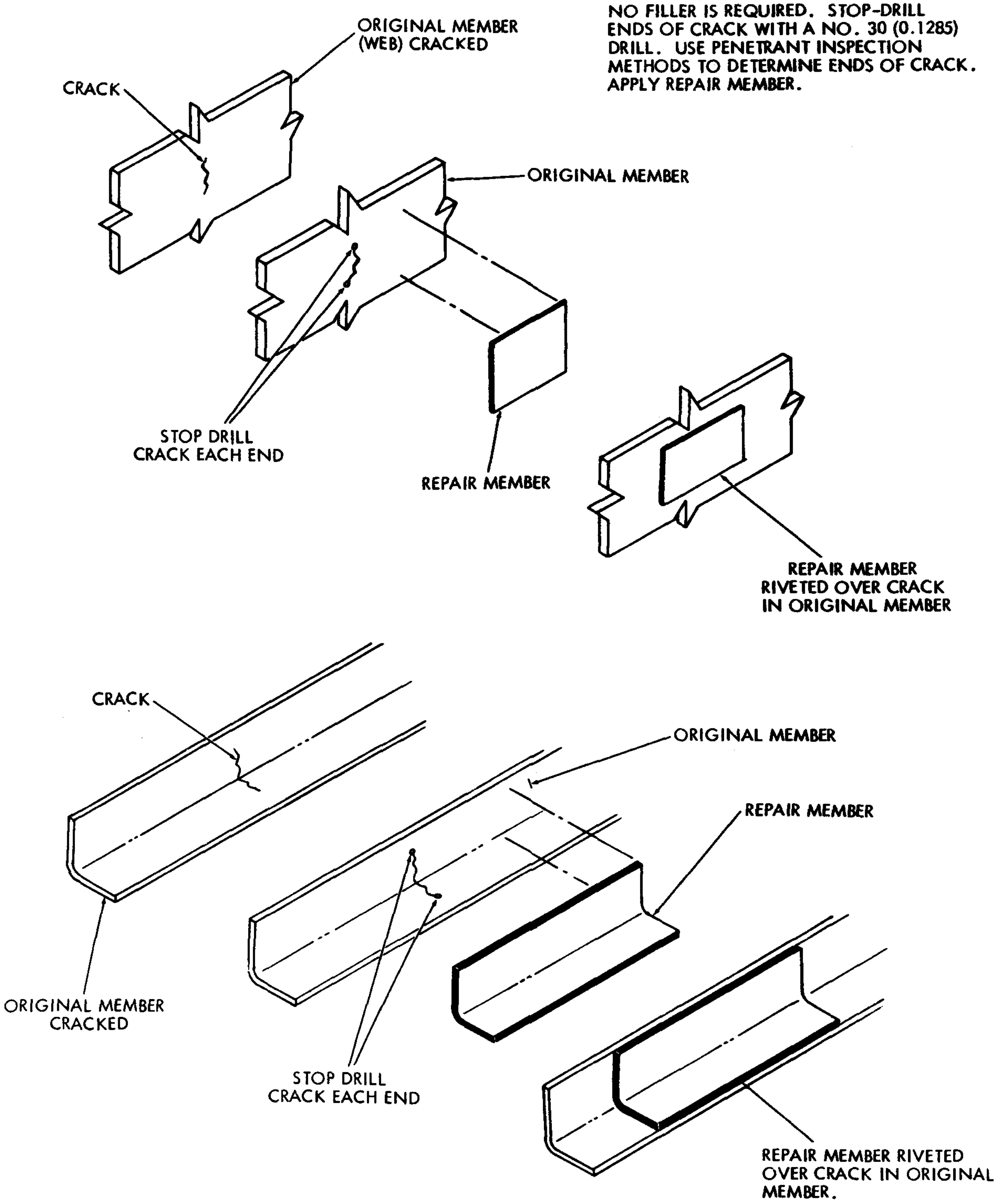
NOTE
SEE TEXT IN THIS SECTION UNDER PARAGRAPH TITLED HARDNESS TESTING FOR OPERATION AND USE OF EQUIPMENT ILLUSTRATED.



FM-120 CONDUCTIVITY MEASURING INSTRUMENT

SMR 151-59.1-0-0596

Figure 59-6. Magnetic Hardness Testers

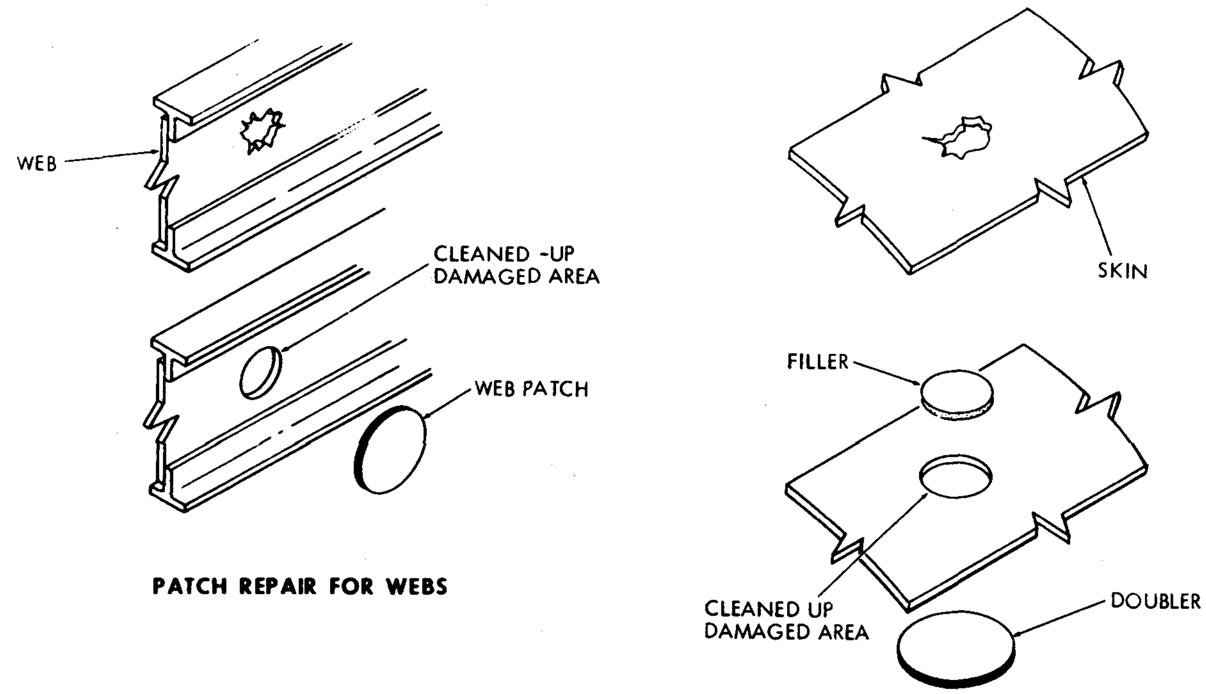

ORIGINAL MEMBER (WEB) CRACKED
CRACK
ORIGINAL MEMBER
STOP DRILL CRACK EACH END
REPAIR MEMBER
REPAIR MEMBER RIVETED OVER CRACK IN ORIGINAL MEMBER
CRACK
ORIGINAL MEMBER
REPAIR MEMBER
ORIGINAL MEMBER CRACKED
STOP DRILL CRACK EACH END
REPAIR MEMBER RIVETED OVER CRACK IN ORIGINAL MEMBER.

NOTE

CRACKS DO NOT HAVE TO BE CUT OUT. NO FILLER IS REQUIRED. STOP-DRILL ENDS OF CRACK WITH A NO. 30 (0.1285) DRILL. USE PENETRANT INSPECTION METHODS TO DETERMINE ENDS OF CRACK. APPLY REPAIR MEMBER.

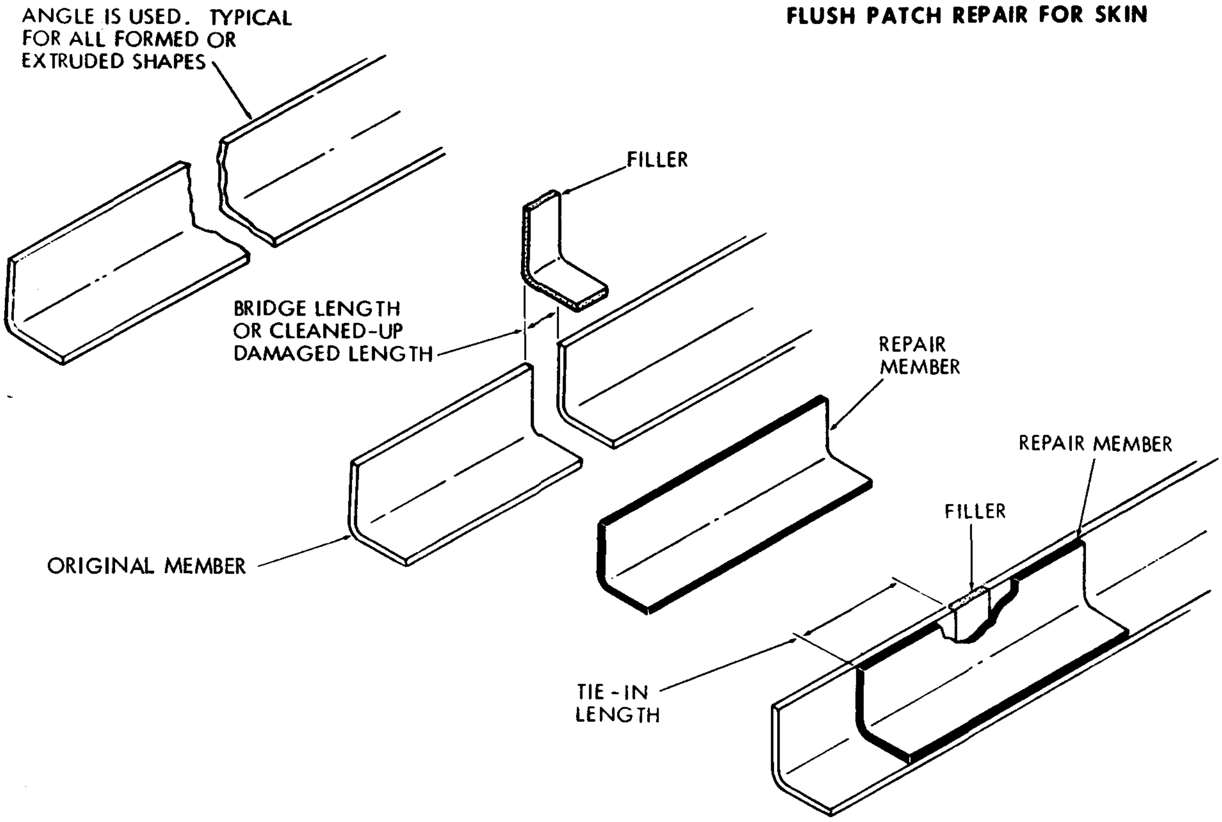
CRACK REPAIRS

Figure 59-7. Classification of Damage (Sheet 1 of 4)



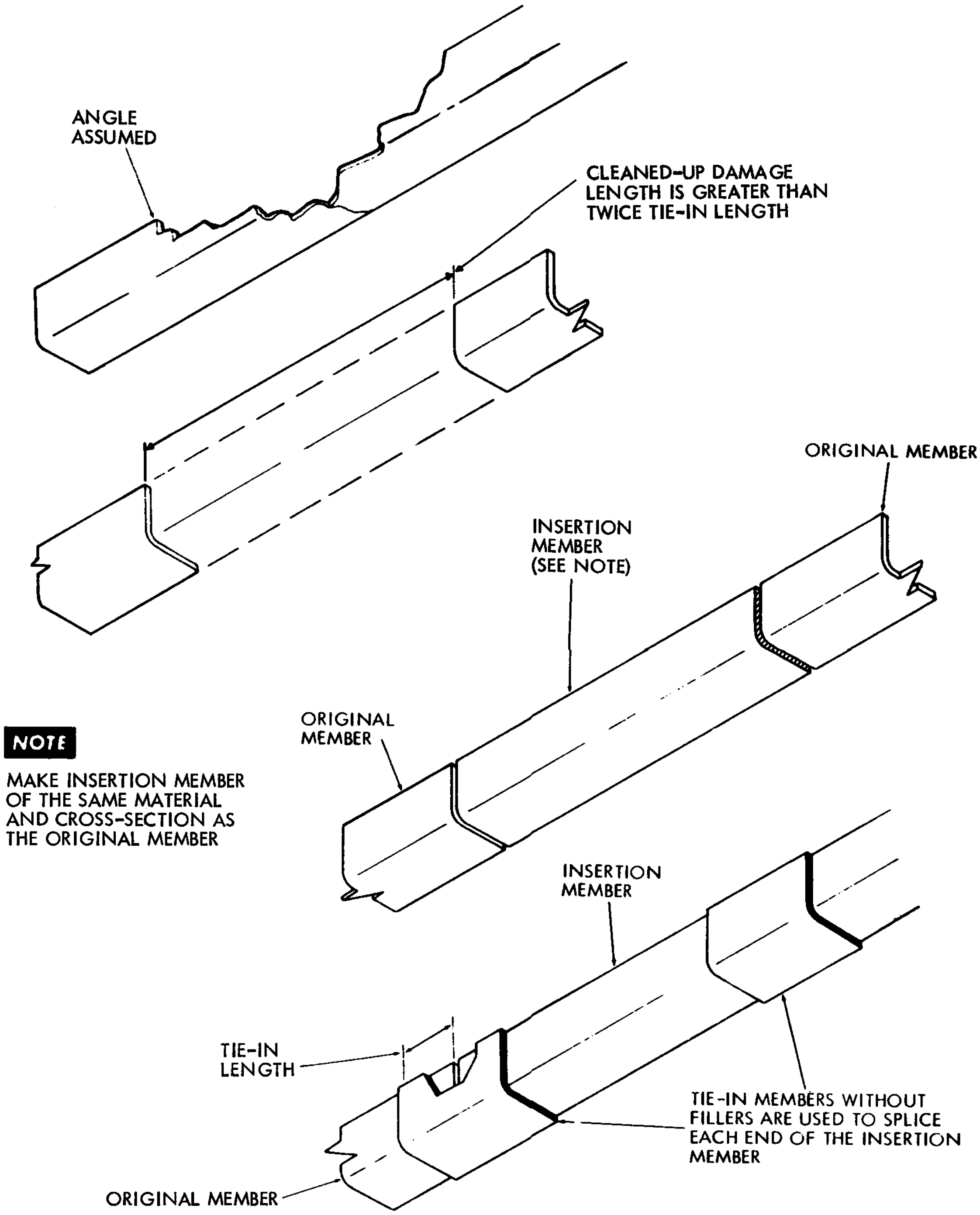
PATCH REPAIR FOR WEBS

FLUSH PATCH REPAIR FOR SKIN


ANGLE IS USED. TYPICAL FOR ALL FORMED OR EXTRUDED SHAPES
FILLER
BRIDGE LENGTH OR CLEANED-UP DAMAGED LENGTH
REPAIR MEMBER
REPAIR MEMBER
FILLER
ORIGINAL MEMBER
TIE - IN LENGTH

PATCH REPAIR FOR STRUCTURAL MEMBERS

Figure 59-7. Classification of Damage (Sheet 2 of 4)

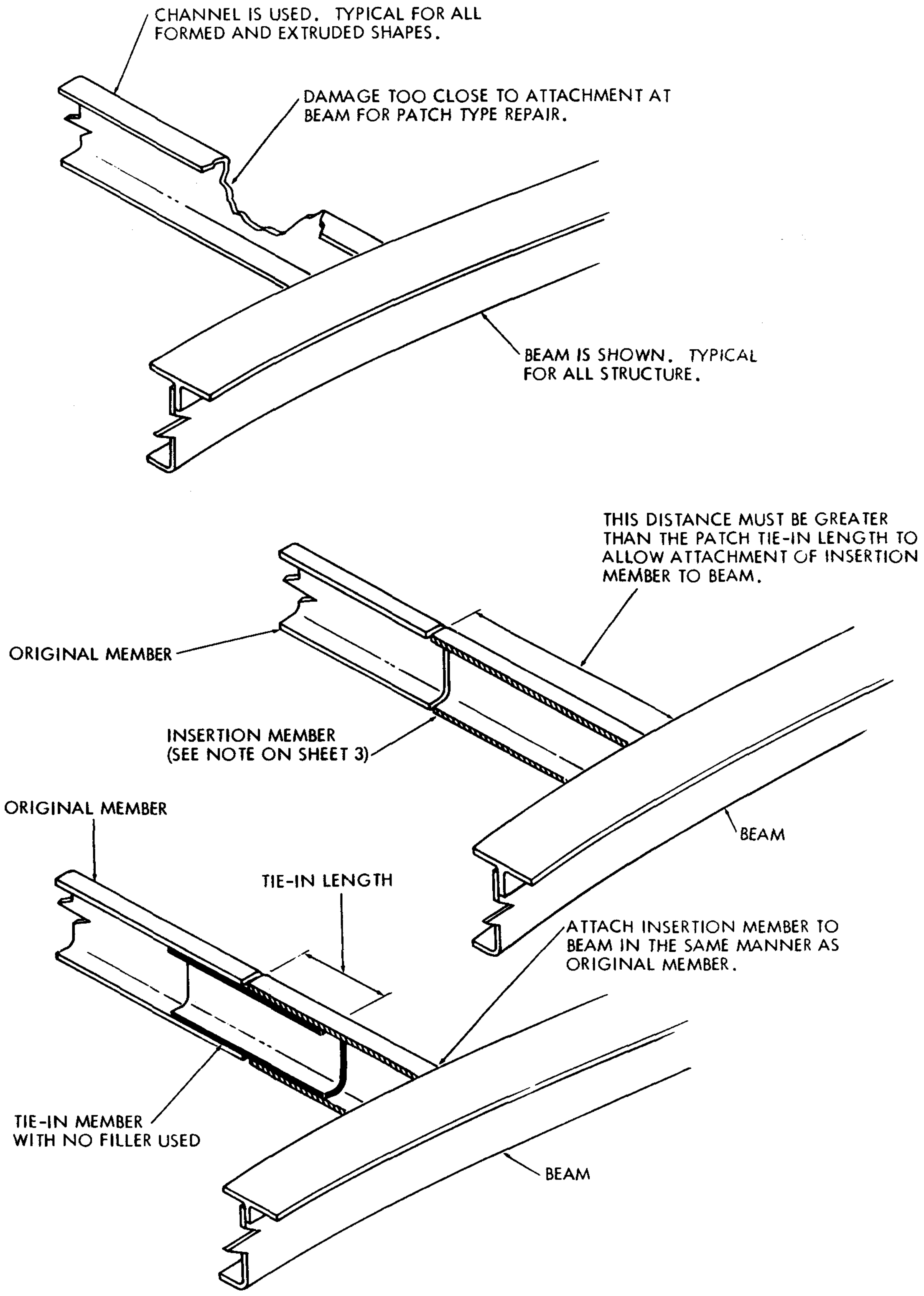

ANGLE ASSUMED
CLEANED-UP DAMAGE LENGTH IS GREATER THAN TWICE TIE-IN LENGTH
ORIGINAL MEMBER
INSERTION MEMBER (SEE NOTE)
ORIGINAL MEMBER
INSERTION MEMBER
TIE-IN LENGTH
TIE-IN MEMBERS WITHOUT FILLERS ARE USED TO SPLICE EACH END OF THE INSERTION MEMBER
ORIGINAL MEMBER

NOTE

MAKE INSERTION MEMBER OF THE SAME MATERIAL AND CROSS-SECTION AS THE ORIGINAL MEMBER

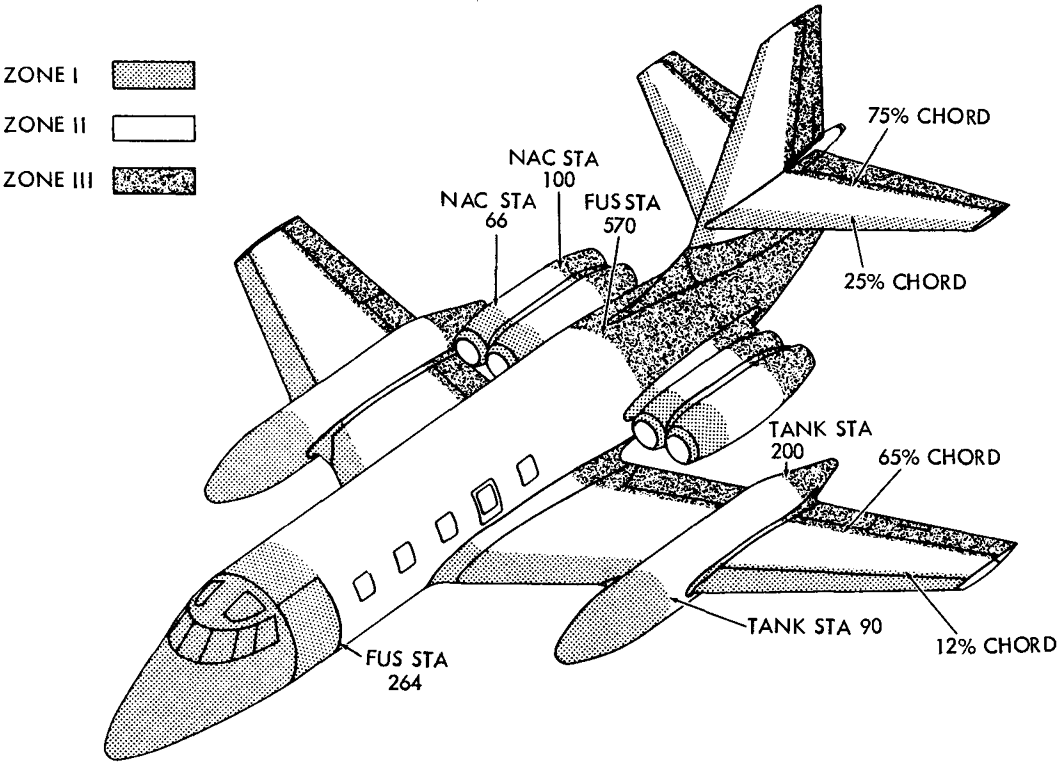
INSERTION REPAIR FOR STRUCTURAL MEMBERS

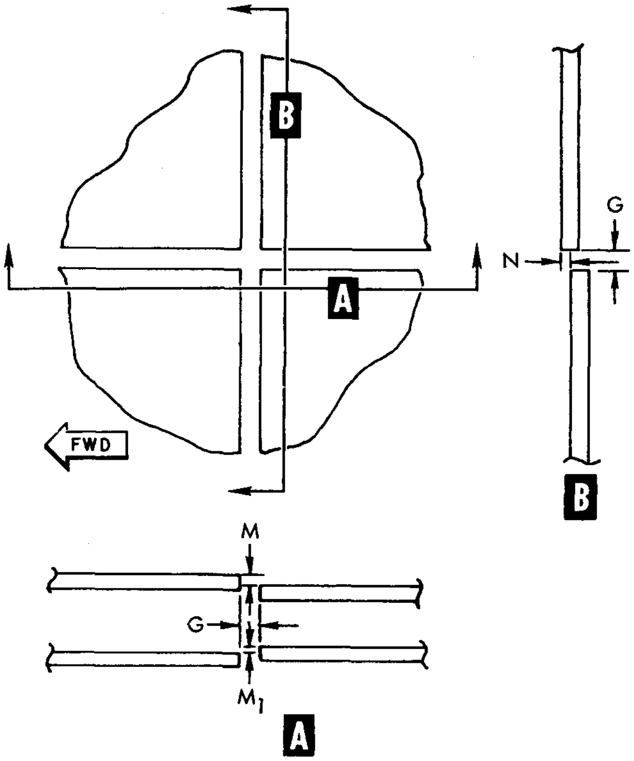
Figure 59-7. Classification of Damage (Sheet 3 of 4)


CHANNEL IS USED. TYPICAL FOR ALL FORMED AND EXTRUDED SHAPES.
DAMAGE TOO CLOSE TO ATTACHMENT AT BEAM FOR PATCH TYPE REPAIR.
BEAM IS SHOWN. TYPICAL FOR ALL STRUCTURE.
THIS DISTANCE MUST BE GREATER THAN THE PATCH TIE-IN LENGTH TO ALLOW ATTACHMENT OF INSERTION MEMBER TO BEAM.
ORIGINAL MEMBER
INSERTION MEMBER (SEE NOTE ON SHEET 3)
BEAM
ORIGINAL MEMBER
TIE-IN LENGTH
ATTACH INSERTION MEMBER TO BEAM IN THE SAME MANNER AS ORIGINAL MEMBER.
TIE-IN MEMBER WITH NO FILLER USED
BEAM

INSERTION REPAIR NEAR ATTACHMENT END FOR STRUCTURAL MEMBERS

Figure 59-7. Classification of Damage (Sheet 4 of 4)





TYPICAL FIXED SKIN BUTT JOINTS

NOTE

△1 FOR 20% OF ANY GIVEN LENGTH, THE TOLERANCE MAY BE DOUBLED.

△2 FOR 20% OF ANY 12-INCH LENGTH, THE TOLERANCE MAY BE DOUBLED.

△3 FOR REMOVABLE PANELS, THE TOLERANCE IS FROM 0.020 TO 0.080 INCH.

△4 THE TOLERANCE FOR GAPS AND NEGATIVE STEPS MAY BE DOUBLED FOR 10% OF THE JOINT LENGTH, AND THE TOLERANCE FOR POSITIVE STEPS MAY BE DOUBLED FOR 5% OF THE JOINT LENGTH.

5. FLUSH ATTACHMENTS, (RIVETS, SCREWS, LOCKBOLTS, ETC) SHOULD BE FLUSH WITHIN THE FOLLOWING TOLERANCES ON THE FUSELAGE, WINGS, EXTERNAL TANKS, AND EMPENNAGE:
   ZONE I: +0.004 TO -0.002 INCH
   ZONES II & III: +0.008 TO -0.004 INCH
6. FLUSH ATTACHMENTS ON THE NACELLES SHOULD BE FLUSH WITHIN THE FOLLOWING LIMITS.
   RIVETS: +0.006 TO -0.002 INCH
   SCREWS AND LOCKBOLTS: +0.007 TO -0.004 INCH

Figure 59-8. Aerodynamic Smoothness Requirements (Sheet 1 of 2)

AREA	SYMBOL	TOLERANCE(INCHES)
FUSELAGE ZONE I	G	△1 0.040
	M	△2 0.015
	M_1	0.015
	N	0.015
FUSELAGE ZONE II	G	△1 0.040
	M	△2 0.020
	M_1	0.020
	N	0.020
FUSELAGE ZONE III	G	△1 0.040
	M	△2 0.020
	M_1	0.020
	N	0.020
WING ZONE I & II	G	△4 △3 0.060
	M	0.020
	M_1	0.020
	N	0.020
WING ZONE III	G	△4 0.060
	M	0.030
	M_1	0.030
	N	0.030
EMPENNAGE ZONE I	G	△4 0.060
	M	△1 0.015
	M_1	0.015
	N	0.030
EMPENNAGE ZONE II & III	G	△4 0.060
	M	△1 0.020
	M_1	0.020
	N	0.040
NACELLE ZONES I, II, & III	G	0.040 INCH FOR 60% OF LINEAR LENGTH 0.060 INCH FOR 30% OF LINEAR LENGTH 0.090 INCH FOR 10% OF LINEAR LENGTH
	M M_1 N	0.010 INCH FOR 95% OF LINEAR LENGTH 0.030 INCH FOR 5% OF LINEAR LENGTH

Figure 59-8. Aerodynamic Smoothness Requirements (Sheet 2 of 2)

	SHEET THICKNESS														
MATERIAL	.012	.016	.020	.025	.032	.040	.050	.063	.071	.080	.090	.100	.125	.160	.190
3003-0	.06	.06	.06	.06	.06	.06	.06	.06	.09	.09	.09	.12	.12	.16	.19
5052-0 6061-0, -W	.06	.06	.06	.06	.06	.06	.06	.09	.09	.09	.12	.12	.16	.19	.22
2024-0, -W 6061-T4	.06	.06	.06	.06	.06	.06	.09	.09	.12	.12	.16	.19	.22	.31	.38
3003-H14	.06	.06	.06	.06	.06	.09	.09	.12	.12	.16	.19	.22	.31	.38	.44
2024-T3	.06	.06	.06	.09	.09	.12	.16	.22	.25	.31	.38	.44	.62	.75	1.00
2024-T36	.06	.09	.09	.09	.12	.16	.19	.25	.31	.38	.44	.50	.75	1.00	1.25
7075-0, -W 7079-0, -W	.06	.06	.06	.06	.09	.09	.12	.16	.19	.22	.25	.31	.38	.50	.62
7075-T6* 7079-T6*	.06	.09	.12	.12	.16	.22	.25	.31	.41	.44	.50	.69	.87	1.00	1.25
7173-0, -W	.06	.06	.06	.06	.09	.09	.12	.19	.22	.25	.31	.38	.50	.75	
7173-T6*	.06	.09	.16	.19	.22	.31	.38	.50	.56	.62	.62	.75	1.00	1.25	

MATERIAL	.012	.016	.020	.025	.032	.040	.045	.050	.063	.080	.090	.112	.125	.160	.190
4130 COND N	.06	.06	.06	.06	.06	.09	.09	.12	.12	.16	.19	.22	.25	.31	.38
ANNEALED CRES	.06	.06	.06	.06	.06	.06	.09	.09	.09	.12	.12	.16	.19	.22	.25
1/4 HARD CRES	.06	.06	.06	.06	.06	.09	.09	.12	.12	.16	.19	.22	.25	.31	.38
1/2 HARD CRES	.06	.06	.06	.09	.09	.12	.12	.12	.06	.25	.25	.31	.38	.50	.62
FULL HARD CRES	.06	.06	.09	.12	.12	.16	.19	.22	.25	.31	.38	.44	.50	.62	.87

* COLD FORMING BARE SHEET IN-T6 TEMPER NOT ALLOWED. MUST BE HOT FORMED AT 300°F ± 25°F.

Figure 59-9. Minimum Bend Radii for Repair Parts

ALUMINUM ALLOYS

MATERIAL THICKNESS T (INCHES)	TRANSITION LENGTH-"A"	PREFERRED		ACCEPTABLE	
		2024-T3, -T81 (SHEET) 2024-T4, -T81 (EXTRUSION) 7075-T6 (HOT)		2024-T42, -T62 (SHEET) 2024-T42, -T62 (EXTRUSION) 7075-T6, 6061-T4	
		"D" MAX	"L" MIN	"D" MAX	"L" MIN
THRU .025	6D PREFERRED 4D MINIMUM	.12	.62	.19	.50
.026 THRU .040		.16	.68	.22	.56
.041 THRU .063		.19	.75	.25	.62
.064 THRU .090		.20	1.00	.25	.88
.091 THRU .125		.25	1.25	.31	1.00

STEEL AND TITANIUM

MATERIAL THICKNESS T (INCHES)	TRANSITION LENGTH-"A"	PERMISSIBLE LIMITS			
		1/4 H CRES LAC 1-580 & 1-582		17-7 COND A 347-1A LOW CARBON STEEL CRES ANNEALED	
		"D" MAX	"L" MIN	"D" MAX	"L" MIN
THRU .025	6D PREFERRED 4D MINIMUM	.09	.62	.19	.50
.026 THRU .040		.12	.75	.22	.56
.041 THRU .063		.16	.88	.25	.62
.064 THRU .090		.19	1.12	.25	.88
.091 THRU .125		.20	1.50	.31	1.00

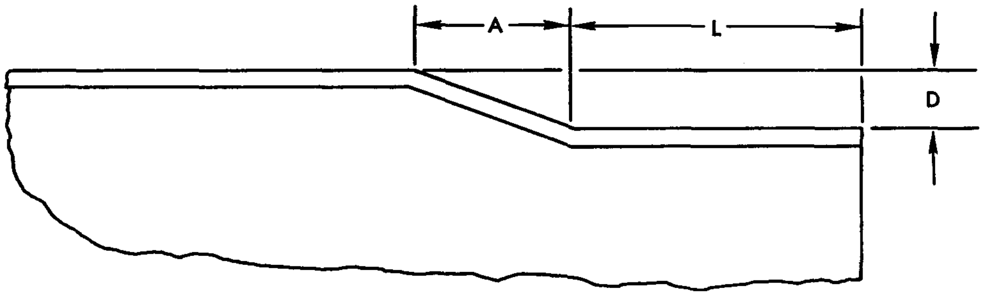


Figure 59-10. Standard Joggles for Repair Parts

RIVET HOLE SIZES

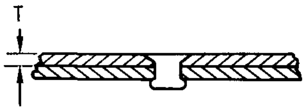
RIVET TYPE	RIVET DIA	FINISH DRILL	HOLE LIMITS (DIA)		PERMISSIBLE HOLE SIZE FOR REWORK ONLY
			MINIMUM	MAXIMUM	
MS20470 MS20426 AND NAS1097	1/16	NO. 52	0.062	0.068	.064 - .071
	3/32	NO. 40	0.097	0.102	.097 - .106
	1/8	NO. 30	0.127	0.135	.128 - .140
	5/32	NO. 21	0.158	0.168	.158 - .174
	3/16	NO. 11	0.189	0.202	.189 - .209
	1/4	6.4MM	0.252	0.266	.252 - .276
	5/16	NO. 0	0.315	0.322	

BOLT AND SCREW HOLES AND EDGE DISTANCES

BOLT AND SCREW SIZE	HOLE SIZE	DRILL REFERENCE	MINIMUM EDGE DISTANCE
6-32	.143-.149	#27	.28
8-32	.168-.174	#18	.31
10-32	.192-.198	#10	.38
1/4-28	.251-.258	6.4MM	.50
5/16-24	.315-.322	#0	.62

MINIMUM EDGE DISTANCE

RIVET DIAMETER	NON FLUSH	FLUSH
		CSK
1/16	.12	.12
3/32	.19	.19
1/8	.25	.25
5/32	.31	.31
3/16	.38	.38
1/4	.50	.50
5/16	.62	.62



COUNTERSINK

NOMINAL DIAMETER	TYPE FLUSH FASTENER	MINIMUM SURFACE SHEET THICKNESS(T) FOR COUNTERSINKING
3/32	MS20426	0.036 *
1/8	NAS1097 MS20426	0.032 0.050
5/32	NAS1097 HL19 MS20426, NAS7025H	0.040 0.050 0.063
6-32	SCREWS	0.063
8-32	SCREWS	0.080
3/16	NAS1097 HL19, NAS1436H MS20426, NAS7026H	0.050 0.063 0.071
10-32	SCREWS	0.090
1/4	HL19, NAS1438H NAS7028H MS20426 SCREWS	0.071 0.090 0.100 0.125
5/16	HL19, NAS1440H MS20426, NAS7030H SCREWS	0.080 0.125 0.160
3/8	HL19, NAS1442H NAS7032H MS20426 SCREWS	0.090 0.125 0.160 0.190

* 0.025 INCH SHEET MAY BE COUNTERSUNK WHEN REQUIRED FOR PLATE NUTS, GANG CHANNEL NUTS, OR SIMILAR TYPE NUTS. BONDED OR SPOTWELDED SHEETS MAY BE USED TO MAKE UP THIS THICKNESS.

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Figure 59-11. Attachment Holes, Edge Distances, and Countersinks

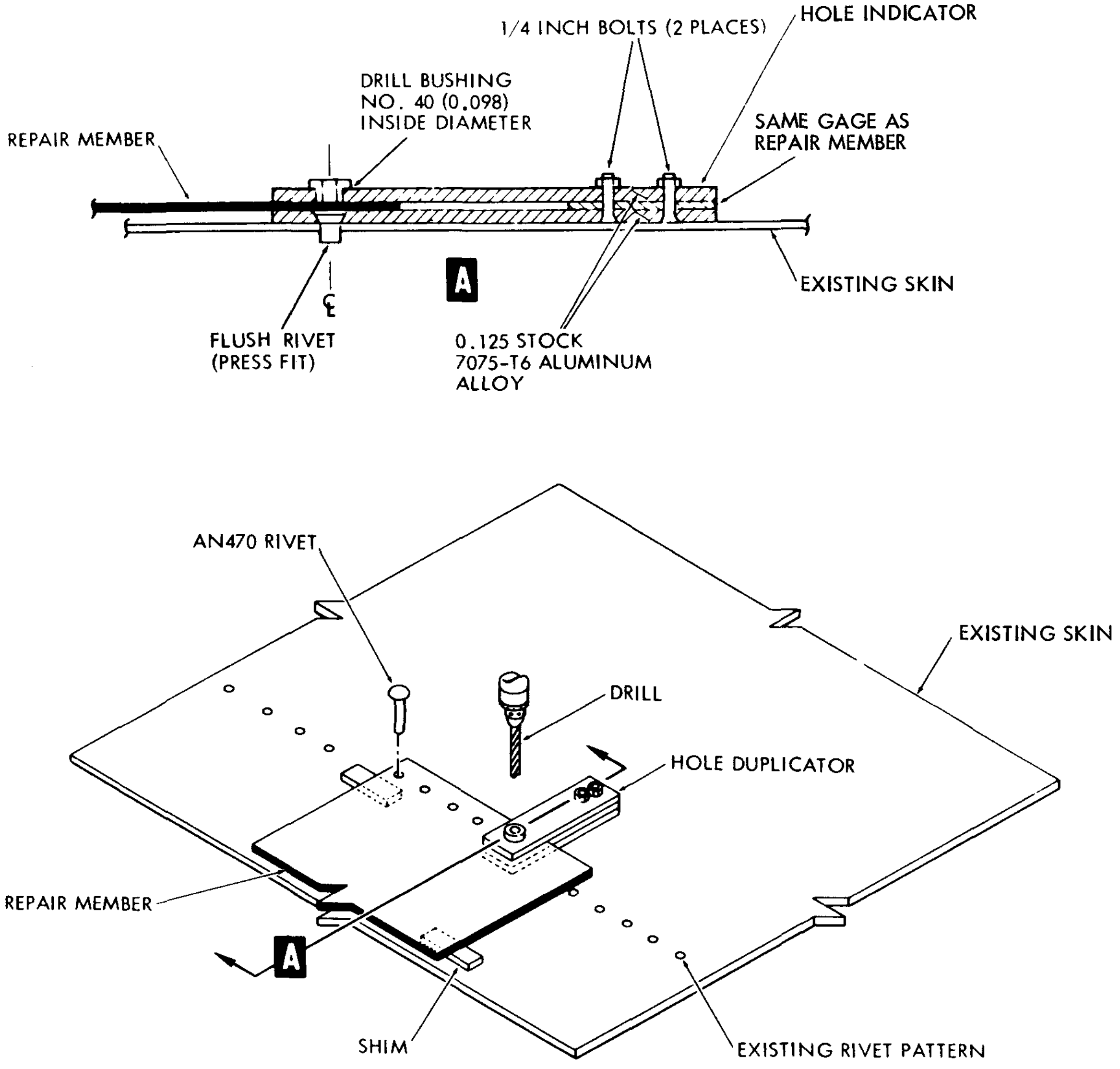
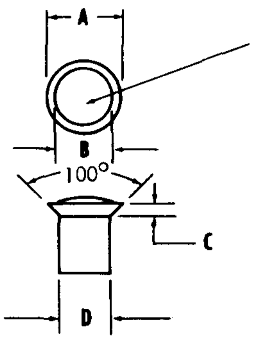


Figure 59-12. Locating Blind Holes

RIVET TYPE	RIVET DIA	DIMENSIONS		
		A	B	C
NAS 1097 SHEAR HEAD	1/8	0.192	0.139	0.029
	5/32	0.243	0.190	0.037
	3/16	0.299	0.203	0.046
MS20426 STANDARD	1/8	0.255	——	0.042
	5/32	0.266	——	0.055
	3/16	0.353	——	0.070

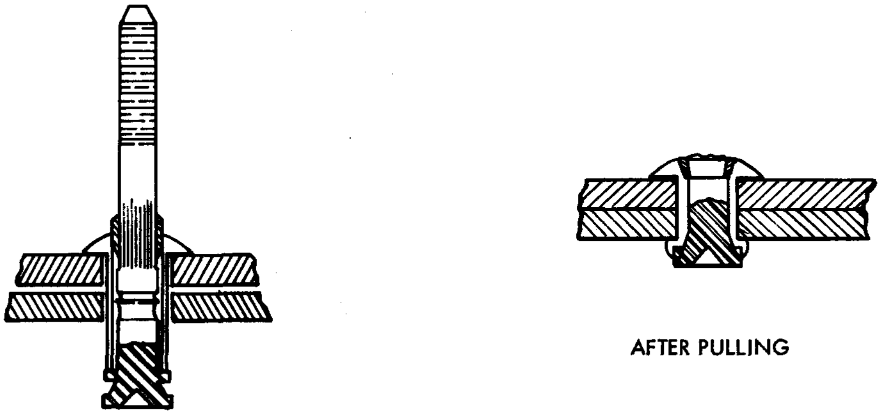


NOTE

ALLOY IDENTIFICATION IS THE SAME FOR THE SHEAR HEAD (NAS1097) RIVET AS FOR THE STANDARD (MS20426) RIVET.

Figure 59-13. Comparison Between NAS1097 Shear Head and MS20426 Standard Flush Rivets

BLIND RIVET HOLE DIAMETER REQUIREMENTS					
BLIND RIVET	RIVET DIA.	PRE-DRILL WITH	SIZE-DRILL WITH	HOLE LIMITS	
				MIN.	MAX.
STANDARD (NAS1398, NAS1399)	1/8 5/32 3/16	NO. 38 (0.101) NO. 30 (0.129) NO. 20 (0.161)	NO. 30 (0.129) NO. 20 (0.161) NO. 10 (0.193)	0.129 0.160 0.192	0.132 0.164 0.196
1/64 OVERSIZE (NAS1738, NAS1739)	1/8 5/32 3/16	NO. 32 (0.116) NO. 26 (0.147) NO. 16 (0.177)	NO. 27 (0.144) NO. 16 (0.177) NO. 5 (0.205)	0.143 0.176 0.205	0.146 0.180 0.209



BEFORE PULLING

AFTER PULLING

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Figure 59-14. Blind Rivet Installations and Inspections (Sheet 1 of 3)

BLIND RIVET DATA

SUBSTITUTION DATA						
EXISTING FASTENER (BLIND RIVET) △1					SUBSTITUTE FASTENER	
SIZE	HEAD TYPE	NAS NO.	CHEERRY NO. VOI-SHAN NO.	ALLFAST NO.	△1 CHERRY MAX FASTENER NO.	△1 NAS OVERSIZE FASTENER NO.
STANDARD	PROT	NAS1398B	CR2263	AF2050	CR3213	NAS1738E
	PROT	NAS1398C	CR2663	AF2040	CR3213	NAS1738E
	PROT	NAS1398D	CR2163	AF2000	CR3213	NAS1738E
	PROT	NAS1398M	CR2563M	AF2090	CR3523	NAS1738M
	PROT	NAS1398MS	CR2563S	AF2090S	CR3253P	NAS1738MW
	PROT	NAS1398MW	CR2563	AF2090W	CR3253P	NAS1738MW
	FLUSH	NAS1399B	CR2262	AF2051	CR3212	NAS1739E
	FLUSH	NAS1399C	CR2662	AF2041	CR3212	NAS1739E
	FLUSH	NAS1399D	CR2162	AF2001	CR3212	NAS1739E
	FLUSH	NAS1399M	CR2562M	AF2091	CR3522	NAS1739M
	FLUSH	NAS1399MS	CR2562S	AF2091S	CR3522P	NAS1739MW
	FLUSH	NAS1399MW	CR2562	AF2091W	CR3552P	NAS1739MW
1/64-INCH OVERSIZE	PROT	NAS1738B	CR2249	AF2250	CR3243	———
	PROT	NAS1738E	CR2239	AF2550	CR3243	———
	PROT	NAS1738M	CR2539	AF2290	CR3553	———
	PROT	NAS1738MW	CR2539P	AF2290W	CR3553P	———
	FLUSH	NAS1739B	CR2248	AF2251	CR3242	———
	FLUSH	NAS1739E	CR2238	AF2251	CR3242	———
	FLUSH	NAS1739M	CR2538	AF2291	CR3552	———
	FLUSH	NAS1739MW	CR2538P	AF2291W	CR3552P	———

△1 ADD DIAMETER AND GRIP TO OBTAIN COMPLETE PART NUMBER (NAS1738MW ( ) - ( )).
INSTALLATION TOOL MUST BE CORRECT FOR FASTENER USED.

JS-59-0-014-2

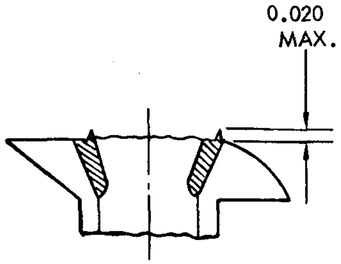
Figure 59-14. Blind Rivet Installations and Inspections (Sheet 2 of 3)

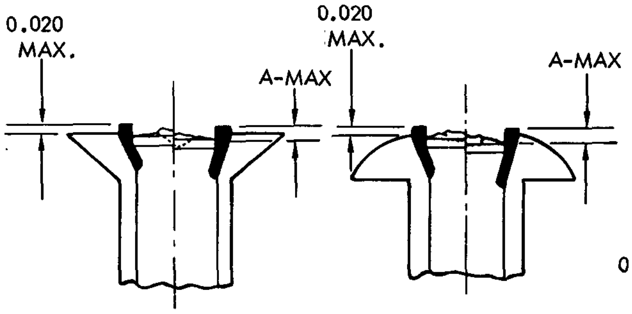
PROTRUDING AND FLUSH HEAD TYPE		
1/8,5/32 & 3/16 DIAMETER		
PART DASH NO.	GRIP RANGE	
	MIN	MAX
1	0.020	0.062
2	0.063	0.125
3	0.126	0.187
4	0.188	0.250
5	0.251	0.312
6	0.313	0.375
7	0.376	0.437
8	0.438	0.500

INSPECTION

IF THE RIVET STEM AND COLLAR ARE FLUSH WITHIN THE LIMITS DESCRIBED IT CAN BE SAFELY CONCLUDED THAT A SATISFACTORY BLIND HEAD AND LOCK HAS BEEN FORMED.

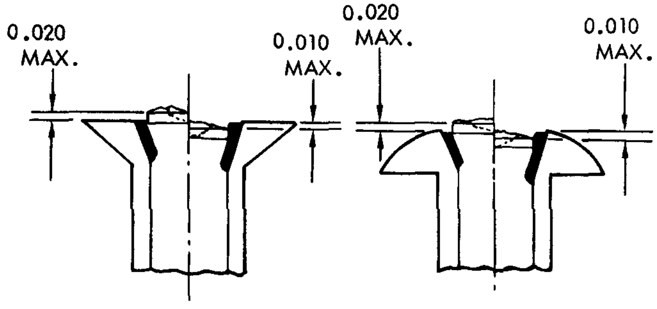
A SLIGHT COLLAR "FLASH" CAUSED BY THE PRESSURES NECESSARY TO DRIVE THE COLLAR IS ACCEPTABLE WITHIN THE LIMIT SHOWN.





COLLAR FLUSHNESS LIMITS

RIVET SIZE	-4 DIA	-5 DIA	-6 DIA
A MAX	0.015	0.020	0.025



STEM FLUSHNESS LIMITS

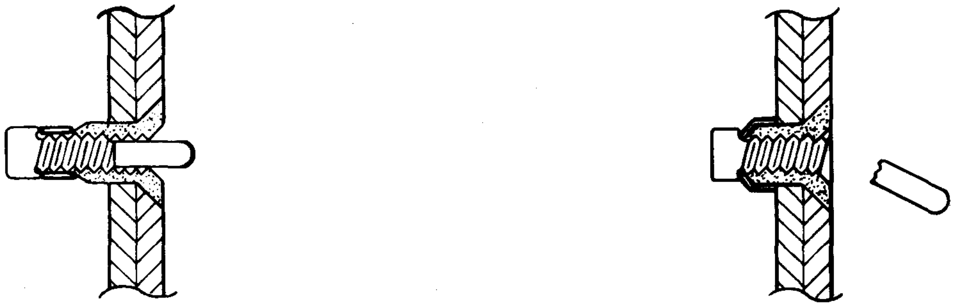
JS-1-2-X0/2-5914-3

Figure 59-14. Blind Rivet Installations and Inspections (Sheet 3 of 3)

DRILLED HOLE SIZES				FLUSH HEAD BLIND BOLTS
BLIND BOLT	NOMINAL DIAMETER	PRE-DRILL	FINAL HOLE SIZE △1	COUNTERSUNK DIAMETER (100°)
NAS1669-08L (PROTRUDING HEAD) NAS1670-08L (FLUSH HEAD)	#8	#25	0.168/0.165	0.332/0.325
NAS1669-3L (PROTRUDING HEAD) NAS1670-3L (FLUSH HEAD)	#10	#15	0.202/0.199	0.385/0.378
NAS1669-4L (PROTRUDING HEAD) NAS1670-4L (FLUSH HEAD)	1/4	D	0.263/0.260	0.507/0.499

NOTE

△1 OBTAIN FINAL HOLES SIZES BY DRILLING OR REAM ON ASSEMBLY ONLY.



BLIND BOLT INSERTED IN HOLE

BLIND BOLT COMPLETELY INSTALLED
STEM WRENCHED OFF AT GROOVE

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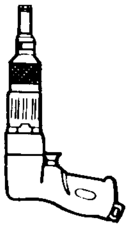
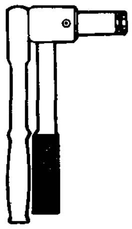
Figure 59-15. Blind Bolt Installation, Inspection, and Removal (Sheet 1 of 4)

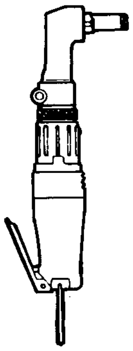
INSTALLATION TOOLS



MODEL MP550
BLIND BOLT
PNEUMATIC PISTOL



MODEL MH75
HAND TOOL
RATCHET WRENCH



MODEL MR550
RIGHT ANGLE DRIVER

NOTE

2. OVERSIZE BLIND BOLTS IN SAME NOMINAL DIAMETERS TAKE SAME ADAPTER PARTS AS INDICATED BELOW FOR REGULAR NOMINAL SIZES.
3. BLIND BOLTS, DRIVERS, AND ALL REQUIRED ADAPTER PARTS OR ASSEMBLIES, ARE AVAILABLE FROM MONOGRAM/AEROSPACE FASTENERS, A DIVISION OF MONOGRAM IND., LOS ANGELES, CALIF.
4. ADAPTER ASSEMBLY NO. IS BASIC NOSE ADAPTER NUMBER PLUS LETTER A, i.e., MPP5A IS THE ASSEMBLY OF MPP-5 AND MP-5 (TYP).

SIZE	MP550 BLIND BOLT PNEUMATIC PISTOL		MR550 RIGHT ANGLE DRIVER		MH75 HAND TOOL	
	NOSE ADAPTER HOLDS NUT	WRENCH ADP TURNS SCREW			NOSE ADAPTER HOLDS NUT	WRENCH ADP TURNS SCREW
08	MPP-5 MPF-5	MP-5 MP-5	MRTP-5 MRTF-5	MRT-5 MRT-5	MHP-5 MHF-5	MH-5 MH-5
3	MPP-6 MPF-6	MP-6 MP-6	MRTP-6 MRTF-6	MRT-6 MRT-6	MHP-6 MHF-6	MH-6 MH-6
4	MPP-8 MPF-8	MP-8 MP-8	MRTP-8 MRTF-8	MRT-8 MRT-8	MHP-8 MHF-8	MH-8 MH-8

JS-I -2-X0/2-5915-2

Figure 59-15. Blind Bolt Installation, Inspection, and Removal (Sheet 2 of 4)

INSPECTION

TABLE OF BREAK-OFF LIMITS

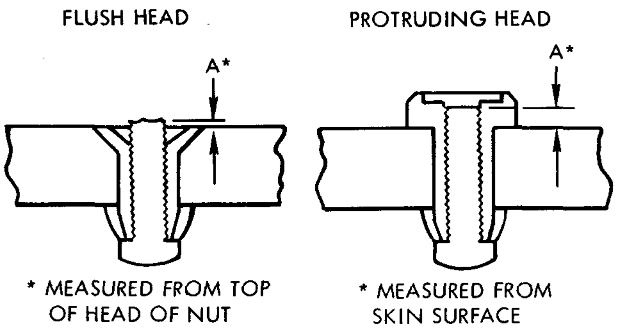
BLIND BOLT SIZE	A * FLUSH	A * PROT
	△5 OR △6	△5
-08	+ .020 - .068	+ .000 + .088
-3	+ .015 - .073	+ .010 + .098
-4	+ .010 - .078	+ .047 + .135

NOTE

△5 REFERS TO BREAK-OFF AS MEASURED FROM SKIN SURFACE.

△6 REFERS TO BREAK-OFF AS MEASURED FROM EXTREME TOP SURFACE OF THE HEAD OF THE NUT.

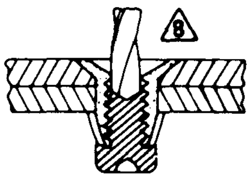
7. MONOGRAM LEAF GAUGES MGP5, 6, 8 (PROT HD) AND MFG5, 6 AND 8 (FLUSH HD) ARE AVAILABLE TO CHECK DIMENSION "A".


FLUSH HEAD
PROTRUDING HEAD
A*
A*
* MEASURED FROM TOP OF HEAD OF NUT
* MEASURED FROM SKIN SURFACE

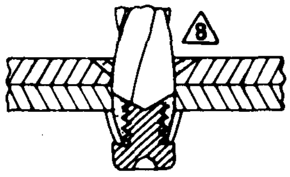
BLIND BOLT SIZE	TORQUE IN IN. POUNDS
-08	3 TO 4
-3	5 TO 6
-4	7 TO 10

JS-1-2-X0/2-5915-3

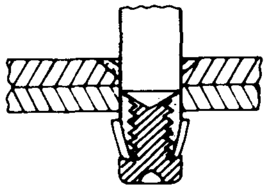
Figure 59-15. Blind Bolt Installation, Inspection, and Removal (Sheet 3 of 4)



SELECT DRILL FROM COLUMN 1. DRILL TO BELOW HEAD-SHANK JUNCTURE.

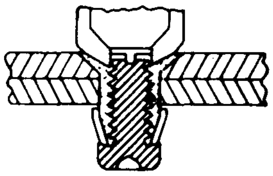


SELECT DRILL FROM COLUMN 2. DRILL TO DEPTH OF PILOT HOLE.

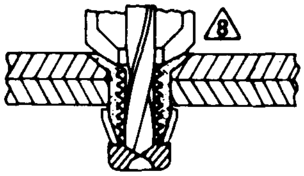


WITH HAMMER AND NOMINAL SIZE PUNCH, SEVER HEAD AND DRIVE OUT SHANK AND BLIND HEAD.

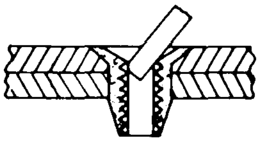
CLAMPED UP BLIND BOLT



PREVENT NUT FROM TURNING BY ENGAGING DRIVING TOOL NOSE ADAPTER. HOLD NOSE ADAPTER WITH HAND TOOL HANDLE OR VICE GRIP PLIERS.



SELECT DRILL FROM COLUMN 1. DRILL THRU SHANK, SEVERING BOLT HEAD.



PICK NUT OUT OF HOLE WITH PUNCH.

BOLT TOO LONG - NOT CLAMPED UP

NOTE

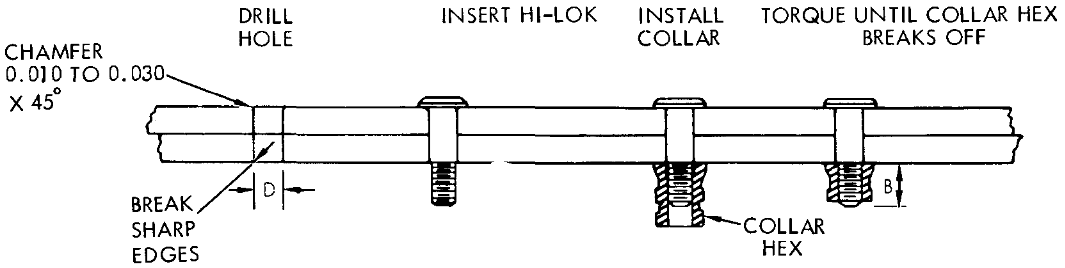
⚠8 USE A DRILL WITH A SPEED OF NOT OVER 500 REVOLUTIONS PER MINUTE.

REMOVAL DRILLS		
BLIND BOLT SIZE	**DRILL SIZE**	
	COLUMN 1	**COLUMN 2**
-08 SERIES (ALL HEAD STYLES)	#42 (.0935)	#23 (.154)
-3 SERIES (ALL HEAD STYLES)	#35 (0.110)	#12 (0.189)
-4 SERIES (ALL HEAD STYLES)	#24 (0.152)	D (0.246)

JS-1-2-X0/2-5915-4

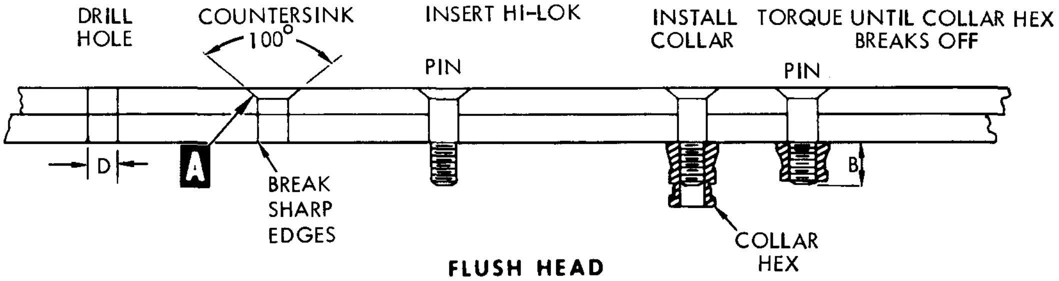
Figure 59-15. Blind Bolt Installation, Inspection, and Removal (Sheet 4 of 4)

INSTALLATION
(COORDINATE WITH TEXT IN THIS SECTION)

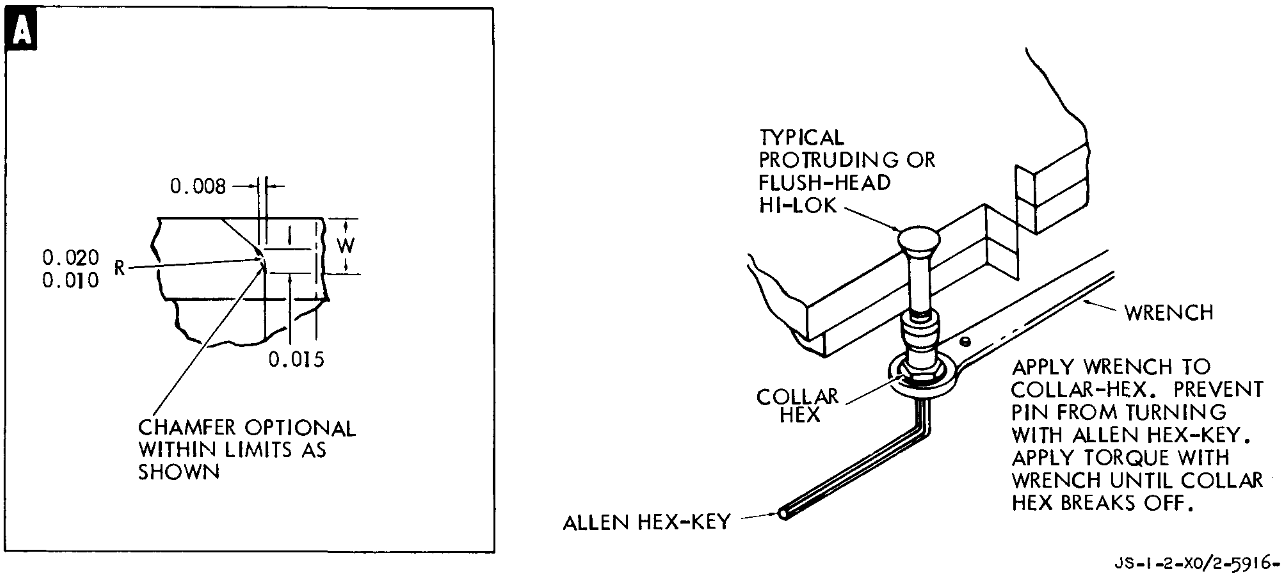


PROTRUDING HEAD

FOR "D" DIMENSION SEE HI-LOK INSTALLATION DATA TABLE.
FOR OTHER CODED DIMENSIONS SEE "INSPECTION".



FLUSH HEAD

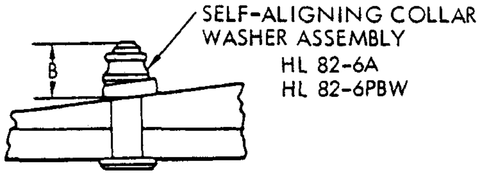

A
0.008
W
0.020
0.010
R
0.015
CHAMFER OPTIONAL WITHIN LIMITS AS SHOWN
TYPICAL PROTRUDING OR FLUSH-HEAD HI-LOK
WRENCH
COLLAR HEX
ALLEN HEX-KEY

APPLY WRENCH TO COLLAR-HEX. PREVENT PIN FROM TURNING WITH ALLEN HEX-KEY. APPLY TORQUE WITH WRENCH UNTIL COLLAR HEX BREAKS OFF.

JS-1-2-X0/2-5916-1

Figure 59-16. Hi-Lok Installation, Inspection, and Removal (Sheet 1 of 4)

INSTALLATION (CONT)



HI-LOK PIN TYPE

INSTALLATION OF HI-LOKS THROUGH SLOPING SURFACE
(EXCEEDING 3° BUT NOT EXCEEDING 7°)
TYPICAL FOR PROTRUDING AND FLUSH-HEAD HI-LOKS.
MEASURE "B" DIMENSION PARALLEL TO AND AT
CENTERLINE OF PIN AS SHOWN.

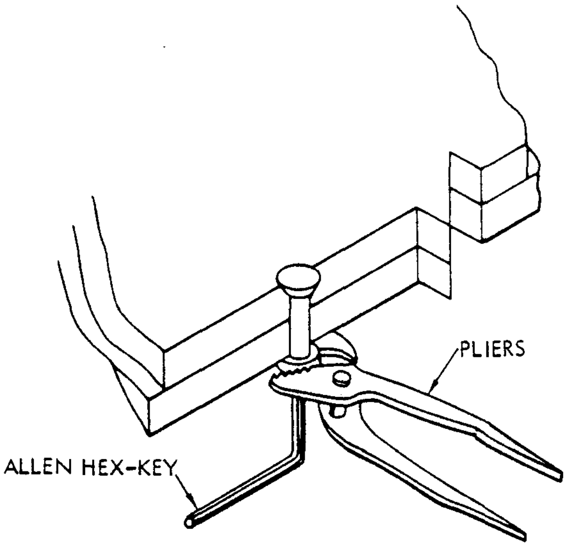
INSPECTION

(COORDINATE WITH TEXT IN THIS SECTION)

REFER TO INSTALLATION SKETCHES IN THIS FIGURE. VALUES FOR CODED DIMENSION "B" ARE GIVEN IN THE HI-LOK INSTALLATION DATA TABLE.

REMOVAL

(COORDINATE WITH TEXT IN THIS SECTION)



USE PLIERS TO REMOVE THE COLLAR. USE ALLEN HEX-KEY TO PREVENT THE PIN FROM TURNING. CAREFULLY DRIVE OUT PIN WITH A PIN PUNCH.

JS-1-2-X0/2-5916-2

Figure 59-16. Hi-Lok Installation, Inspection, and Removal (Sheet 2 of 4)

HI-LOK INSTALLATION DATA								
PART NUMBER △8			GAGE DIM	PRE DRILL SIZE	HOLE DIAMETER D			
PRO-TRUDING HEAD	FLUSH HEAD	COLLAR	B LIMITS		ALUM. ALLOYS ALL GRIP LENGTHS		FERROUS & TI ALL GRIP LENGTHS	
					MIN	MAX	MIN	MAX
HL928-5	HL929-5	HL70-5	0.302 0.384	#30	0.1610	0.1640	0.1635	0.1665
HL928-6	HL929-6	HL70-6	0.315 0.397	#20	0.1870	0.1900	0.1895	0.1925
HL928-8	HL929-8	HL70-8	0.385 0.467	#1	0.2470	0.2500	0.2495	0.2525
HL928-10	HL929-10	HL70-10	0.490 0.572	9/32	0.3095	0.3125	0.3120	0.3150

HI-LOK INSTALLATION DATA											
PART NUMBER △8			GAGE DIM	PRE DRILL SIZE	HOLE DIAMETER D				SUBSTITUTION DATA △1		
PRO-TRUDING HEAD	FLUSH HEAD	COLLAR	B LIMITS		ALUM. ALLOYS ALL GRIP LENTHS		FERROUS & TI ALL GRIP LENGTHS		SCREW OR BOLT	NUT	WASHER △2
					MIN	MAX	MIN	MAX			
HL18PB5	HL19PB5	HL70-5	0.302 0.384	#30	0.1610	0.1640	0.1635	0.1665	NAS1632 NAS1622	MS21042-08	AN960D8L
HL18PB6	HL19PB6	HL70-6	0.315 0.397	#20	0.1870	0.1900	0.1895	0.1925	NAS1633 NAS1623	MS21042-3	AN960D10L
HL18PB8	HL19PB8	HL70-8	0.385 0.467	#1	0.2470	0.2500	0.2495	0.2525	NAS1104	MS21042-4	AN960D416L
HL18PB10	HL19PB10	HL70-10	0.490 0.572	9/32	0.3095	0.3125	0.3120	0.3150	NAS1105	MS21042-5	AN960D516L

1/64TH OVERSIZE HI-LOK INSTALLATION DATA								
OVERSIZE FOR HI-LOK PIN △3	PIN PART NUMBER		COLLAR	GAGE DIMENSIONS	HOLE DIAMETER D			
	PRO-TRUDING HEAD	FLUSH HEAD		B LIMITS	ALUM. ALLOYS ALL GRIP LENGTHS		FERROUS & TI ALL GRIP LENGTHS	
					MIN	MAX	MIN	MAX
HL928/9-5 HL18/9PB5	HL928-6 HL18PB6	HL929-6 HL19PB6	HL70-6	0.315/0.397	0.1870	0.1900	0.1895	0.1925
HL928/9-6 HL18/9B6	HL962-6 HL62PB6	HL963-6 HL63PB6	HL79-6 OR HL84-6	0.315/0.397	0.2001	0.2031	0.2026	0.2056
HL928/9-8 HL18/9BP8	HL962-8 HL62PB8	HL963-8 HL63PB8	HL79-8 OR HL84-8	0.385/0.467	0.2626	0.2656	0.2651	0.2681
HL928/9-10 HL18/9PB10	HL962-10 HL62PB10	HL963-10 HL63PB10	HL79-10 OR HL84-10	0.490/0.572	0.3251	0.3281	0.3276	0.3306

JS-1-2-X0/9-5916-3

Figure 59-16. Hi-Lok Installation, Inspection, and Removal (Sheet 3 of 4)

1/32ND OVERSIZE HI-LOK INSTALLATION DATA

OVERSIZE FOR HI-LOK PIN	PIN PART NUMBER		COLLAR	GAGE DIMENSIONS	HOLE DIAMETER D			
					ALUM. ALLOYS ALL GRIP LENGTHS		FERROUS & TI ALL GRIP LENGTHS	
	PRO-TRUDING HEAD	FLUSH HEAD		B LIMITS	MIN	MAX	MIN	MAX
HL928/9-5 HL18/9PB5	HL928-6 HL18PB6	HL929-6 HL19PB6	HL70-6	0.315/0.397	0.1870	0.1900	0.1895	0.1925
HL928/9-6 HL18/9PB6	HL964-6 HL218-6	HL965-6 HL219-6	HL84-6	0.315/0.397	0.2157	0.2187	0.2182	0.2212
HL928/9-8 HL18/9PB8	HL964-8 HL218-8	HL965-8 HL219-8	HL84-8	0.385/0.467	0.2782	0.2812	0.2807	0.2837
HL928/9-10 HL18/9PB10	HL964-10 HL218-10	HL965-10 HL219-10	HL84-10	0.490/0.572	0.3407	0.3437	0.3432	0.3462

NOTE

△1 OBTAIN APPROVAL BY THE AIRCRAFT MANUFACTURER BEFORE USING THESE SUBSTITUTE FASTENERS. THIS DATA IS BASED UPON THE PREMISE THAT THE REUSED HOLE IS NOT DAMAGED OR ELONGATED, THAT SELECT FIT OF THE SUBSTITUTE FASTENER IS REQUIRED, OR THAT A NEW HOLE PROVIDES A DRIVE FIT. IF THE HOLE IS ENLARGED, USE THE APPROPRIATE 1/64TH OVERSIZE HI-LOK.

△2 A MAXIMUM OF TWO WASHERS (TOTAL PER FASTENER INSTALLATION) MAY BE USED UNDER THE PROTRUDING HEAD OR NUT.

△3 DO NOT USE 1/64TH AND 1/32ND OVERSIZE HI-LOK PINS IN VIOLATION OF THE TABLE UNLESS APPROVED BY AIRCRAFT MANUFACTURER.

4. NO GAP IS ALLOWED BETWEEN ASSEMBLED MEMBERS OF THE FASTENER AFTER THE COLLAR-HEX IS BROKEN OFF.

5. A MAXIMUM GAP OF 0.004 INCH IS PERMITTED UNDER ONE SIDE OF THE PROTRUDING (MANUFACTURED) HEAD PROVIDING THE NUMBER OF SUCH ATTACHMENTS DO NOT EXCEED 10 PERCENT OF THE TOTAL FASTENERS IN THE PATTERN AND ARE NOT ADJACENT TO ONE ANOTHER.

6. A MAXIMUM GAP OF 0.003 INCH IS PERMITTED BETWEEN THE FLUSH-MANUFACTURED AND THE COUNTERSINK, PROVIDING THE GAP DOES NOT EXTEND FOR MORE THAN ONE-THIRD THE PERIPHERY OF THE HEAD.

7. NO GAP IS ALLOWED UNDER THE COLLAR AFTER COLLAR-HEX BREAKOFF.

△8 SEVERAL DIFFERENT PART NUMBER HI-LOKS OTHER THAN THOSE LISTED WILL OCCUR IN MANY AREAS OF THE AIRPLANE STRUCTURE. IF A SUBSTITUTION FOR A HI-LOK NOT LISTED HERE IS NECESSARY, CONTACT THE AIRCRAFT MANUFACTURER FOR AN ACCEPTABLE SUBSTITUTE.

9. NUTS MAY BE SUBSTITUTED FOR HL70-( ), HL79-( ) AND HL84-( ) COLLARS IN ACCORDANCE WITH THE FOLLOWING TABLE:

SIZE	NUT PART NO.	INSTALLATION TORQUE
-5	KFN 618-5	15-25 IN-LBS.
-6	KFN 618-6	25-35 IN-LBS.
-8	KFN 618-8	60-80 IN-LBS.
-10	KFN 618-10	140-160 IN-LBS

JS-1-2-X0/9-5916-4

Figure 59-16. Hi-Lok Installation, Inspection, and Removal (Sheet 4 of 4)

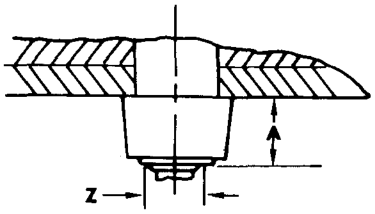
HI-LOK FASTENER		RATCHET WRENCH PART NO. [1]	SOCKET PART NO. [1]	ALLEN WRENCH PART NO. [1]	HAND TOOL ASSY PART NO. [1]
FIRST DASH NO.	NOM DIA.				
-5, -6	5/32, 3/16	HLH101-21	HLH102-6	NO. 607 5/64 HEX	HLH110-6
-8	1/4	HLH101-21	HLH102-8	NO. 609 3/32 HEX	HLH110-8
-10	5/16	HLH101-23	HLH102-10	NO. 609 1/8 HEX	HLH110-10
-12	3/8	HLH101-23	HLH102-12	NO. 609 5/32 HEX	HLH110-12

NOTE

[1] HI-SHEAR CORP. PART NUMBERS.

JS-1-2-X0/2-5917

Figure 59-17. Hand Tools for Hi-Lok Fastener Installation

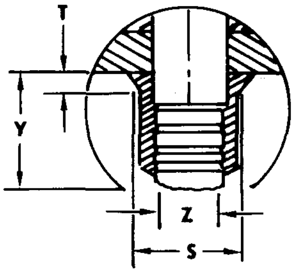


NAS1080, K PULL TYPE PIN (SHEAR)

LOCKBOLT NOMINAL DIAMETER	COLLAR TYPE	PULL TYPE LOCKBOLTS		
		INSPECTION GAGE	A	Z MAX
5/32	NAS1080K05	HG90-5	0.181 0.261	—
3/16	NAS1080K06	HG90-6	0.186 0.279	—
1/4	NAS1080K08	HG90-8	0.226 0.318	—
5/16	NAS1080K10	HG90-10	0.265 0.366	—
3/8	NAS1080K12	HG90-12	0.298 0.403	—

JS-1-2-X0/2-5918-1

Figure 59-18. Lockbolt Collar Inspection (Sheet 1 of 3)

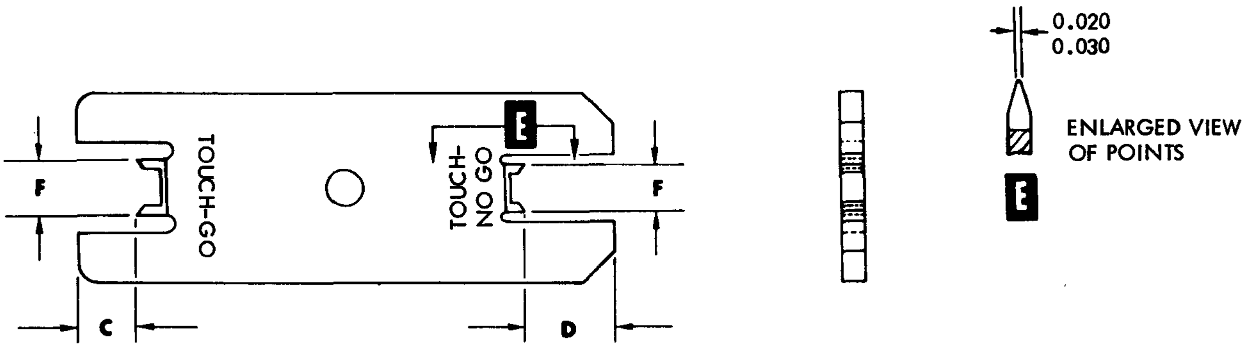


PULL TYPE

LOCKBOLT DIA NOM	COLLAR SWAGE AND PIN PROTRUSION DIMENSIONS			
	TENSION LOCKBOLT			
	PULL TYPE			
	T MAX	Y	S MAX	Z MAX
5/32	0.030	0.230 0.167	0.254	0.137
3/16	0.060	0.280 0.208	0.304	0.165
1/4	0.038	0.374 0.295	0.401	0.225

JS-1-2-X0/2-5918-2

Figure 59-18. Lockbolt Collar Inspection (Sheet 2 of 3)



HG-76
GAGE USED FOR CHECKING DRIVEN
SHEAR TYPE HUCKBOLT PIN

USE OF GAGES

1. CENTER GAGE OVER PIN AND COLLAR WITH "TOUCH-GO" END DOWN.
   IF POINTS DO NOT TOUCH, PIN IS TOO SHORT
   OR PULL-UP IS NOT COMPLETE; REJECT.
   IF POINTS DO TOUCH, USE OPPOSITE END OF GAGE
2. CENTER GAGE OVER PIN AND COLLAR "TOUCH-NO GO" END DOWN.
   IF POINTS DO NOT TOUCH, INSTALLATION IS SATISFACTORY.
   IF POINTS DO TOUCH, PIN IS TOO LONG; REJECT.

GAGE NO.	FOR USE WITH HUCKBOLT	C +0.001 -0.000	D +0.000 -0.001	F ± 0.001
HG-76-1	SAL-T6	0.191	0.264	0.164
HG-76-2	SAL-T8	0.243	0.316	0.224
HG-76-3	SAL-T10	0.268	0.341	0.268
HG-76-4	SAL-T12	0.319	0.390	0.339

JS-1-2-X0/2-5918-3

Figure 59-18. Lockbolt Collar Inspection (Sheet 3 of 3)

FE AND TI

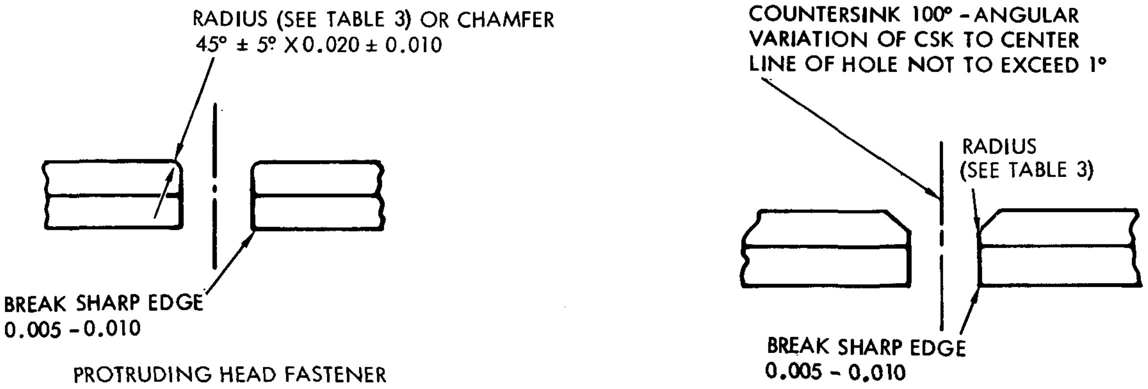
DASH NO.	NOM DIA	HOLE SIZE (GRIP THRU -12)	HOLE SIZE (GRIP -12 & UP)	HOLE SIZE (ALL GRIP LENGTHS)
-5	5/32	0.1645/0.1615	0.1655/0.1615	0.1655/0.1615
-6	3/16	0.1910/0.1880	0.1920/0.1880	0.1920/0.1880
-8	1/4	0.2510/0.2480	0.2520/0.2480	0.2520/0.2480
-10	5/16	0.3135/0.3105	0.3145/0.3105	0.3145/0.3105

TABLE 1. STANDARD FIT HOLES FOR TENSION TYPE LOCKBOLTS

FE AND TI

DASH NO.	NOM DIA	HOLE SIZE (GRIP THRU -12)	HOLE SIZE (GRIP -12 & UP)	HOLE SIZE (ALL GRIP LENGTHS)
-5	5/32	0.1575/0.1545	0.1585/0.1545	0.1585/0.1545
-6	3/16	0.1910/0.1880	0.1920/0.1890	0.1920/0.1890
-8	1/4	0.2510/0.2480	0.2520/0.2490	0.2520/0.2490
-10	5/16	0.3135/0.3105	0.3145/0.3115	0.3145/0.3115

TABLE 2. STANDARD FIT HOLES FOR SHEAR TYPE LOCKBOLTS


RADIUS (SEE TABLE 3) OR CHAMFER
45° ± 5° X 0.020 ± 0.010
BREAK SHARP EDGE
0.005 - 0.010
COUNTERSINK 100° - ANGULAR
VARIATION OF CSK TO CENTER
LINE OF HOLE NOT TO EXCEED 1°
RADIUS
(SEE TABLE 3)
BREAK SHARP EDGE
0.005 - 0.010

PROTRUDING HEAD FASTENER

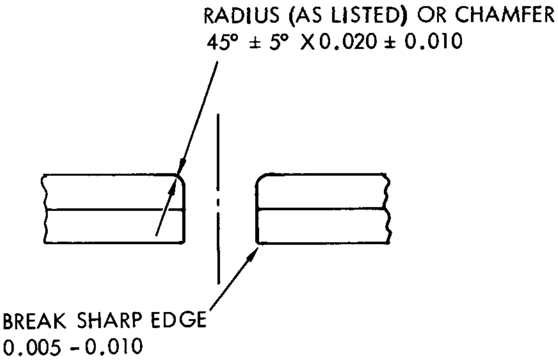
FLUSH HEAD FASTENER

RADIUS SHEAR TYPE		
DIA	MIN	MAX
-5 -6 -8 -10	0.015 0.020 0.020 0.030	0.025 0.030 0.030 0.040

TABLE 3. HOLE PREPARATION FOR SHEAR TYPE LOCKBOLTS

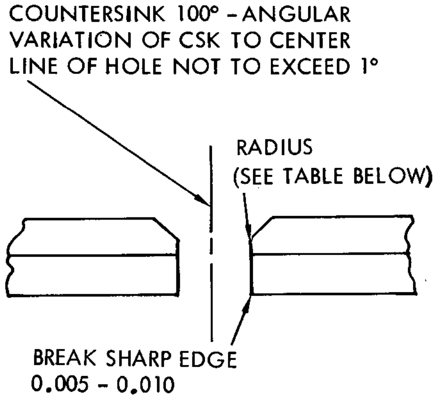
JS-1-2-X0/2-5919-1

Figure 59-19. Lockbolt Installation and Removal (Sheet 1 of 4)



NOM PIN DIA	5/32 THRU 1/4	5/16
RADIUS	0.015 - 0.025	0.020 - 0.030

PROTRUDING HEAD FASTENER


COUNTERSINK 100° - ANGULAR
VARIATION OF CSK TO CENTER
LINE OF HOLE NOT TO EXCEED 1°
RADIUS
(SEE TABLE BELOW)
BREAK SHARP EDGE
0.005 - 0.010

NOM PIN DIA		5/32	3/16	1/4	5/16
RADIUS	MIN	0.015	0.020		0.030
	MIN	0.020	0.030		0.040

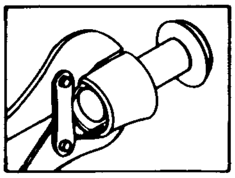
FLUSH HEAD FASTENER

TABLE 4. HOLE PREPARATION
FOR TENSION TYPE LOCKBOLTS

NOTE

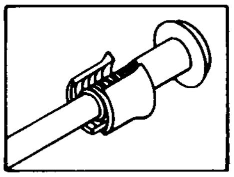
1. USE 1/64 AND 1/32 OVERSIZE PROTRUDING HEAD HILOKS OF COMPARATIVE STRENGTH, SIZE AND GRIP FOR OVERSIZE REPLACEMENT OF PROTRUDING HEAD LOCKBOLTS. FOR DIRECT SUBSTITUTION HL-928/HL-929 ☞ HI-LOKS MAY BE USED IN LIEU OF LOCKBOLTS.

REMOVAL



REMOVE COLLAR

VIEW A



DRIVE PIN OUT

VIEW B

REMOVAL OF LOCKBOLT PIN ON STUMP

COLLAR REMOVAL TOOLS	
LOCKBOLT PIN DIA	TOOL PART NO.
3/16	42-6
1/4	42-8
5/16	42-10

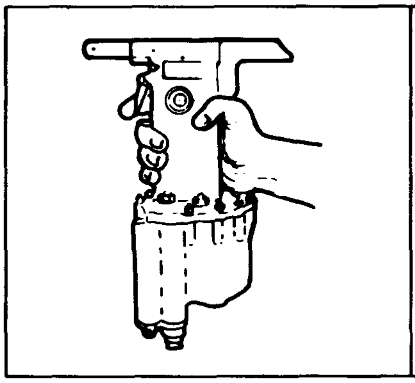
JS-1-2-X0/8-5919-2

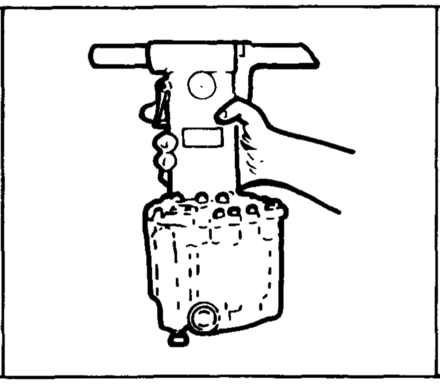
Figure 59-19. Lockbolt Installation and Removal (Sheet 2 of 4)

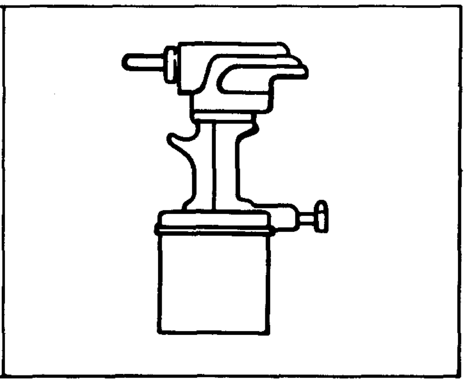
FASTENER	FASTENER DIAMETER	INSTALLATION TOOL											
		HAND			PNEU-DRAULIC	PNEUMATIC 90 PSI		HYDRAULIC 5000 PSI		HYDRAULIC 3000 PSI			
		115	116	194	200	352	353	504	505	609	610 613	611 612	614
		NOSE ASSEMBLY											
*	6				99-608	99-608				99-541	99-395	99-156	
	8	99-609					99-609			99-216	99-126	99-165	
	10	99-622	99-664				99-622	99-664	99-664			99-135	
	12	99-613	99-651				99-613	99-651	99-651			99-145	99-410

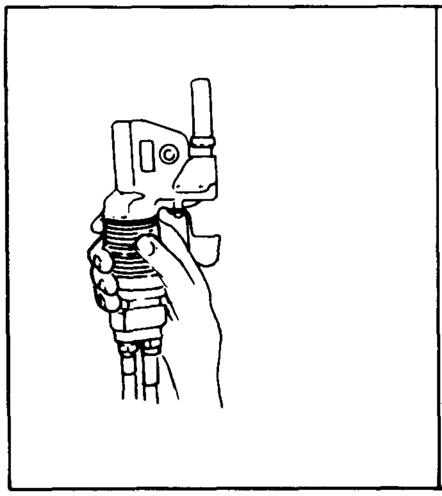
* NAS 1400, 1500 AND 7000 SERIES LOCKBOLTS

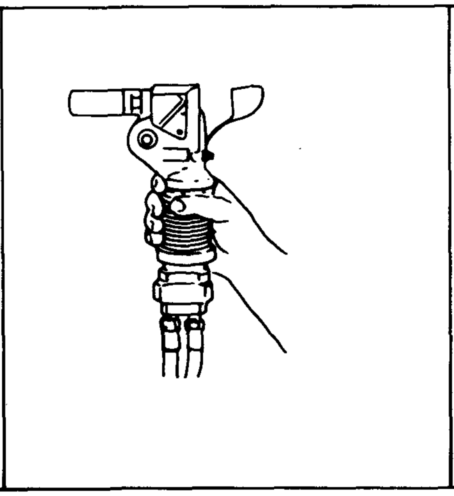
INSTALLATION TOOLS

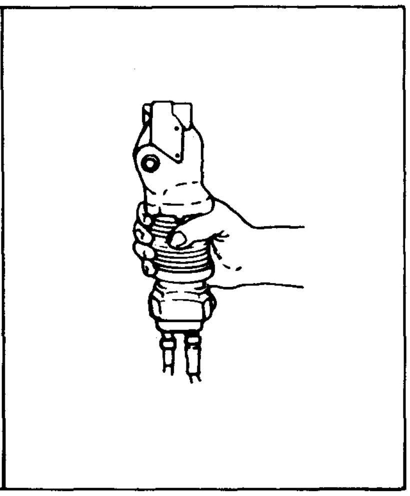

HUCK MODEL 352
PNEUMATIC PULL TOOL


HUCK MODEL 353
PNEUMATIC PULL TOOL


HUCK MODEL 200
PNEUMATIC INSTALLATION TOOL


HUCK MODEL 612 OR 613
HYDRAULIC PULL TOOL

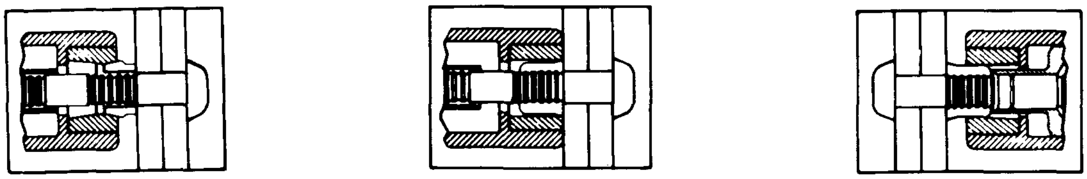

HUCK MODEL 610 OR 611
HYDRAULIC PULL TOOL


HUCK MODEL 609
CLEARANCE HYDRAULIC PULL TOOL

JS-1-2-X0/2-5919-3

Figure 59-19. Lockbolt Installation and Removal (Sheet 3 of 4)

DRIVING CYCLE OF THE LOCKBOLT PIN (PULL TYPE)



COLLAR AND PIN IN POSITION PRIOR TO DRIVING WITH GUN NOSE OVER PIN PULL GROOVES.

LOCKBOLTS IN PROCESS OF DRIVING. COLLAR SWAGED INTO LOCKING GROOVES.

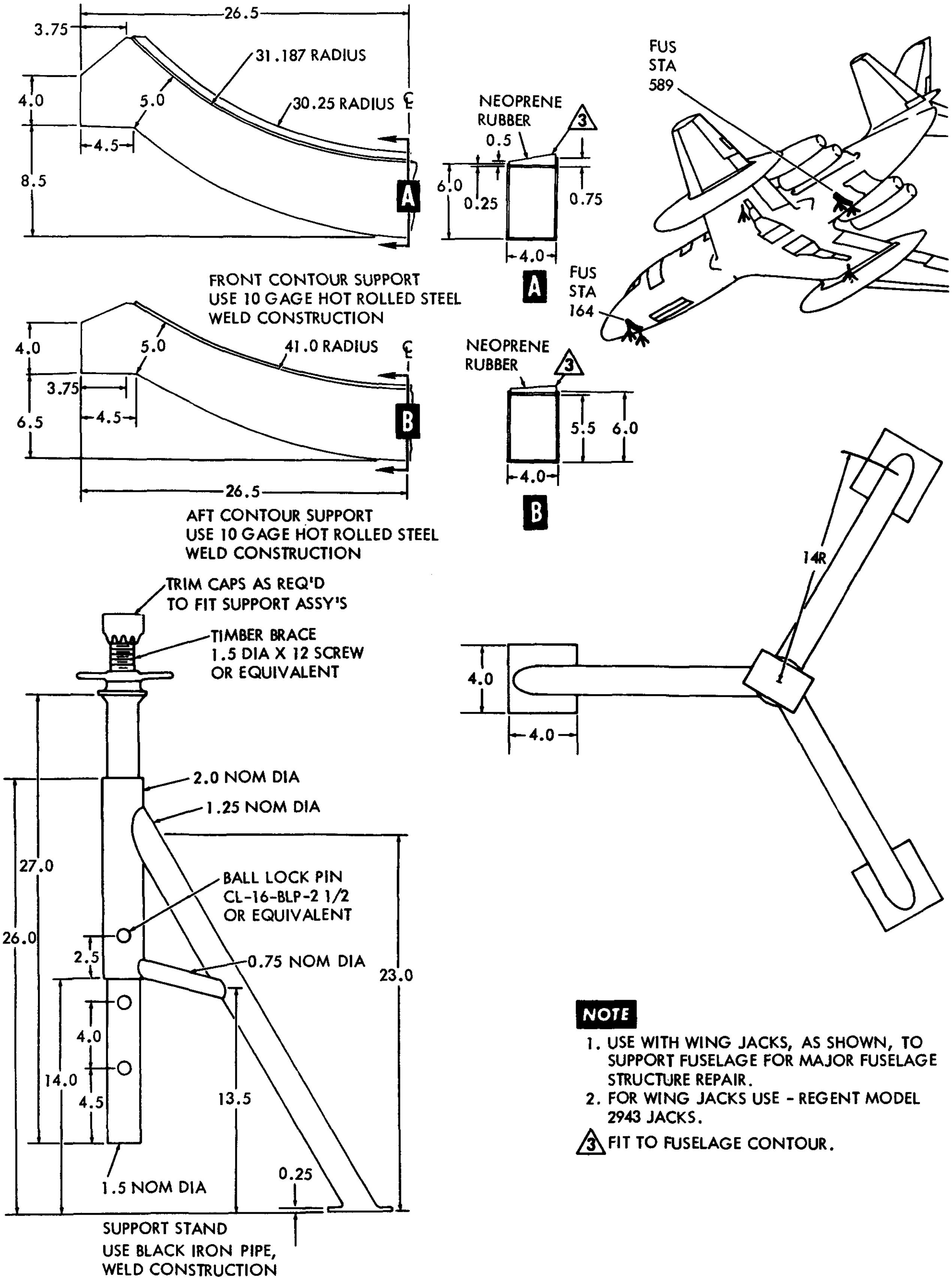
LOCKBOLT INSTALLED AND PIN BROKEN OFF AT BREAK-NECK GROOVE.

NOTE

1. PIN CANNOT BE PULLED UNTIL GUN IS IN FULL RETURN POSITION.
2. WASHERS MAY BE USED UNDER THE HEAD OR COLLAR WHEN CORRECT BOLT LENGTH IS UNAVAILABLE.
3. DO NOT RE-USE PINS THAT HAVE BEEN DRIVEN OUT.

JS-1-2-X0/2-5919-4

Figure 59-19. Lockbolt Installation and Removal (Sheet 4 of 4)

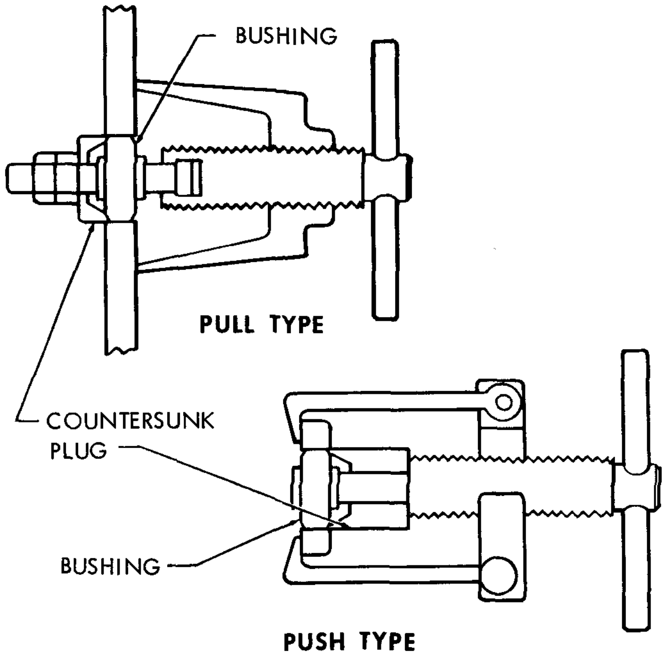


NOTE

1. USE WITH WING JACKS, AS SHOWN, TO SUPPORT FUSELAGE FOR MAJOR FUSELAGE STRUCTURE REPAIR.
2. FOR WING JACKS USE - REGENT MODEL 2943 JACKS.

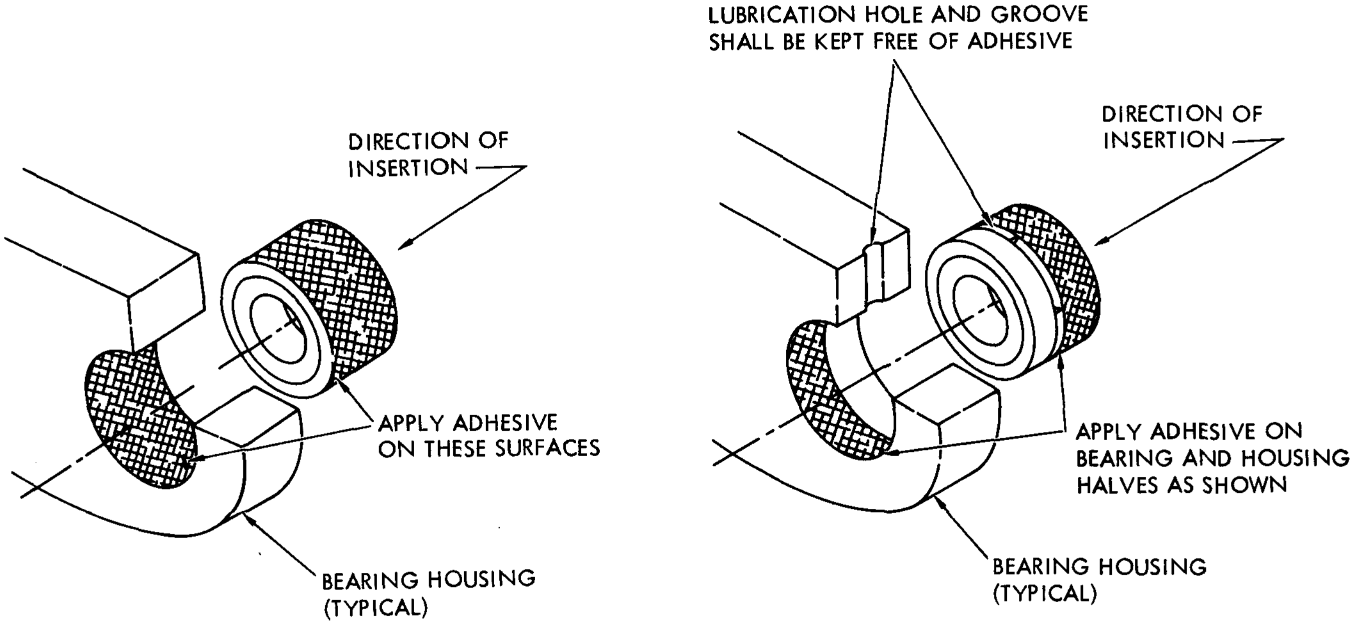
△3 FIT TO FUSELAGE CONTOUR.

Figure 59-20. Fuselage Support Cradles



JS-1-2-X0/2-5921

Figure 59-21. Bushing Removal and Installation


DIRECTION OF INSERTION
APPLY ADHESIVE ON THESE SURFACES
BEARING HOUSING (TYPICAL)
LUBRICATION HOLE AND GROOVE SHALL BE KEPT FREE OF ADHESIVE
DIRECTION OF INSERTION
APPLY ADHESIVE ON BEARING AND HOUSING HALVES AS SHOWN
BEARING HOUSING (TYPICAL)

JS-1-2-X0/6-5922

Figure 59-22. Installation of Bearing Using Adhesive

SECTION 59.2

WING GROUP

Contents Page

DESCRIPTION 59-59

REPAIRS 59-59
Fuel Tank Structural Repair Sealing 59-59
Repair of External Wing Tank Landing Light Cover Assemblies 59-59
Repair of Lower Root Rib Shear Fittings 59-59
Repair of Wing Corrosion Damage 59-59
Replacement of Aileron Trailing Edge Assemblies 59-60
Repair Requirements for JW60 Landing Gear Actuator and Auxiliary Spar Outboard Support Fitting 59-60
Repair of Damaged Flux Valve Bracket 59-60

FIGURES
59-22 Leading Edge Flap Aft Beam Upper Cap Repair 59-61
59-23 Leading Edge Flap Hinge Negligible Damage Limits 59-62
59-23A Aileron Hinge Negligible Damage Limits 59-62A
59-24 Wing Lower Root Rib Shear Fitting Repair 59-62
59-25 Deleted
59-26 Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) 59-66
59-27 Deleted
59-28 Deleted
59-28A Basic Wing Layout 59-70A
59-28B Typical Lower Surface Skin Panel Corrosion Repair........ 59-70B
59-28C Wing Lower Surface Skin Panel Corrosion Grindout Limits 59-70P
59-28D Wing Lower Surface Panel Riser Cap Corrosion Grindout Limits 59-70P
59-28E Landing Gear Actuator and Auxiliary Spar Outboard Support Fitting (JW60) Repair 59-70Q

59.2.1 DESCRIPTION. See Chapter 57 for a complete description of the wing and wing structure. Detail component material and identification figures are also given in Chapter 57 to aid in establishing the detail part and the applicable repair figure in Section 59.2 and 59.8.

59.2.2 REPAIRS.

59.2.2.1 FUEL TANK STRUCTURAL REPAIR SEALING. Refer to figure 59-28B, Notes 10, 11 and 23 for requirements on sealing box beam structure after repairs.

59.2.2.2 REPAIR OF EXTERNAL WING TANK LANDING LIGHT COVER ASSEMBLIES. The cover assemblies are composed of Plexiglas and fiberglass covers cemented together. When the Plexiglas portion of the cover assembly is cracked, repair as follows:

a. Stop-drill both ends of any crack up to 2 inches long, using a No. 30 drill.

b. Replace the Plexiglas cover if a crack is longer than 2 inches or if there are more than three cracks in the cover.

59.2.2.3 REPAIR OF LOWER ROOT RIB SHEAR FITTINGS. (Airplanes 5001 through 5059, 5061, and 5064.) Instances have been found where hairline cracks have originated from the end of machined slots in fittings. If cracks exist, stop-drill each crack with 0.127 - 0.133 inch (No. 30 drill) diameter hole at lower end of cracks. See figure 59-24.

Note

> **If cracks have progressed beyond the point indicated in figure 59-24 and into the base of fittings, DO NOT STOP-DRILL. Contact the manufacturer, Lockheed-Georgia Company, Marietta, Georgia.**

59.2.2.4 REPAIR OF WING CORROSION DAMAGE. When corrosion occurs in wing panels that exceeds the allowable corrosion limits as shown in Figure 59-28C and 59-28D, then repair as shown in Figure 59-28B. Refer to paragraph 51.7.3 and Service Bulletin 329-168 for treatment of wing corrosion damage and additional information.

The tables in Figure 59-28C and 59-28D give permissible reductions in area of the JetStar lower surface wing panels due to corrosion before structural repair is required. The permissible area reductions apply to the

59.2.2.4 (Cont).

4-inch wide panel bay, expect for the spanwise splice fastener and the spar cap fastener areas. The area reductions do not apply when the depth of grindout exceeds 0.010 inch within 3/8 inch of any riser or when the depth of grindout exceeds 1/2 of the original panel thickness. Assistance of Lockheed Engineering will be required for corrosion repair to upper surface panels.

Application of Data:

The wing panels are divided into bays bounded by the ribs and spars (except bays No. 3 and 3A). They are identified as shown in Figure 59-28A (Basic Wing Layout).

In calculating the area reduction due to corrosion, sum up the area lost along a line perpendicular (chordwise) to the panel risers. Select the location in each of the 4" wide panel bays which yields the maximum lost net section. The area lost between two risers is calculated by multiplying the corrosion pit diameter or chordwise dimension by the maximum pit depth and summing these areas for each 4" panel bay. Pit dimensions are those after complete removal of all corrosion as shown in Figure 59-28A.

To obtain the area reduction permissible across each 4" panel bay, locate the wing bay in Figure 59-28A (Basic Wing Layout) in which the panel falls. Read from the table in Figure 59-28C the area reduction permissible of the appropriate bay.

To obtain the area reduction permissible for each riser cap follow the above procedure except read area reduction permissible values from the table in Figure 59-28D. Summation of the area lost on riser cap and the adjacent 4" skin panel (left or right of riser) along a line perpendicular to the panel riser shall not exceed the limits shown in the table in Figure 59-28C.

Area reductions permissible along the spanwise (parallel to panel risers) panel length are equal to 4 times the area reduction permissible in the chordwise direction.

Note

If additional corrosion is found after initial inspection, be sure and include all previously found corrosion in determining permissible area reduction.

59.2.2.5 REPLACEMENT OF AILERON TRAILING EDGE ASSEMBLIES. Allowable rivets to be used for the installation of the aileron trailing edge assemblies are listed below.

HOLE DIAMETER	APPROVED FASTENERS
0.141 to 0.144	NAS1739E4-X CR3242-4-X CR3552P-4-X
0.156 to 0.159	CR3522P-5-X CR3212-5-X CR3222-5-X
0.173 to 0.176	NAS1739E5-X CR3242-5-X CR3252-5-X CR3552P-5-X

Note

1. The smallest diameter rivet, using the above noted hole size, will be installed.
2. Indiscriminate oversize installation is not permitted.
3. These rivets shall be installed in countersink holes with the heads of the rivets protruding 0.002 to 0.004 inch.
4. Holes for blind rivets should be carefully drilled. A light interference fit is desirable for rivets in this application.

59.2.2.6 REPAIR REQUIREMENTS FOR JW60 LANDING GEAR ACTUATOR AND AUXILIARY SPAR OUTBOARD SUPPORT FITTING. If cracks are found as a result of the inspection in paragraph 58.2.6, see figure 59-28E and proceed as follows:

a. If crack is more than 0.5 inch from the inboard edge of the slot, stop drill the crack.

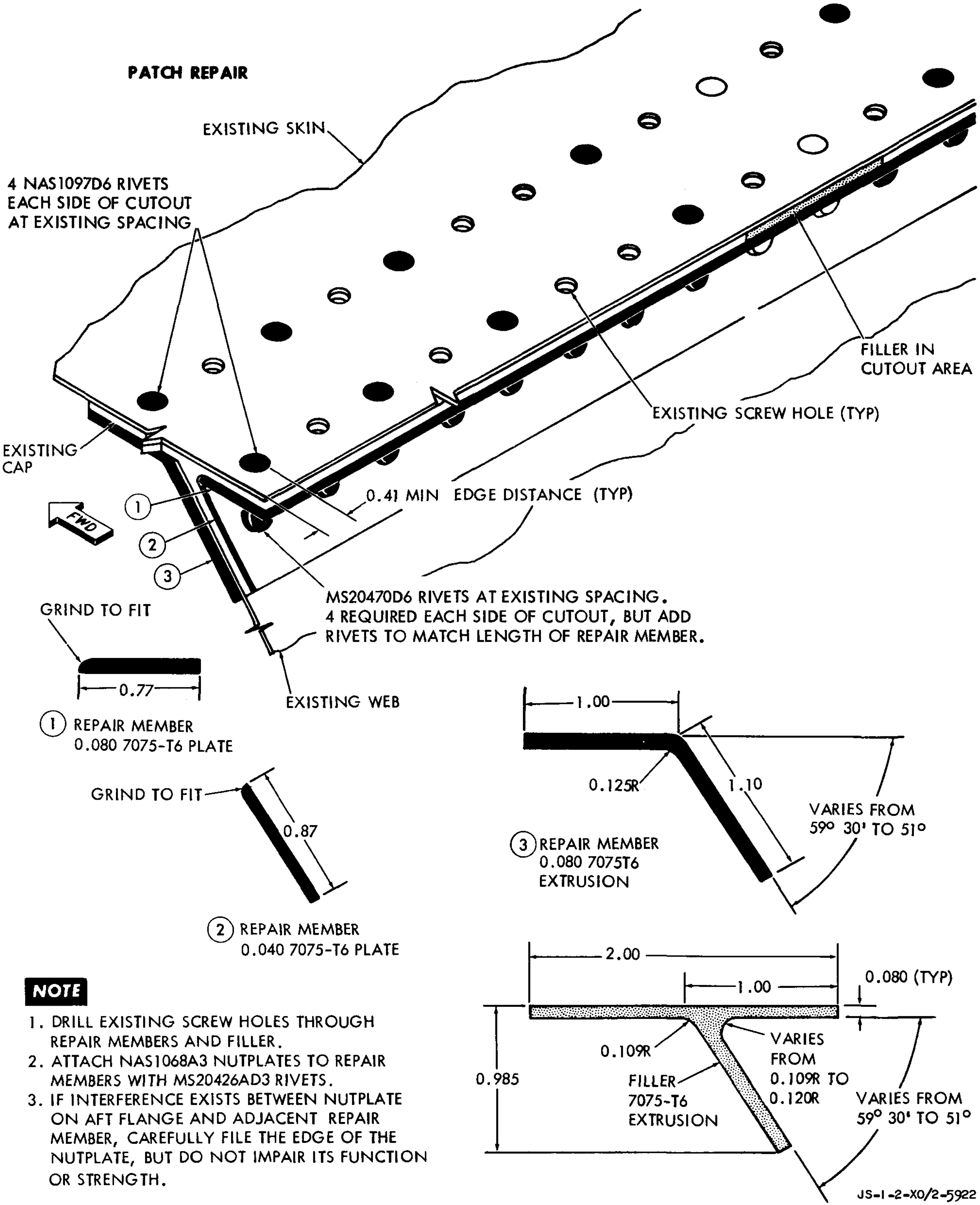
b. If crack is less than 0.5 inch from the inboard edge of the slot, replace the JW60 fitting assembly.

c. If crack extends more than 0.5 inch down the vertical side of the slot, replace the JW60 fitting assembly.

Measure all cracks found to be within acceptable limits and log into aircraft records.

Refinish reworked parts with Alodine 1200S and two coats of zinc chromate primer per Specification TT-P-1757. (Refer to Section 51.8.)

59.2.2.7 REPAIR OF DAMAGED FLUX VALVE BRACKET. The Flux Valve Bracket, P/N JR318-4L/R, may be repaired with standard methods herein. In case of extensive damage, it may be more feasible to replace this bracket with a new one of local fabrication. The original bracket is made of 0.040 2024-T3 aluminum with 0.12 minimum angle bend radii. For this gauge and minimum radius, the current recommendation is heat treated 2024-T4 with forming in the "O" condition. Any replacement bracket should hold the flux valve in the same position as the original and use only nonmagnetic materials.


PATCH REPAIR
EXISTING SKIN
4 NAS1097D6 RIVETS EACH SIDE OF CUTOUT AT EXISTING SPACING
EXISTING CAP
FWD
①
②
③
0.41 MIN EDGE DISTANCE (TYP)
FILLER IN CUTOUT AREA
EXISTING SCREW HOLE (TYP)
MS20470D6 RIVETS AT EXISTING SPACING. 4 REQUIRED EACH SIDE OF CUTOUT, BUT ADD RIVETS TO MATCH LENGTH OF REPAIR MEMBER.
EXISTING WEB
GRIND TO FIT
0.77
① REPAIR MEMBER 0.080 7075-T6 PLATE
GRIND TO FIT
0.87
② REPAIR MEMBER 0.040 7075-T6 PLATE
1.00
0.125R
1.10
VARIES FROM 59° 30' TO 51°
③ REPAIR MEMBER 0.080 7075T6 EXTRUSION
2.00
1.00
0.080 (TYP)
0.985
0.109R
VARIES FROM 0.109R TO 0.120R
FILLER 7075-T6 EXTRUSION
VARIES FROM 59° 30' TO 51°
JS-1-2-X0/2-5922

NOTE

1. DRILL EXISTING SCREW HOLES THROUGH REPAIR MEMBERS AND FILLER.
2. ATTACH NAS1068A3 NUTPLATES TO REPAIR MEMBERS WITH MS20426AD3 RIVETS.
3. IF INTERFERENCE EXISTS BETWEEN NUTPLATE ON AFT FLANGE AND ADJACENT REPAIR MEMBER, CAREFULLY FILE THE EDGE OF THE NUTPLATE, BUT DO NOT IMPAIR ITS FUNCTION OR STRENGTH.

Figure 59-22. Leading Edge Flap Aft Beam Upper Cap Repair

NOTE

1. NEGLIGIBLE DAMAGE IS NOT PERMITTED IN EITHER HINGE HALF WITHIN 10 INCHES, INBOARD OR OUTBOARD OF AN ACTUATOR STATION.
2. OUTBOARD OF LE STATION 282 A MAXIMUM OF 10 CRACKED OR MISSING NODES ARE PERMISSIBLE, EXCEPT NOT MORE THAN 2 CRACKED OR MISSING NODES MAY BE ADJACENT.
3. THE CRACKS IN THE NODES MAY BE EITHER PARALLEL TO THE HINGE LINE, OR NORMAL TO THE HINGE LINE. THE CRACKS NORMAL TO THE HINGE LINE MAY NOT EXTEND PAST THE BASE OF THE NODE. IN EITHER CASE, THE CRACKS SHOULD BE STOP-DRILLED AND THESE AREAS THEN PERIODICALLY INSPECTED TO MAKE SURE THAT THE CRACKS ARE NOT EXTENDING INTO THE ADJACENT STRUCTURE.

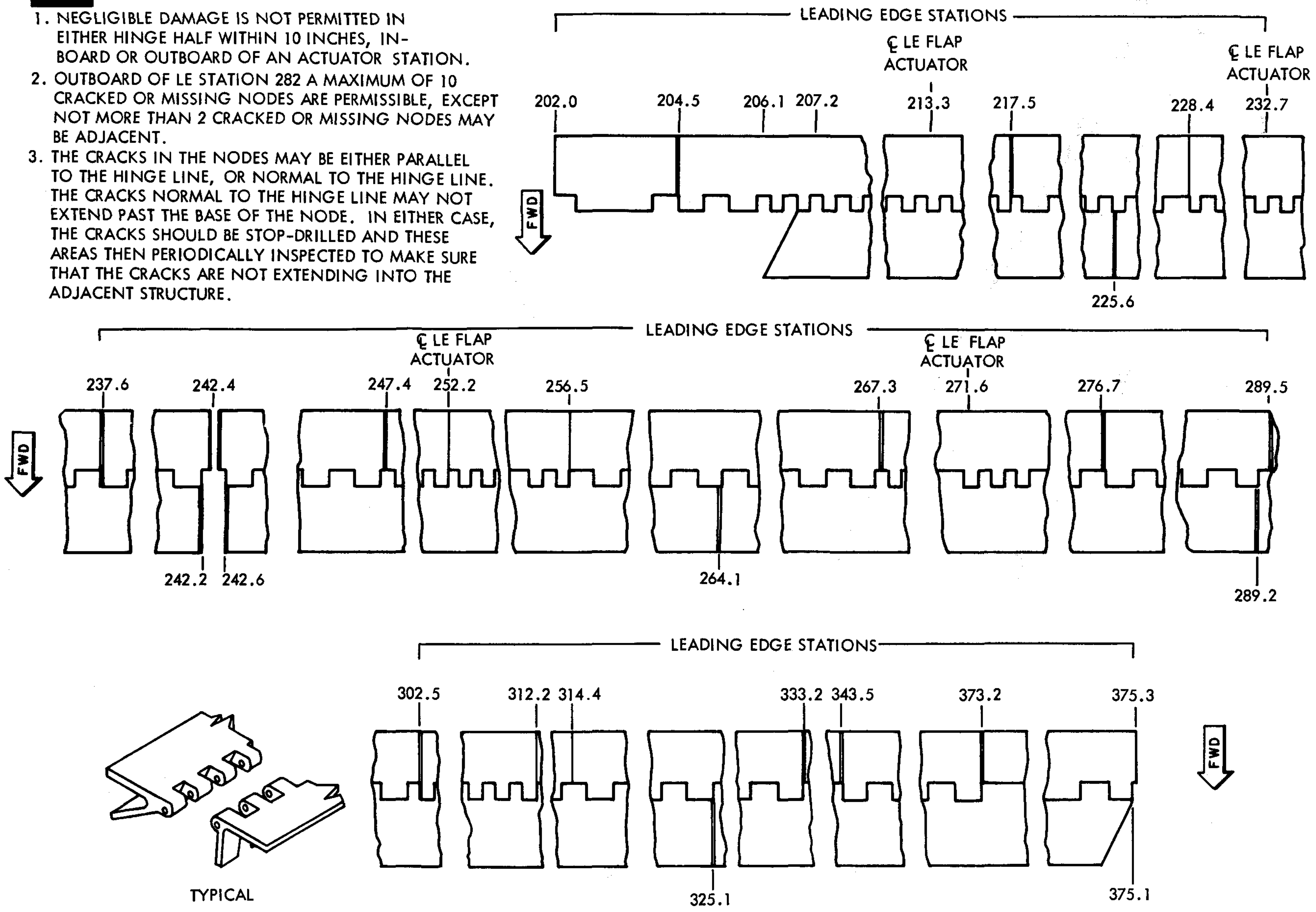
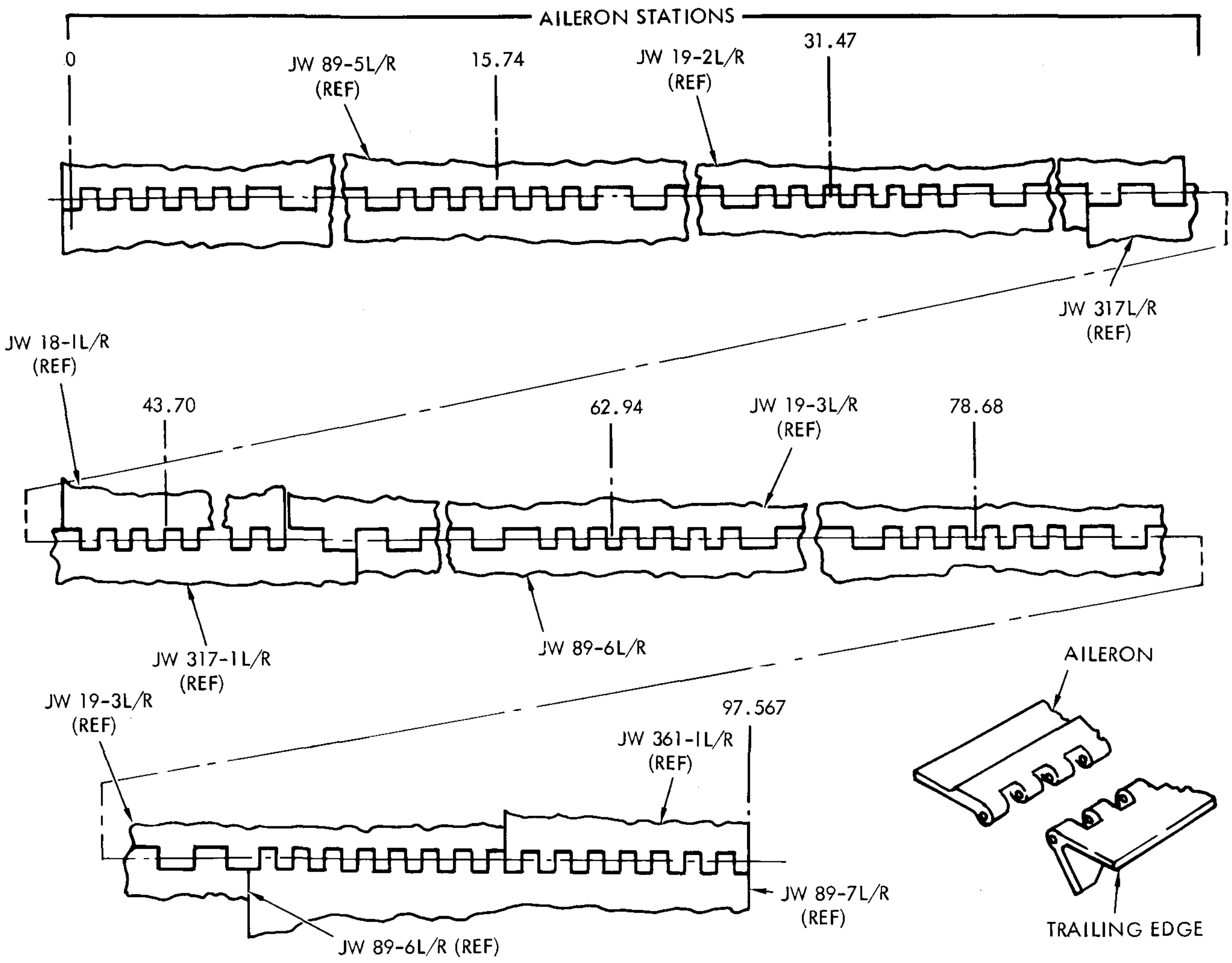


Figure 59-23. Leading Edge Flap Hinge Negligible Damage Limits



NOTE

1. NEGLIGIBLE DAMAGE IS NOT PERMITTED IN EITHER HINGE IN THE FOLLOWING AREAS:
   - A. WITHIN 5 INCHES INBOARD OR OUTBOARD OF AILERON STATION 43.70
   - B. WITHIN 2 INCHES INBOARD OR OUTBOARD OF AILERON STATION 31.47 OR 62.94
   - C. 5 INCHES INBOARD OF THE OUTBOARD END (AIL. STA. 97.567)
2. IN THE REMAINING AREA, A MAXIMUM OF 5 CRACKED OR MISSING NODES ARE PERMISSIBLE. HOWEVER AT AILERON STATION 2.0, 15.74 & 78.68 ONLY ONE SMALL NODE MAY BE DAMAGED WITHIN 4.0 INCH LENGTH AND IN NO CASE SHALL CRACKED OR MISSING NODES BE ADJACENT TO EACH OTHER.
3. THE CRACKS IN THE NODES MAY BE EITHER PARALLEL TO THE HINGE LINE OR NORMAL TO THE HINGE LINE. THE CRACKS NORMAL TO THE HINGE LINE MAY NOT EXTEND BEYOND THE BASE OF THE NODE. IN EITHER CASE, THE CRACKS SHOULD BE STOP DRILLED AND THESE AREAS THEN PERIODICALLY INSPECTED TO MAKE SURE THAT THE CRACKS ARE NOT EXTENDING INTO THE ADJACENT STRUCTURE.

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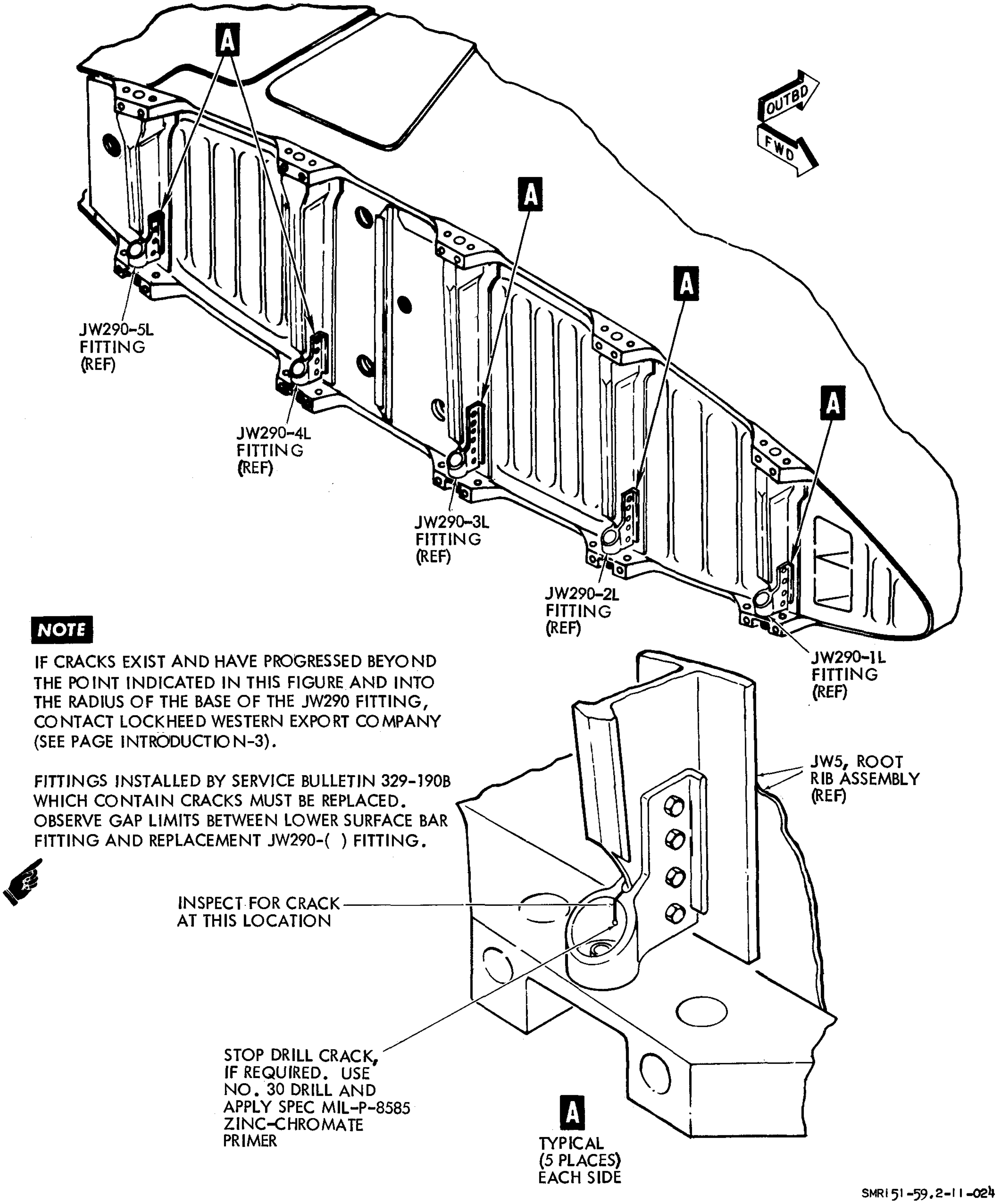
Figure 59-23A. Aileron Hinge Negligible Damage Limits


A
A
A
A
OUTBD
FWD
JW290-5L FITTING (REF)
JW290-4L FITTING (REF)
JW290-3L FITTING (REF)
JW290-2L FITTING (REF)
JW290-1L FITTING (REF)
JW5, ROOT RIB ASSEMBLY (REF)
INSPECT FOR CRACK AT THIS LOCATION
STOP DRILL CRACK, IF REQUIRED. USE NO. 30 DRILL AND APPLY SPEC MIL-P-8585 ZINC-CHROMATE PRIMER
A
TYPICAL (5 PLACES) EACH SIDE
SMR151-59.2-11-024

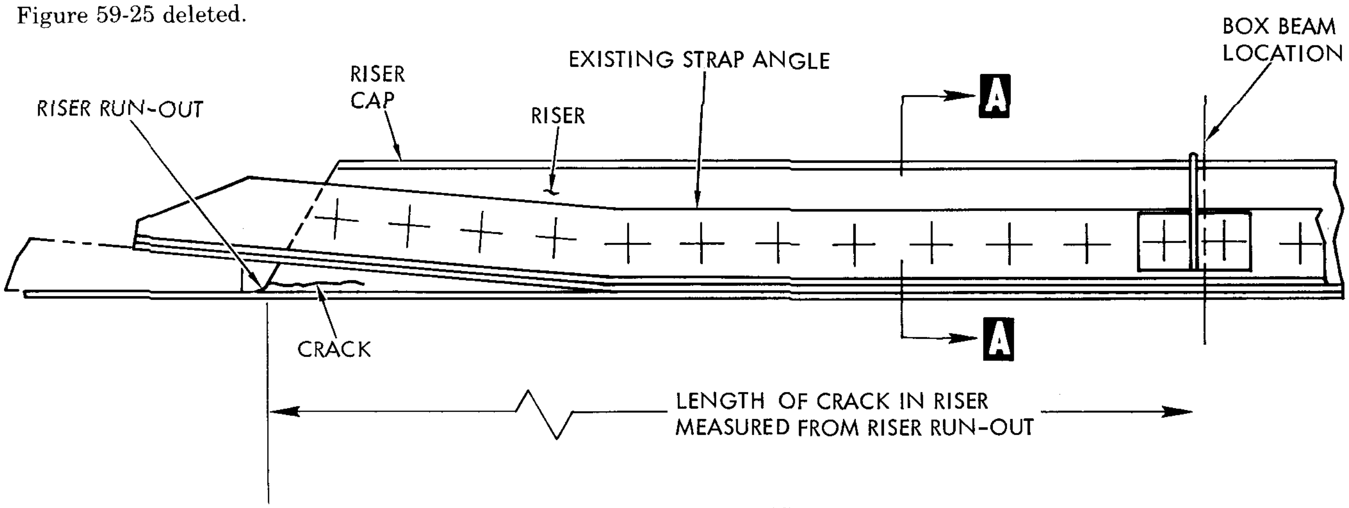
NOTE

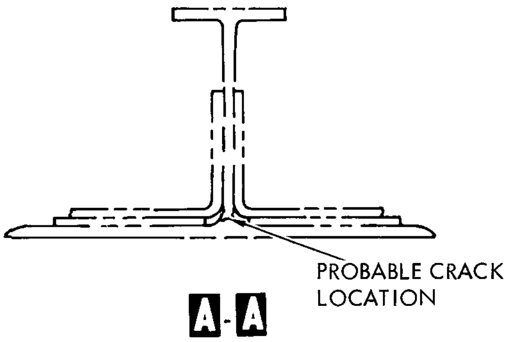
IF CRACKS EXIST AND HAVE PROGRESSED BEYOND THE POINT INDICATED IN THIS FIGURE AND INTO THE RADIUS OF THE BASE OF THE JW290 FITTING, CONTACT LOCKHEED WESTERN EXPORT COMPANY (SEE PAGE INTRODUCTION-3).

FITTINGS INSTALLED BY SERVICE BULLETIN 329-190B WHICH CONTAIN CRACKS MUST BE REPLACED. OBSERVE GAP LIMITS BETWEEN LOWER SURFACE BAR FITTING AND REPLACEMENT JW290-( ) FITTING.

Figure 59-24. Wing Lower Root Rib Shear Fitting Repair (LH Shown RH Opposite)

Figure 59-25 deleted.





NOTE

1. ALL AVAILABLE ENGINEERING DATA SHOWN IN TABLES A & B HAS BEEN FAA APPROVED.

△2 IF RISER IS CRACKED AT ANY LOCATION OR IS OF ANY LENGTH, THE FULL LENGTH RISER REPAIR SHALL BE ACCOMPLISHED.

△3 REPAIR IS GOOD FOR RISER CRACKED BETWEEN DIMENSION SHOWN. FIRST DIMENSION INDICATES DISTANCE FROM INBOARD RISER RUNOUT.

STEP 1

DETERMINE RISER NUMBER FROM FIGURE 58.2-8-5/2 AND VERIFY THE LENGTH OF CRACK FROM POINT OF RISER RUN-OUT AS SHOWN ABOVE.

STEP 2

FIND LENGTH OF CRACK IN TABLE A OR B AND NOTE CORRESPONDING ENGINEERING REPAIR INSTALLATION NUMBER FOR THAT LENGTH CRACK.

STEP 3

RELAY INFORMATION TO LOCKHEED GEORGIA JET-STAR CUSTOMER SERVICE DEPARTMENT TO DETERMINE IF THIS REPAIR KIT IS AVAILABLE.

STEP 4

IF CRACK LENGTH IS NOT SHOWN IN TABLE A OR B OR IS NOT SUITABLE, (IE: SOME REPAIR KITS MAY BE LONGER THAN REQUIRED TO SUIT CRACK LENGTH), CONTACT LOCKHEED WESTERN EXPORT COMPANY (SEE PAGE INTRODUCTION-3) FOR ASSISTANCE.

WARNING

IN SOME INSTANCES, CRACKS HAVE BEEN FOUND IN RISER/SKIN RADIUS ON ONE SIDE THAT PENETRATES THE SKIN OUTER SURFACE ON THE OTHER SIDE OF THE RISER. IN THIS CASE, CONTACT LOCKHEED WESTERN EXPORT COMPANY (SEE PAGE INTRODUCTION-3) FOR REPAIR INFORMATION.

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Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 1 of 6)

TABLE A. INBOARD RISER REPAIR KIT INDEX

RISER NUMBER	LENGTH OF CRACK FROM INBOARD RISER RUNOUT	REPAIR INSTALLATION KIT IDENTITY	REMARKS
2	Full length	JWK21-502 L/R	△2
3	Up to 14.0 in.	JWK21-503 L/R	
3	Up to 18.0 in.	JWK21-603 L/R	
3	Between 19.0 and 27.0 in.	JWK21-703 L/R	△3
3	Between 19.0 and 31.0 in.	JWK21-803 L/R	△3
4	Up to 13.0 in.	JWK21-504 L/R	
4	Up to 19.0 in.	JWK21-604 L/R	
4	Up to 33.0 in.	JWK21-704 L/R	
5	Up to 15.0 in.	JWK21-505 L/R	
5	Up to 34.0 in.	JWK21-605 L/R	
5	Up to 54.16 in.	JWK21-705 L/R	
6	Up to 17.0 in.	JWK21-506 L/R	
6	Between 38.0 and 46.0 in.	JWK21-606 L/R	△3
7	Up to 20.0 in.	JWK21-507 L/R	
7	Between 30.0 and 37.0 in.	JWK21-707 L/R	△3
7	7.0 in.	JWK21-807 L/R	
7	37.0 in.	JWK21-907 L/R	
8	Up to 22.0 in.	JWK21-508 L/R	
8	Up to 29.0 in.	JWK21-608 L/R	
10	Up to 13.88 in.	JWK21-510 L/R	
10	Up to 26.38 in.	JWK21-610 L/R	
11	16.0 in.	JWK21-511 L/R	
11	26.28 in.	JWK21-611 L/R	
12	11.06 in.	JWK21-512 L/R	
12	16.75 in.	JWK21-612 L/R	
13	13.24 in.	JWK21-513 L/R	
13	17.49 in.	JWK21-613 L/R	
13	35.30 in.	JWK21-713 L/R	
14	14.10 in.	JWK21-514 L/R	

Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 2 of 6)

TABLE A. INBOARD RISER REPAIR KIT INDEX (Cont)

RISER NUMBER	LENGTH OF CRACK FROM INBOARD RISER RUNOUT	REPAIR INSTALLATION KIT IDENTITY	REMARKS
14	33.05 in.	JWK21-714 L/R	
14	33.05 in	JWK21-914 L/R	
14	16.6 in.	JWK21-1014 L/R	
15	16.5 in.	JWK21-515 L/R	
15	18.1 in.	JWK21-715 L/R	
15	34.8 in.	JWK21-815 L/R	
16	18.13 in.	JWK21-516 L/R	
17	10.36 in.	JWK21-517 L/R	
17	19.8 in.	JWK21-617 L/R	
18	11.15 in.	JWK21-518 L/R	
18	22.35 in.	JWK21-618 L/R	
19	11.98 in.	JWK21-519 L/R	
19	24.45 in.	JWK21-619 L/R	
22	13.66 in.	JWK23-20 L/R	
22	Full length	JWK23-21 L/R	
22	17.0 in.	JWK23-22 L/R	
23	12.30 in.	JWK23-23 L/R	
23	28.0 in.	JWK23-523 L/R	
23	22.09 in.	JWK23-623 L/R	
24	20.80 in.	JWK23-24 L/R	
25	23.25 in.	JWK23-25 L/R	
26	15.20 in.	JWK23-26 L/R	
27	13.40 in.	JWK23-27 L/R	Airplanes 5001-5101
27	33.73 in.	JWK23-527 L/R	
27	17.69 in.	JWK23-627 L/R	Airplanes 5001-5101
28	19.33 in.	JWK23-28 L/R	
28	3.80 in.	JWK23-828 L/R	
28	21.80 to 72.10 in.	JWK23-928 L/R	△3

Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 3 of 6)

TABLE A. INBOARD RISER REPAIR KIT INDEX (Cont)

RISER NUMBER	LENGTH OF CRACK FROM INBOARD RISER RUNOUT	REPAIR INSTALLATION KIT IDENTITY	REMARKS
29	17.56 in.	JWK23-29 L/R	
29	115.58 to 127.73 in.	JWK23-629 L/R	△3
29	23.8 in.	JWK23-729 L/R	
30	12.96 in.	JWK23-30 L/R	
30	26.56 in.	JWK23-66 L/R	
30	132.95 to 145.10 in.	JWK23-630 L/R	△3
30	91.00 to 106.35 in.	JWK23-730 L/R	△3
31	24.70 in.	JWK23-31 L/R	
33	20.60 in.	JWK23-33 L/R	
33	10.00 in.	JWK23-633 L/R	
34	18.65 in.	JWK23-34 L/R	
34	28.25 to 38.95 in.	JWK23-634 L/R	△3
35	16.70 in.	JWK23-35 L/R	
35	25.65 to 37.35 in.	JWK23-535 L/R	△3
35	82.70 to 94.95 in.	JWK23-635 L/R	△3
36	21.00 in.	JWK23-36 L/R	
36	55.70 in.	JWK23-636 L/R	
36	Up to 16.05 in. and 38.25 to 62.80 in.	JWK23-736 L/R	△3
36	33.36 in.	JWK23-836 L/R	
37	19.60 in.	JWK23-37 L/R	
37	32.6 in.	JWK23-637 L/R	
38	17.40 in.	JWK23-38 L/R	
		JWK23-638 L	Adds 2.5 inches to previously installed -38L repair.

Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 4 of 6)

TABLE B. OUTBOARD RISER REPAIR KIT INDEX

RISER NUMBER	LENGTH OF CRACK FROM OUTBOARD RISER RUNOUT	REPAIR INSTALLATION KIT IDENTITY	REMARKS
2	Full Length	JWK21-502 L/R	Also in Table A. △2
3	13.25 in.	JWK22-503 L/R	
4	13.80 in.	JWK22-504 L/R	
5	16.55 in.	JWK22-505 L/R	
5	22.33 in.	JWK22-605 L/R	
6	28.19 in.	JWK22-506 L/R	
13	9.90 in.	JWK22-513 L/R	
13	13.20 in.	JWK22-613 L/R	
13	21.42 in.	JWK22-713 L/R	
14	18.50 in.	JWK22-514 L/R	
14	3.70 in.	JWK22-614 L/R	
14	6.7 in.	JWK22-714 L/R	
14	8.0 in.	JWK22-814 L/R	
15	8.00 in.	JWK22-515 L/R	
15	12.40 in.	JWK22-615 L/R	
15	4.24 in	JWK22-715 L/R	
16	7.60 in.	JWK22-516 L/R	
16	16.90 in.	JWK22-616 L/R	
16	8.10 in.	JWK22-716 L/R	
17	12.20 in.	JWK22-517 L/R	
17	13.75 in.	JWK22-617 L/R	
17	18.00 in.	JWK22-717 L/R	
18	16.00 in.	JWK22-518 L/R	
18	24.70 in.	JWK22-618 L/R	
19	13.30 in.	JWK22-519 L/R	
19	Full Length	JWK22-619 L/R	
23	14.00 in.	JWK22-523 L/R	Airplanes 5001-5101
23	14.10 in.	JWK22-623 L/R	Airplanes 5102-5200
23	30.78 in.	JWK22-723 L/R	

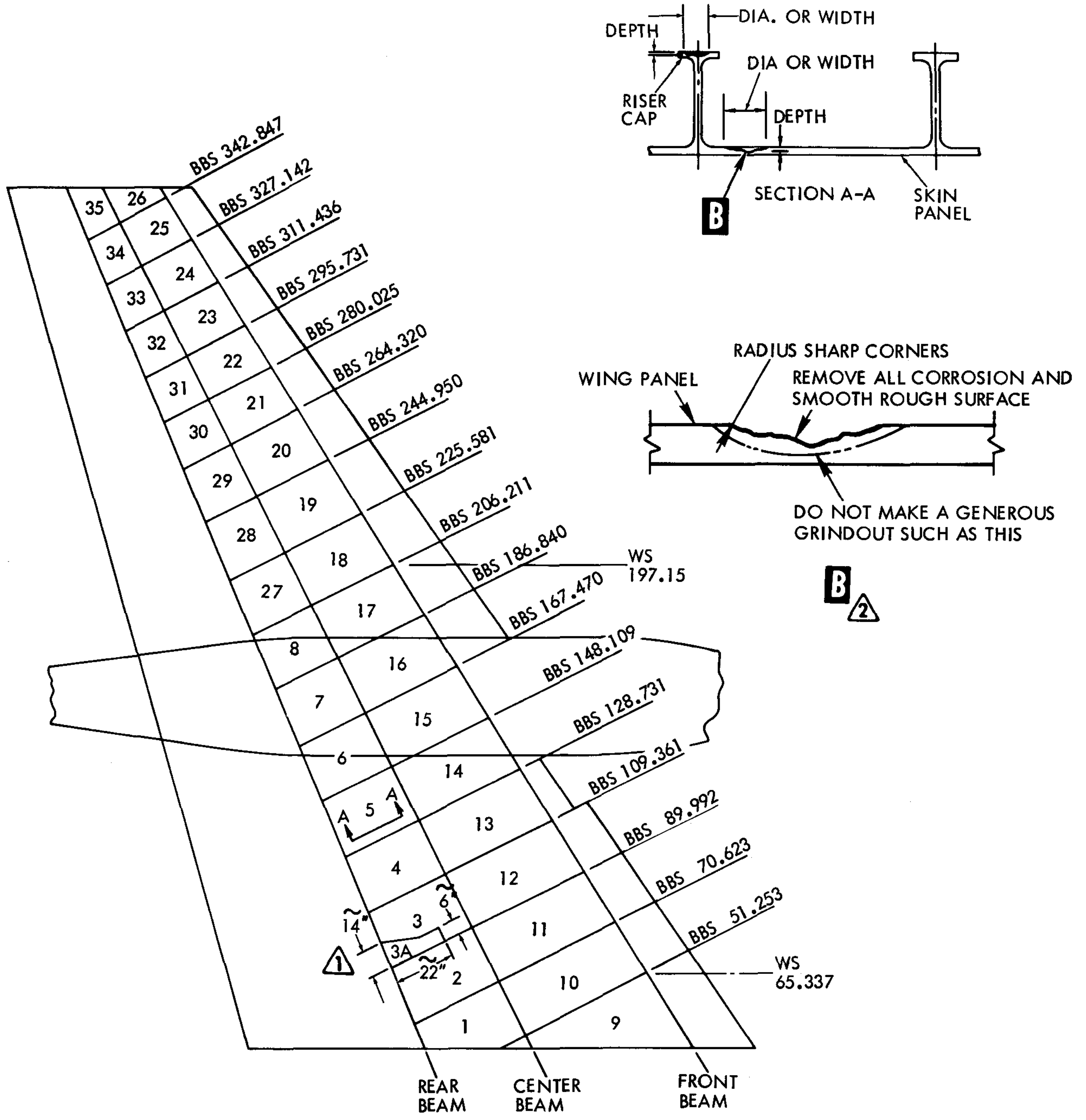
Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 5 of 6)

TABLE B. OUTBOARD RISER REPAIR KIT INDEX (Cont)

RISER NUMBER	LENGTH OF CRACK FROM OUTBOARD RISER RUNOUT	REPAIR INSTALLATION KIT IDENTITY	REMARKS
24	8.20 in.	JWK22-524 L/R	
24	27.60 in.	JWK22-624 L/R	
25	20.00 in.	JWK22-525 L/R	
25	26.95 to 35.95 in.	JWK22-625 L/R	
25	38.00 in.	JWK22-725 L/R	
26	11.00 in.	JWK22-526 L/R	Airplanes 5001-5101
26	20.30 in.	JWK22-626 L/R	Airplanes 5001-5101
26	26.90 in.	JWK22-726 L/R	Airplanes 5001-5101
27	13.50 in.	JWK22-527 L/R	
27	26.60 in.	JWK22-627 L/R	
27	51.00 in.	JWK22-727 L/R	
27	13.50 in.	JWK22-927 L/R	Airplanes 5102-5162
33	BBS 159.85 to 175.95 in. and BBS 197.65 to 214.75 in.	JWK22-533 L/R	
36	18.50 in.	JWK22-536 L/R	
37	24.60 in.	JWK22-537 L/R	Airplanes 5001-5101
37	12.60 in.	JWK22-637 L/R	Airplanes 5001-5101
37		JWK22-737 L/R	
37	24.60 in.	JWK22-837 L/R	
37	24.60 in.	JWK22-937 L/R	Airplanes 5102-5162

Figure 59-26. Wing Integrally Stiffened Skin Panel Riser Repair (No Skin Damage) (Sheet 6 of 6)

Pages 59-67 thru 59-70, including figures 59-27 and 59-28 deleted.



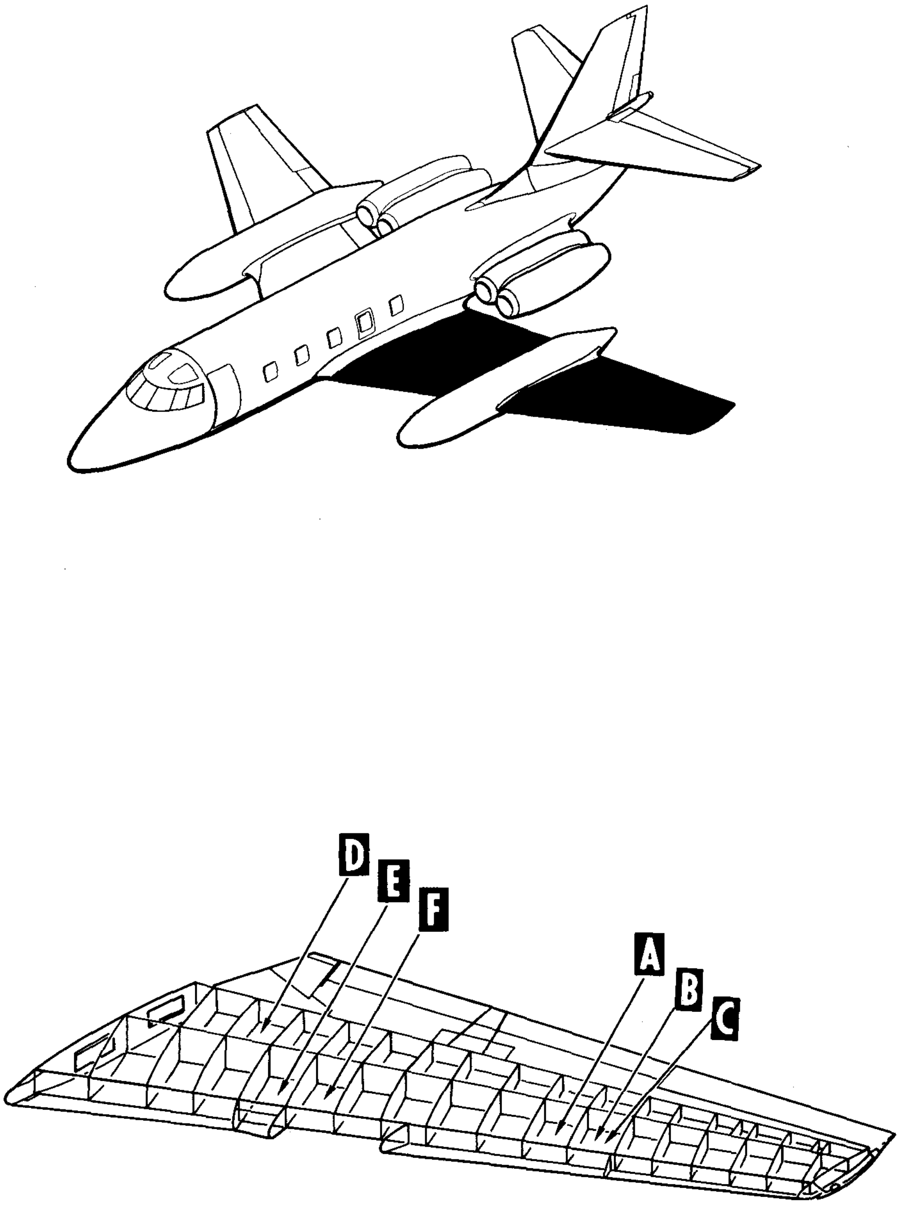
NOTE

△1 BAY NO. 3A DESIGNATES THE THICK SKIN AREA OF BAY NO. 3

△2 USE DENTIST DRILL TYPE TOOL WITH FINE ROTARY FILE, FINE STONE, OR RUBBER ABRASIVE WHEEL. CLEAN UP DEBRIS.

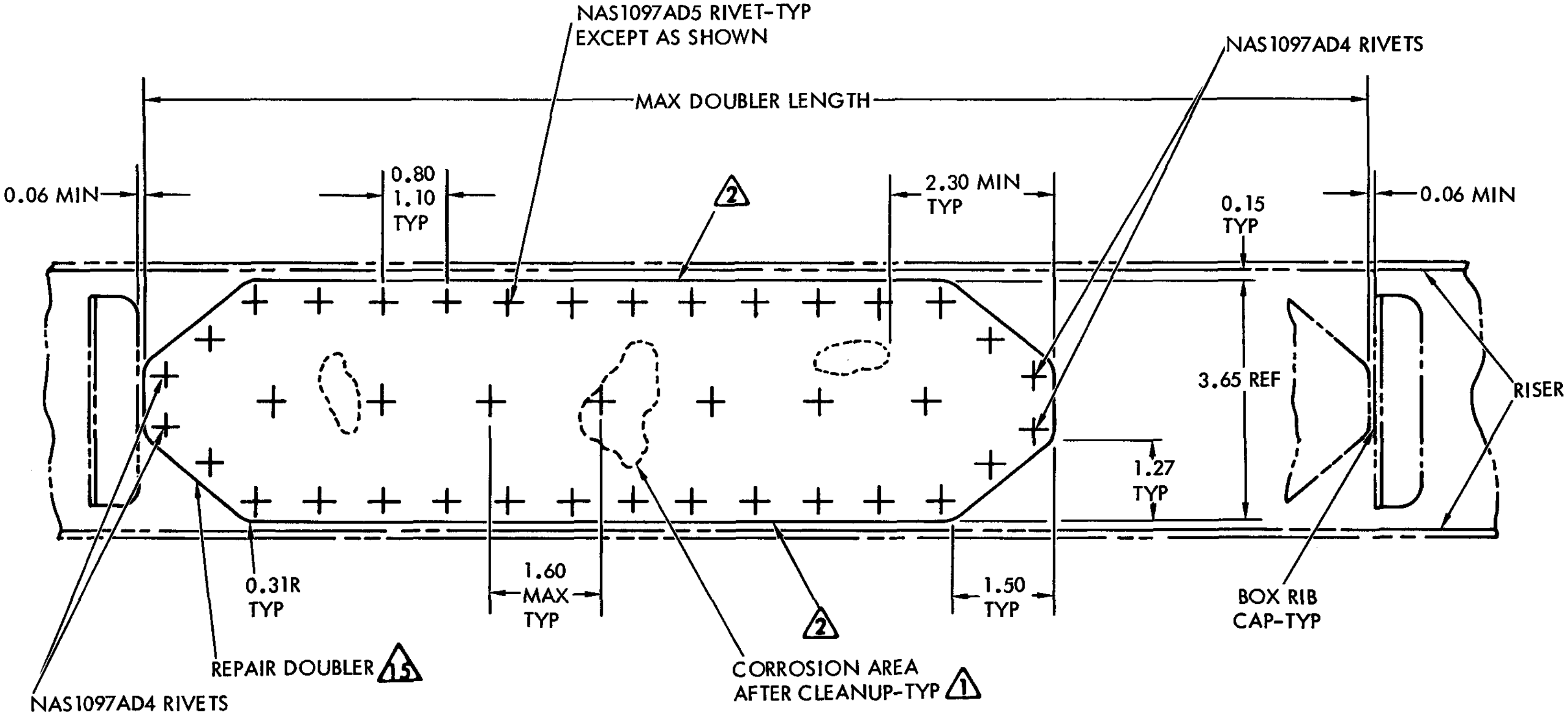
JS-1-2-X0/2-5928A

Figure 59-28A. Basic Wing Layout



JS-1-2-XO/2-5928B-1

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 1 of 12)



A

TYPICAL LOWER SURFACE REPAIR

JS-1-2-X0/2-5928B-2

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair
(Sheet 2 of 12)

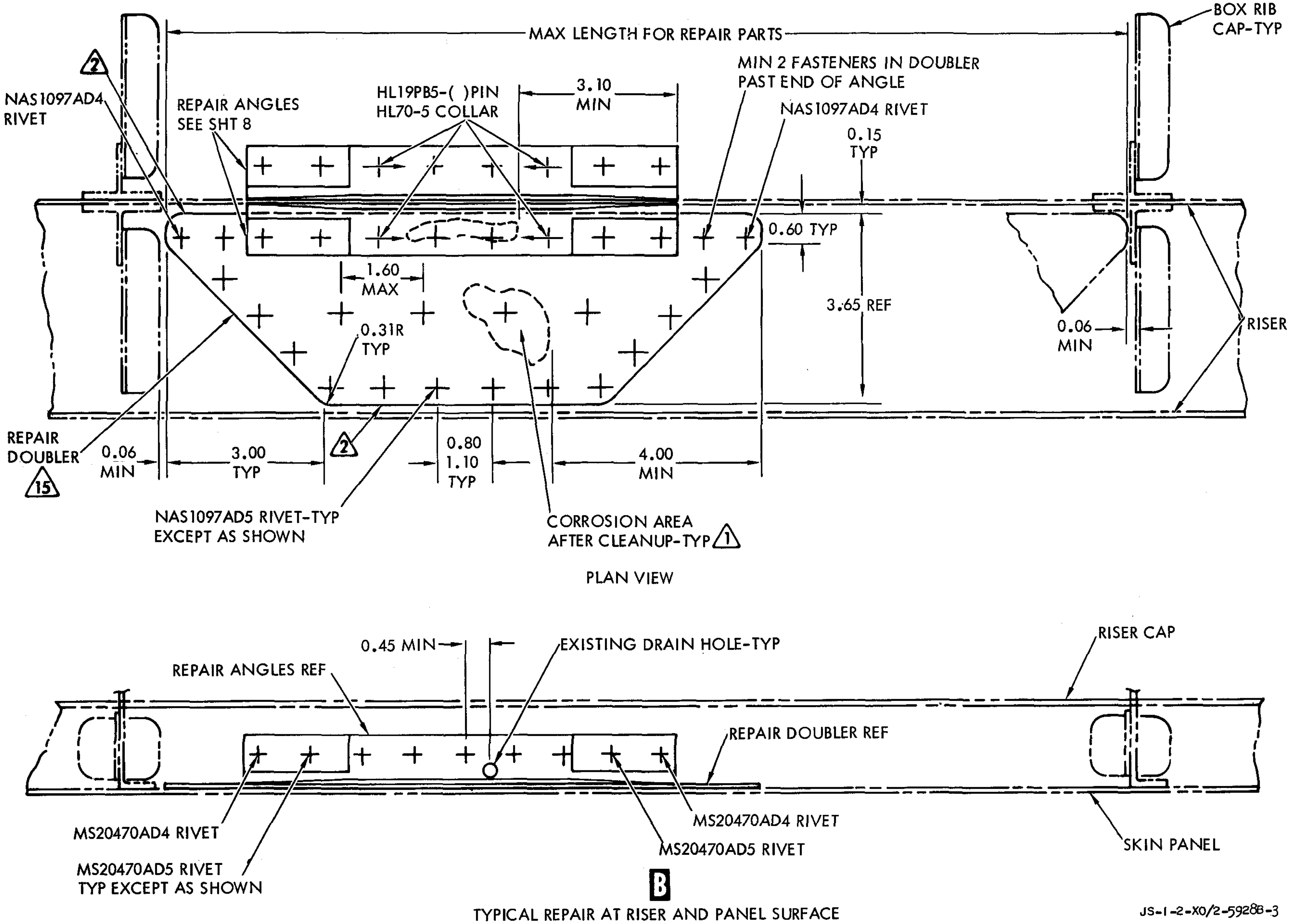


Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 3 of 12)

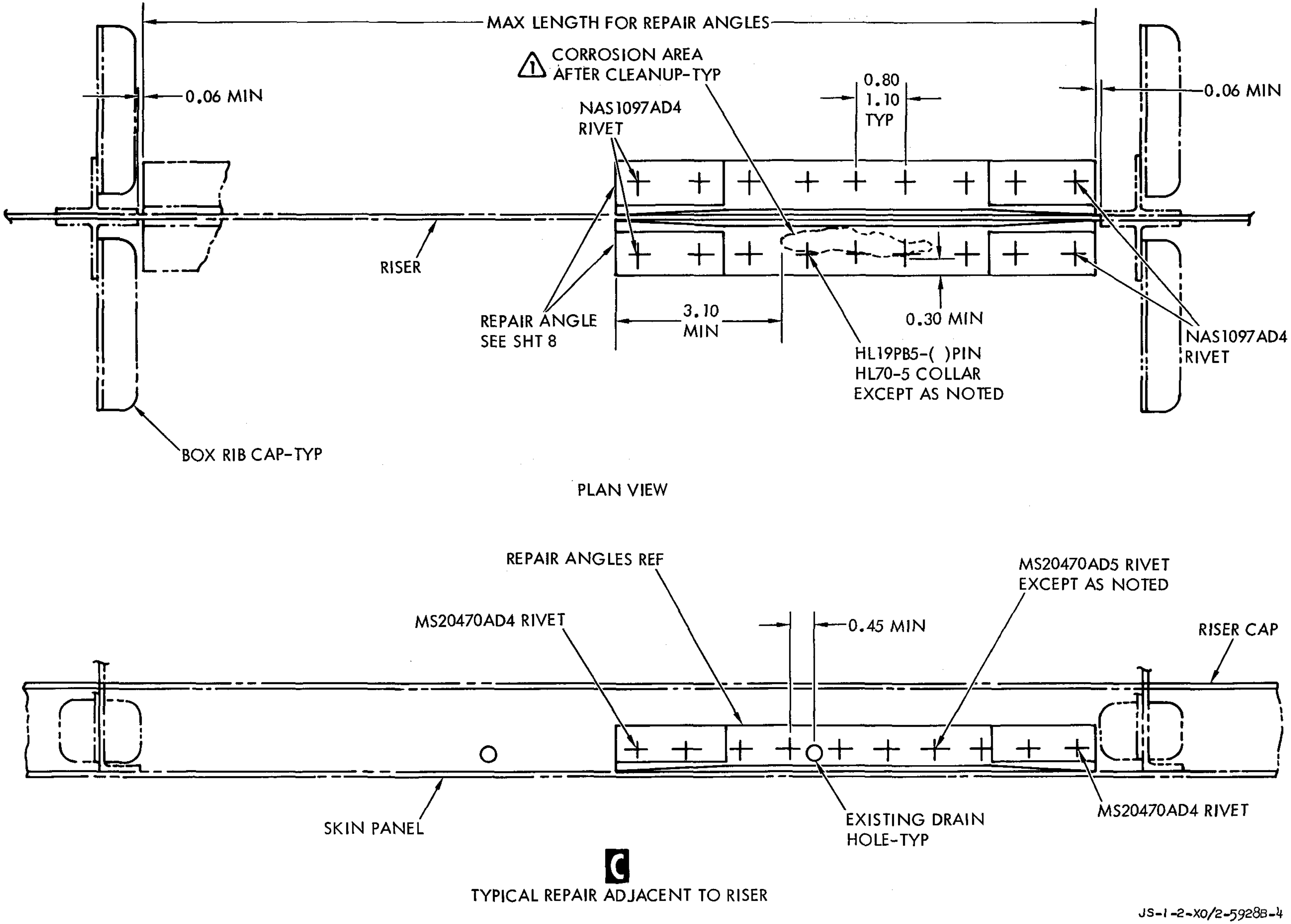


Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 4 of 12)

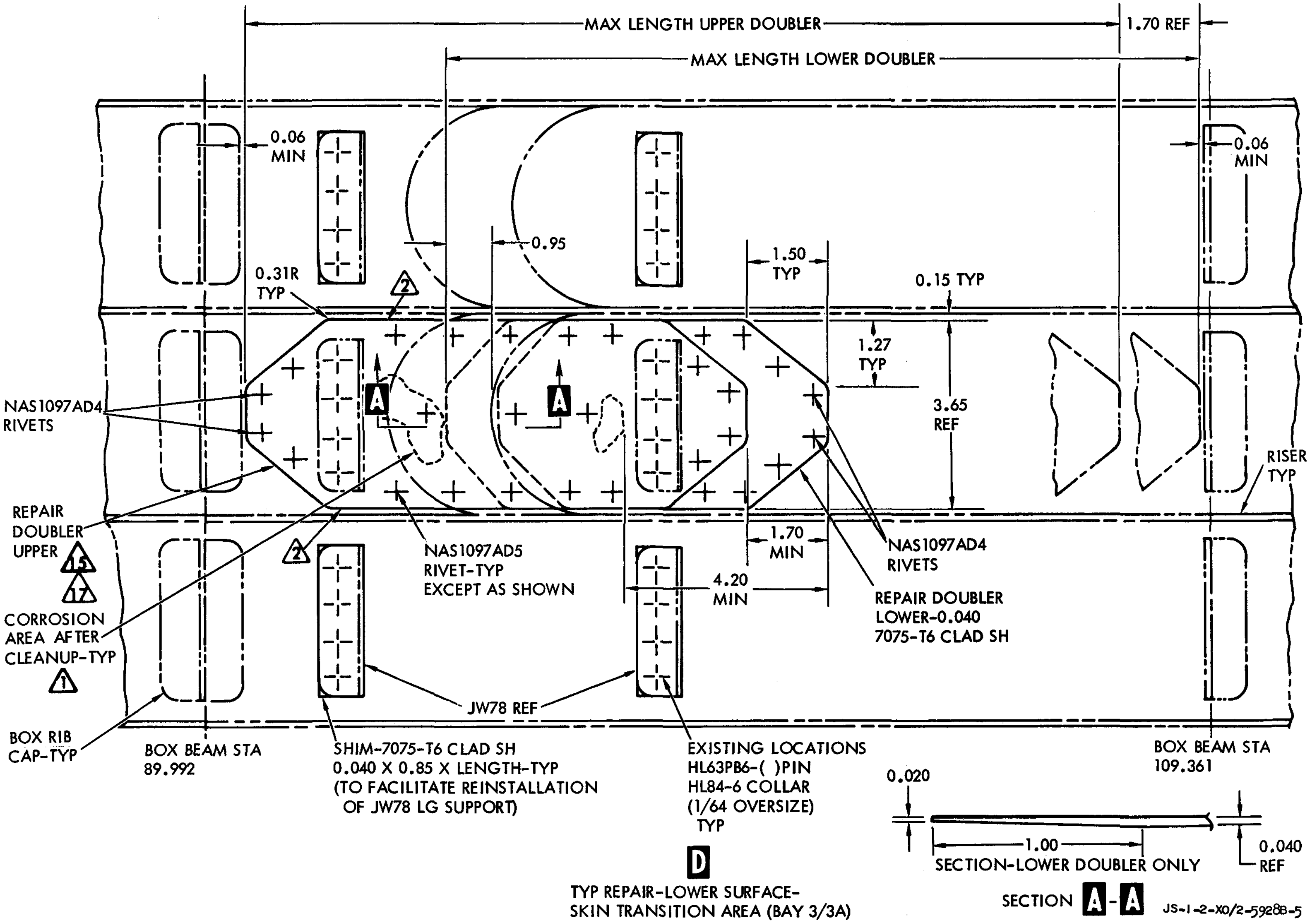

MAX LENGTH UPPER DOUBLER
1.70 REF
MAX LENGTH LOWER DOUBLER
0.06 MIN
0.06 MIN
0.95
0.31R TYP
2
1.50 TYP
0.15 TYP
1.27 TYP
3.65 REF
A
A
NAS1097AD4 RIVETS
RISER TYP
REPAIR DOUBLER UPPER
15
17
2
1.70 MIN
NAS1097AD5 RIVET-TYP EXCEPT AS SHOWN
4.20 MIN
NAS1097AD4 RIVETS
REPAIR DOUBLER LOWER-0.040 7075-T6 CLAD SH
CORROSION AREA AFTER CLEANUP-TYP
1
JW78 REF
BOX RIB CAP-TYP
BOX BEAM STA 89.992
SHIM-7075-T6 CLAD SH 0.040 X 0.85 X LENGTH-TYP (TO FACILITATE REINSTALLATION OF JW78 LG SUPPORT)
EXISTING LOCATIONS HL63PB6-( )PIN HL84-6 COLLAR (1/64 OVERSIZE) TYP
BOX BEAM STA 109.361
0.020
1.00
SECTION-LOWER DOUBLER ONLY
0.040 REF
D
TYP REPAIR-LOWER SURFACE-SKIN TRANSITION AREA (BAY 3/3A)
SECTION A-A
JS-1-2-X0/2-59288-5

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 5 of 12)

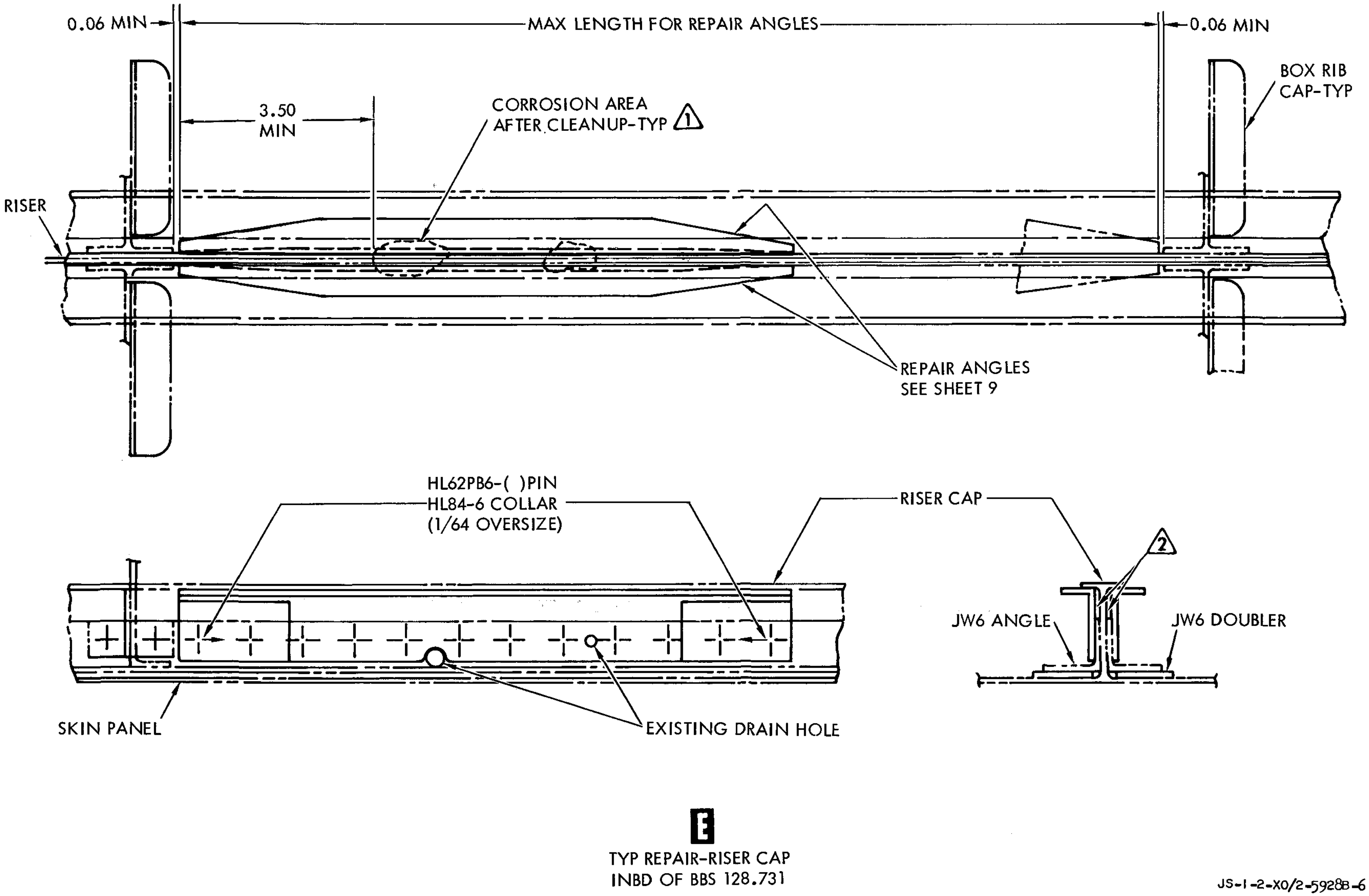


Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 6 of 12)

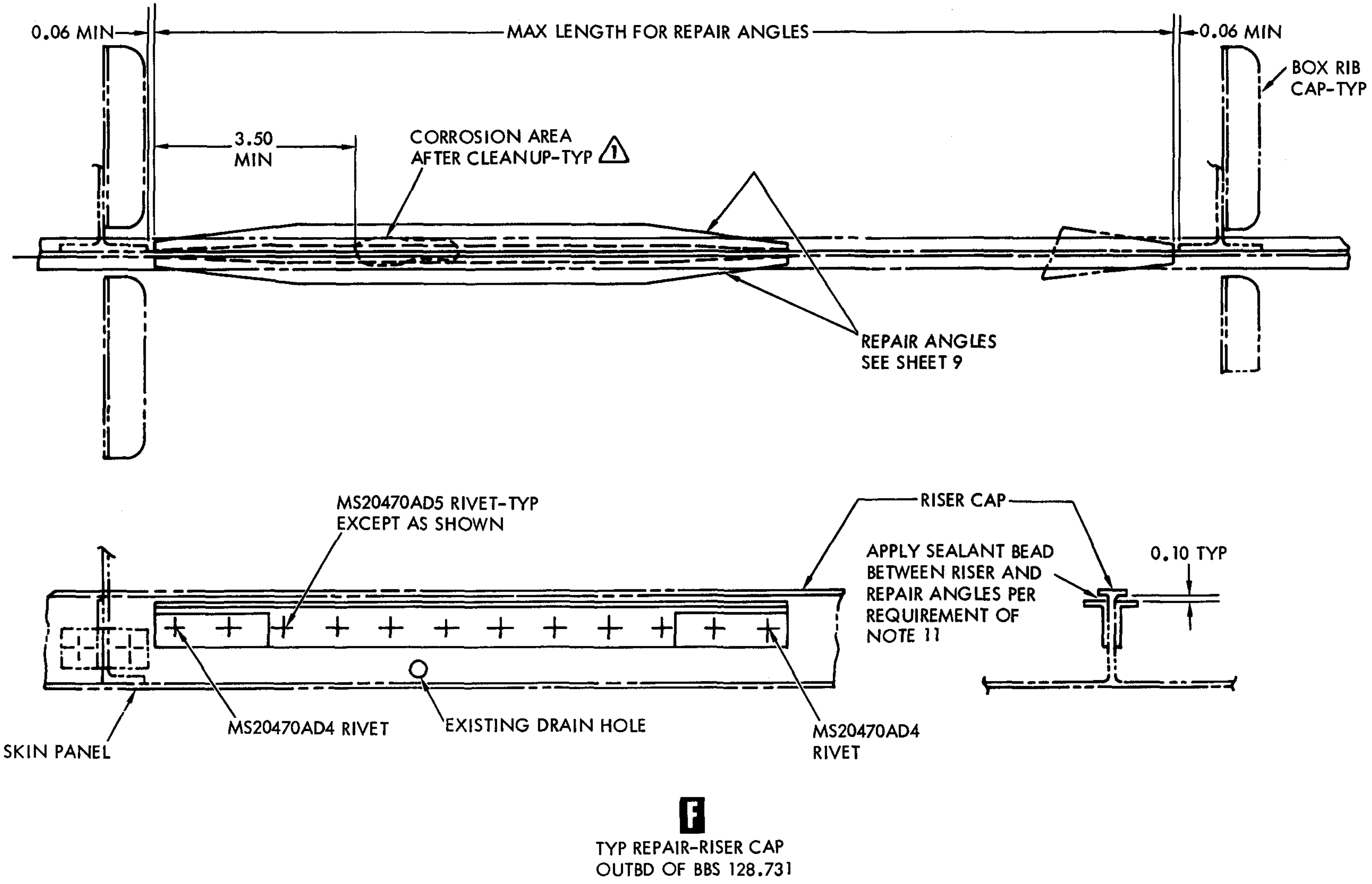
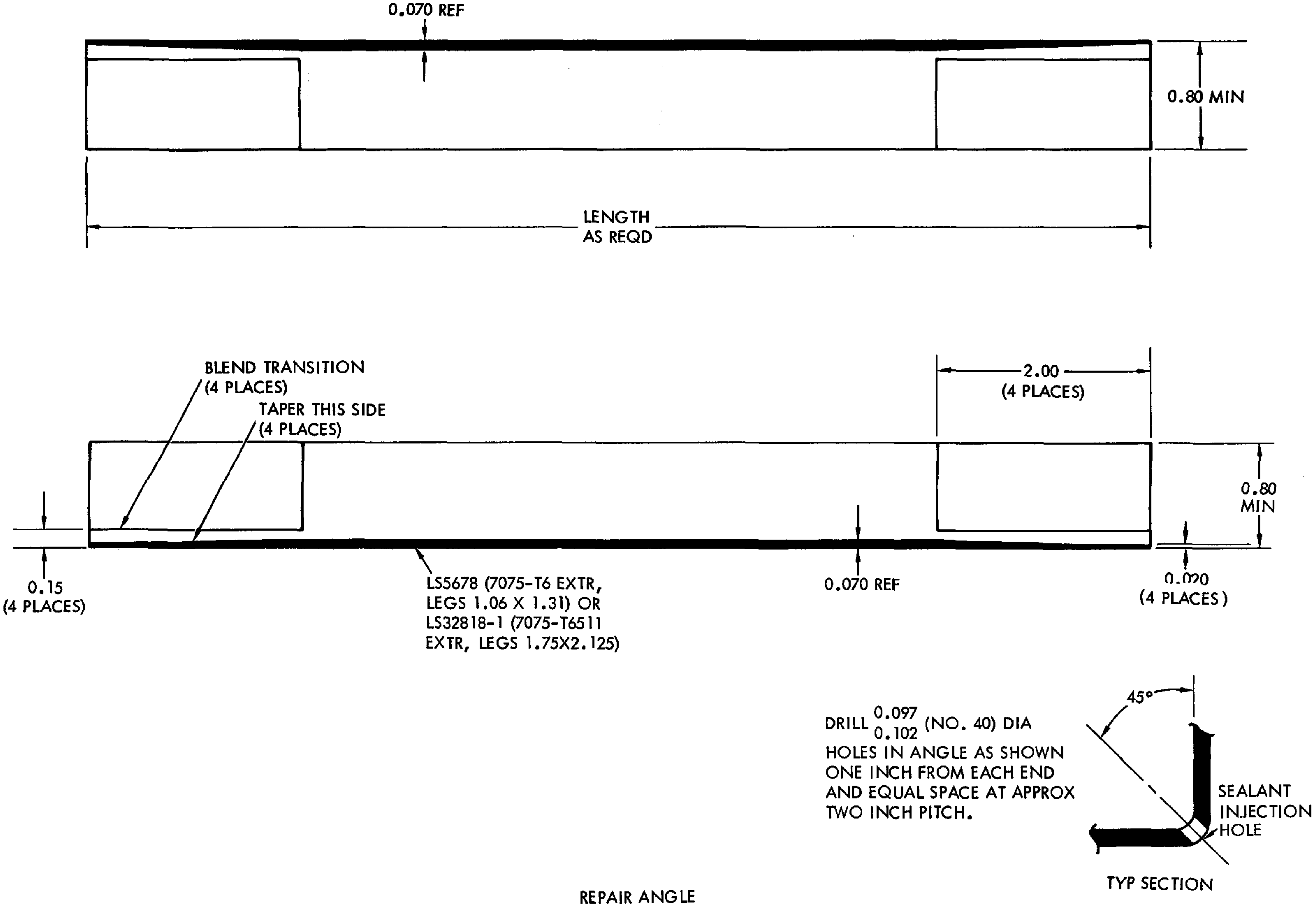

0.06 MIN
MAX LENGTH FOR REPAIR ANGLES
0.06 MIN
BOX RIB CAP-TYP
3.50 MIN
CORROSION AREA AFTER CLEANUP-TYP 1
REPAIR ANGLES SEE SHEET 9
MS20470AD5 RIVET-TYP EXCEPT AS SHOWN
RISER CAP
APPLY SEALANT BEAD BETWEEN RISER AND REPAIR ANGLES PER REQUIREMENT OF NOTE 11
0.10 TYP
SKIN PANEL
MS20470AD4 RIVET
EXISTING DRAIN HOLE
MS20470AD4 RIVET
F
TYP REPAIR-RISER CAP OUTBD OF BBS 128.731
JS-1-2-X0/2-5928B-7

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 7 of 12)

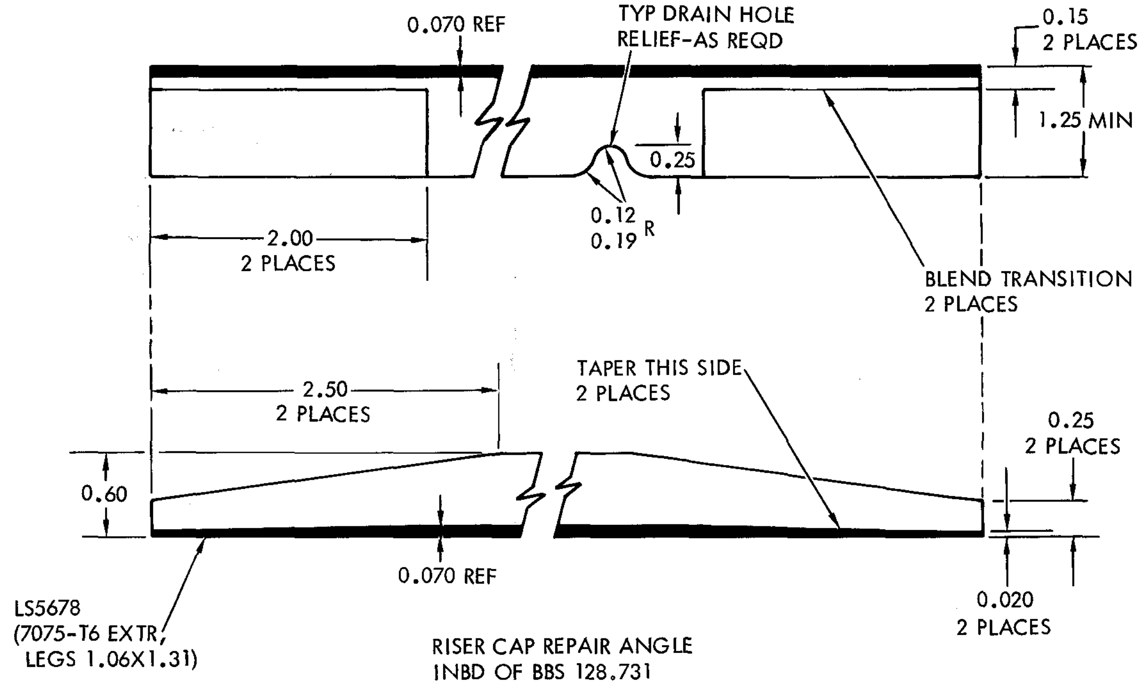

0.070 REF
0.80 MIN
LENGTH AS REQD
BLEND TRANSITION (4 PLACES)
TAPER THIS SIDE (4 PLACES)
2.00 (4 PLACES)
0.80 MIN
0.15 (4 PLACES)
LS5678 (7075-T6 EXTR, LEGS 1.06 X 1.31) OR LS32818-1 (7075-T6511 EXTR, LEGS 1.75X2.125)
0.070 REF
0.020 (4 PLACES)
45°
SEALANT INJECTION HOLE
TYP SECTION

DRILL $\frac{0.097}{0.102}$ (NO. 40) DIA HOLES IN ANGLE AS SHOWN ONE INCH FROM EACH END AND EQUAL SPACE AT APPROX TWO INCH PITCH.

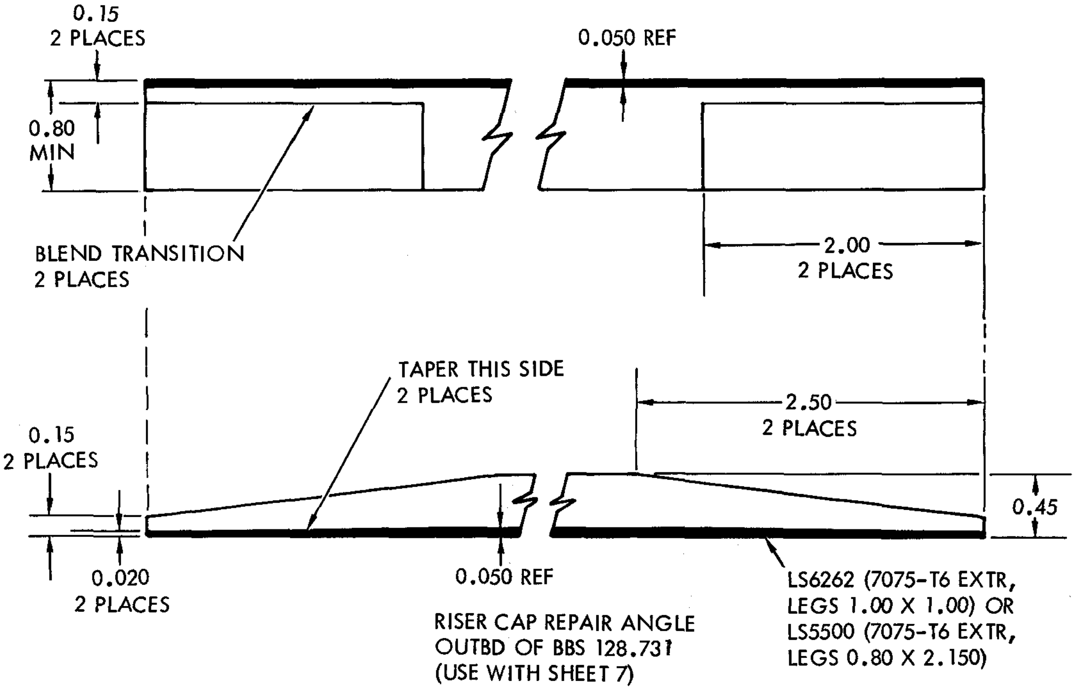
REPAIR ANGLE
(USE WITH SHEETS 3 AND 4)

JS-1-2-X0/2-5928B-8

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 8 of 12)

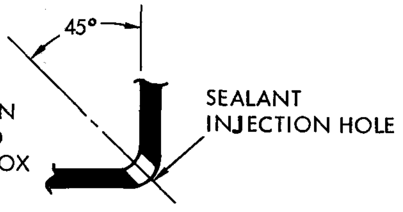


RISER CAP REPAIR ANGLE INBD OF BBS 128.731 (USE WITH SHEET 6)



RISER CAP REPAIR ANGLE OUTBD OF BBS 128.731 (USE WITH SHEET 7)

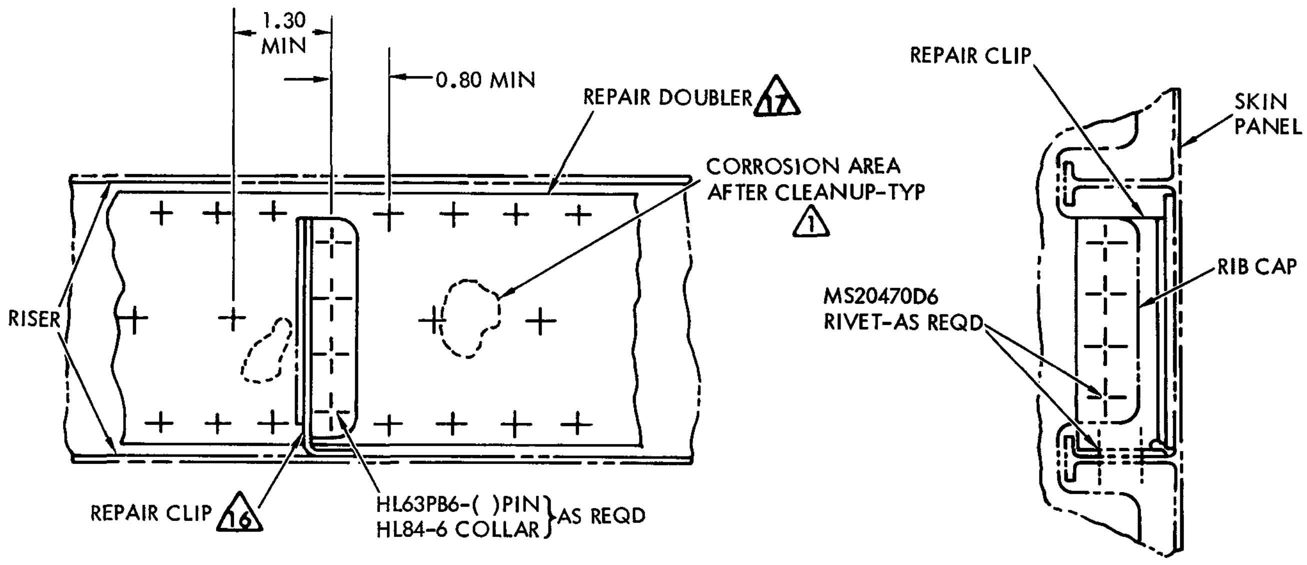
DRILL $^{0.097}_{0.102}$ (NO. 40) DIA HOLES IN ANGLE AS SHOWN ONE INCH FROM EACH END AND EQUAL SPACE AT APPROX TWO INCH PITCH.



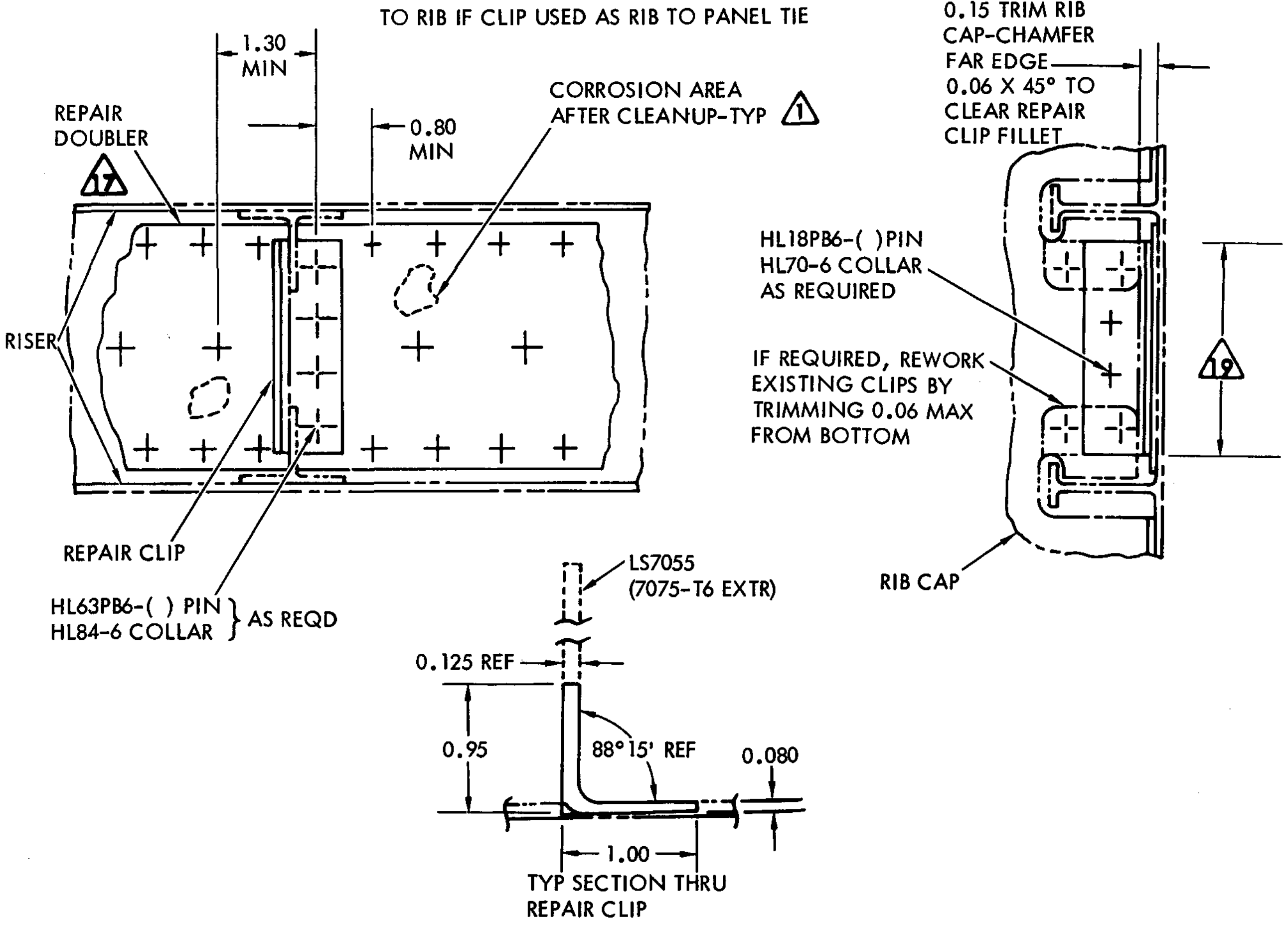
TYP SECTION

JS-1-2-X0/2-5928B-9

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 9 of 12)



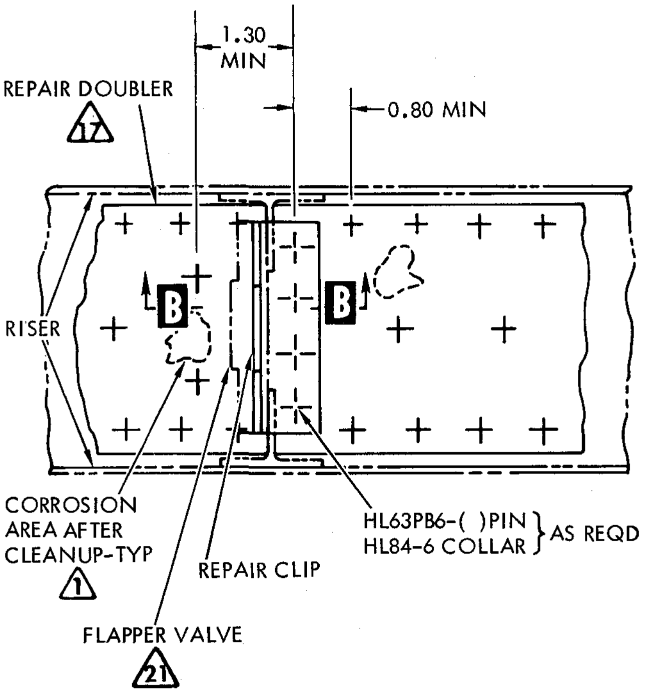
LOWER SURFACE REPAIR FOR DAMAGE ADJACENT TO RIB IF CLIP USED AS RIB TO PANEL TIE

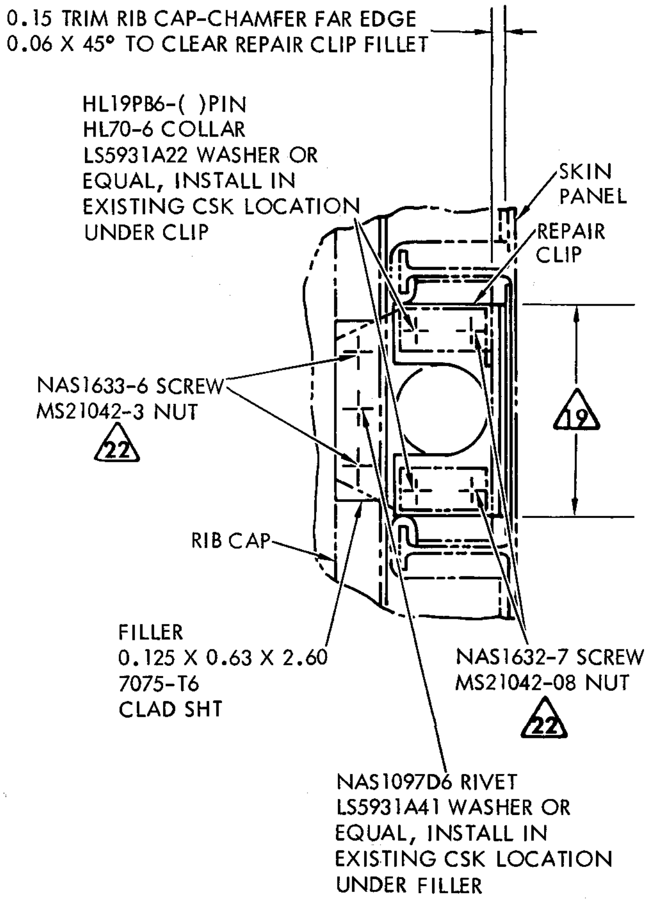

1.30 MIN
0.80 MIN
REPAIR DOUBLER 17
CORROSION AREA AFTER CLEANUP-TYP 1
RISER
REPAIR CLIP
HL63PB6-( ) PIN
HL84-6 COLLAR
AS REQD
0.15 TRIM RIB CAP-CHAMFER FAR EDGE 0.06 X 45° TO CLEAR REPAIR CLIP FILLET
HL18PB6-( )PIN
HL70-6 COLLAR
AS REQUIRED
IF REQUIRED, REWORK EXISTING CLIPS BY TRIMMING 0.06 MAX FROM BOTTOM
19
RIB CAP
LS7055 (7075-T6 EXTR)
0.125 REF
0.95
88° 15' REF
0.080
1.00
TYP SECTION THRU REPAIR CLIP

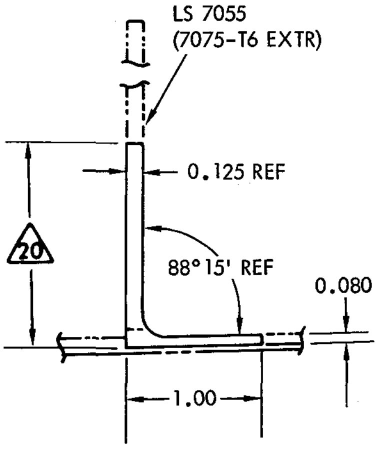
LOWER SURFACE REPAIR FOR DAMAGE ADJACENT TO RIB IF PANEL TIE IS RIB CAP INTEGRAL FOOT

JS-1-2-X0/2-59288-10

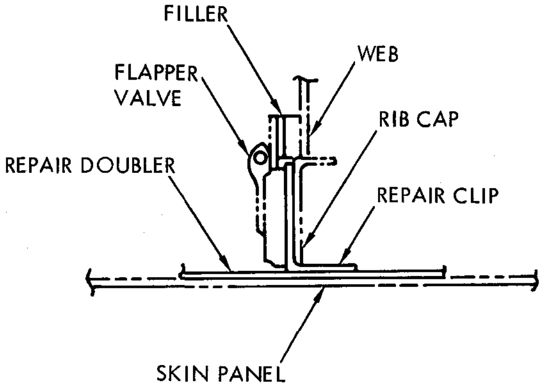
Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 10 of 12)




0.15 TRIM RIB CAP-CHAMFER FAR EDGE 0.06 X 45° TO CLEAR REPAIR CLIP FILLET
HL19PB6-( )PIN
HL70-6 COLLAR
LS5931A22 WASHER OR EQUAL, INSTALL IN EXISTING CSK LOCATION UNDER CLIP
SKIN PANEL
REPAIR CLIP
NAS1633-6 SCREW
MS21042-3 NUT
22
19
RIB CAP
FILLER
0.125 X 0.63 X 2.60
7075-T6
CLAD SHT
NAS1632-7 SCREW
MS21042-08 NUT
22
NAS1097D6 RIVET
LS5931A41 WASHER OR EQUAL, INSTALL IN EXISTING CSK LOCATION UNDER FILLER



TYP SECTION THRU REPAIR CLIP



SECTION B-B

LOWER SURFACE REPAIR FOR DAMAGE ADJACENT TO RIB AT FLAPPER VALVE LOCATION

JS-1-2-X0/2-5928B-11

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 11 of 12)

NOTE

△1 FILL FLUSH ALL GRIND-OUT VOIDS WITH EPOXY PUTTY 3M CO. EC 1751 OR EQUAL. TROWEL WITH SPATULA TO FORM A SMOOTH LEVEL SURFACE.

△2 FILL AREA FLUSH WITH EPOXY PUTTY 3M CO. EC1751 OR EQUAL.

3. MAXIMUM PERMISSIBLE GAP BETWEEN PANEL RISERS AND ANY SUPPORT ANGLE WITH SETUP SCREWS OR CLECOS INSTALLED THRU THE WING PANEL SURFACE AND SUPPORT ANGLE ONLY IS 0.005. WHEN GAP EXCEEDS 0.005, SHIM AS REQUIRED (MAX 0.060 OF 7075-T6 MATERIAL) TO ELIMINATE GAP BETWEEN RISER AND ANGLE ON INSTALLATION, DO NOT PULL UP WITH FASTENERS.

4. CENTERLESS SYMBOL (-¦-) INDICATES EXISTING FASTENER HOLE LOCATION.

5. ENSURE DRAIN HOLES ARE CLEAR AND FREE OF SEALANT/EPOXY.

6. REDRILL ALL EXISTING DRAIN HOLES COVERED BY REPAIR EXCEPT AS NOTED. FINISH HOLE EDGE PER SECTION 51.7.3.1.

7. INSTALL FASTENERS AND DRILL HOLES PER SECTION 59.1. UNLESS OTHERWISE NOTED, SPACE FASTENERS APPROXIMATELY AS SHOWN MAINTAINING EDGE DISTANCE OF 0.38 FOR 3/16 DIA, 0.31 FOR 5/32 DIA, AND 0.28 FOR 1/8 DIA FASTENERS. SEAL FASTENERS PER PARAGRAPH 51.7.3.4.

8. SQUEEZE RIVETING REQUIRED ON ALL INTEGRAL STIFFENERS (RISERS).

☞ 9. ALLOW EC 1751 EPOXY PUTTY TO CURE APPROXIMATELY 24 HOURS AT 70° F.

10. COAT ALL FAYING SURFACES OF DOUBLER AND ANGLES PER SECTION 28.2.3.4 WITH MIL-S-81733 TYPE II SEALANT. (SEE FIGURE 28-12.)

11. SEAL AND TOPCOAT ALL EDGES AND FASTENERS PER SECTION 28.2.3.4 USING ADHESION PROMOTER, MIL-S-8802 TYPE SEALANT AND TOPCOAT.

☞ 12. FABRICATE PER 59.1.11. FINISH ALL DETAILS BEFORE INSTALLATION WITH CHEMICAL FILM PROTECTION TREATMENT PER MIL-C-5541 AND TWO COATS OF POLYURETHANE PER MIL-C-27725.

13. FINISH ALL CORROSION DAMAGED AREAS WITH ALODINE AFTER CLEAN UP PER SECTION 51.7.3.1 AND FILL ALL REWORKED CORROSION PITS NOT REQUIRING STRUCTURAL REPAIR WITH ADHESION PROMOTER AND MIL-S-81733 SEALANT. CURE AND OVERCOAT PER 28.2.3.4.

14. PROTECT ADJACENT STRUCTURE DURING REWORK OF EXISTING INSTALLED PARTS. BREAK SHARP EDGES ON REWORKED PARTS AND FINISH PER SECTION 51.7.3.1.

△15 7075-T6 CLAD SHEET, REQUIRED THICKNESS SHOWN (BASED ON CHORD AREA REDUCTION CALCULATED PER SECTION 59.2.2.4.)

AREA REDUCTION (IN^2)	DOUBLER THICKNESS
UP TO 0.060	0.040
0.061 TO 0.085	0.050

WHEN CHORD AREA REDUCTION EXCEEDS 0.085 IN^2, ASSISTANCE OF AN FAA DESIGNATED ENGINEERING REPRESENTATIVE (DER) WILL BE REQUIRED.

△16 MAKE FROM 7075-0 CLAD SHEET, THICKNESS AND BEND RADIUS TO MATCH EXISTING PART. REDUCE HEIGHT BY DOUBLER THICKNESS, USE EXISTING CLIP AS TEMPLATE, HEAT TREAT TO T6 AFTER FORMING.

△17 FOR INFORMATION NOT SHOWN, SEE SHEET 2.

18. WHEN REPLACING EXISTING 3/16 INCH FASTENERS, IT IS PERMISSIBLE TO USE HL62PB/HL63PB PINS (1/64 OVERSIZE) AND HL84 COLLARS AS REQUIRED FOR ACCEPTABLE FIT IN EXISTING HOLE LOCATIONS.

△19 LENGTH TO BE SAME AS DELETED RIB CAP FOOT.

△20 HEIGHT TO BE DETERMINED BY DOUBLER THICKNESS. CLIP MUST CLEAR RADIUS ON RIB CAP. CUT-OUT TO MATCH EXISTING HOLE.

△21 324700 VALVE - REWORK IF REQUIRED BY TRIMMING 0.06 MAX FROM BOTTOM OF HOUSING.

△22 HOLES REQUIRED THRU REPAIR PARTS AT EXISTING LOCATIONS. FOR NAS 1633 SCREWS USE 0.193 - 0.199 (NO. 10) DIA. FOR NAS 1632 SCREWS USE 0.165 - 0.171 (NO. 19) DIA.

23. FILL SEALANT INJECTION HOLES IN REPAIR ANGLES PER SECTION 28.2.3.4.d.

JS-59-9-028B-12

Figure 59-28B. Typical Lower Surface Skin Panel Corrosion Repair (Sheet 12 of 12)

Bay	Area Reduction Permissible (Sq. Inches)	Bay	Area Reduction Permissible (Sq. Inches)	Bay	Area Reduction Permissible (Sq. Inches)
1	.039	6	.048	12	.019
2	.039	7	.047	13	.041
3	.003	8	.046	14	.031
3A	.019	9	.044	15	.030
4	.035	10	.044	16	.035
5	.049	11	.044	17	.036
				18 thru 35	.020

NOTE

THESE DATA NOT APPLICABLE TO 731 MOD JETSTARS.

NOTE

AREA REDUCTIONS DO NOT APPLY FOR 4.0 INCHES ON EACH SIDE OF THE SPANWISE SPLICE FASTENER CENTERLINE OR WITHIN 4.0 INCHES OF THE SPAR CAP FASTENERS. THE AREA REDUCTIONS DO NOT APPLY WHEN THE DEPTH OF GRINDOUT EXCEEDS 0.010 INCH WITHIN 3/8 INCH OF ANY RISER OR WHEN THE DEPTH OF GRINDOUT EXCEEDS 1/2 OF THE ORIGINAL PANEL THICKNESS.

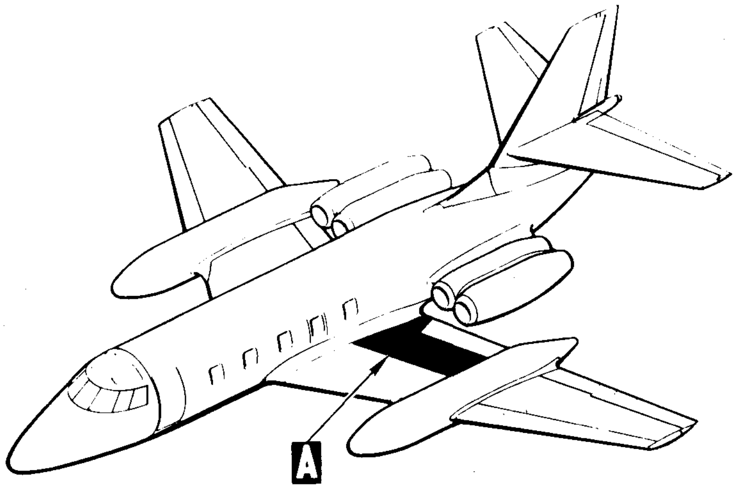
Figure 59-28C. Wing Lower Surface Skin Panel Corrosion Grindout Limits

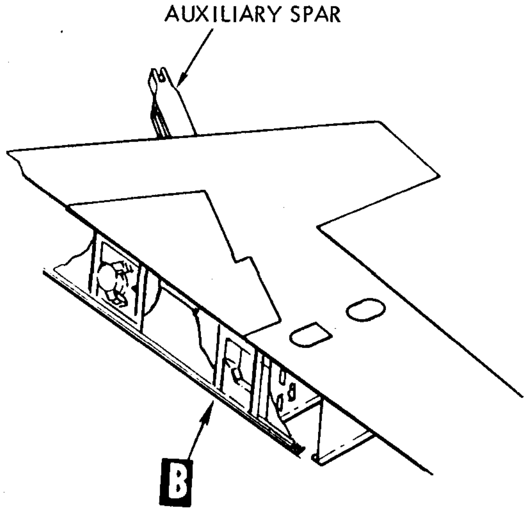
Bay	Area Reduction Permissible (Sq. Inches)	Bay	Area Reduction Permissible (Sq. Inches)	Bay	Area Reduction Permissible (Sq. Inches)
1	.008	7	.006	14	.006
2	.008	8	.005	15	.006
3	.003	9	.008	16	.005
3A	.007	10	.008	17	.005
4	.007	11	.007	18 thru 35	.003
5	.007	12	.007		
6	.006	13	.006		

NOTE

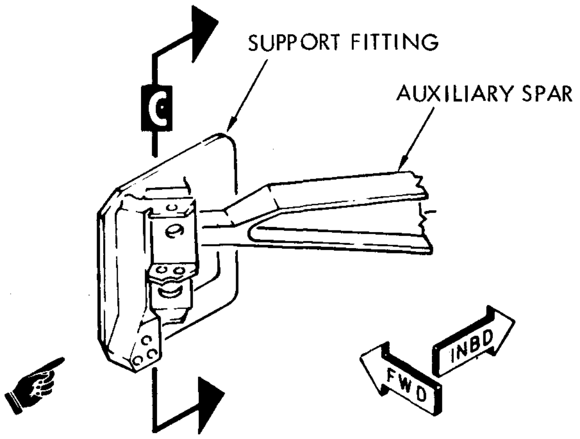
THESE DATA NOT APPLICABLE TO 731 MOD JETSTARS.

Figure 59-28D. Wing Lower Surface Panel Riser Cap Corrosion Grindout Limits

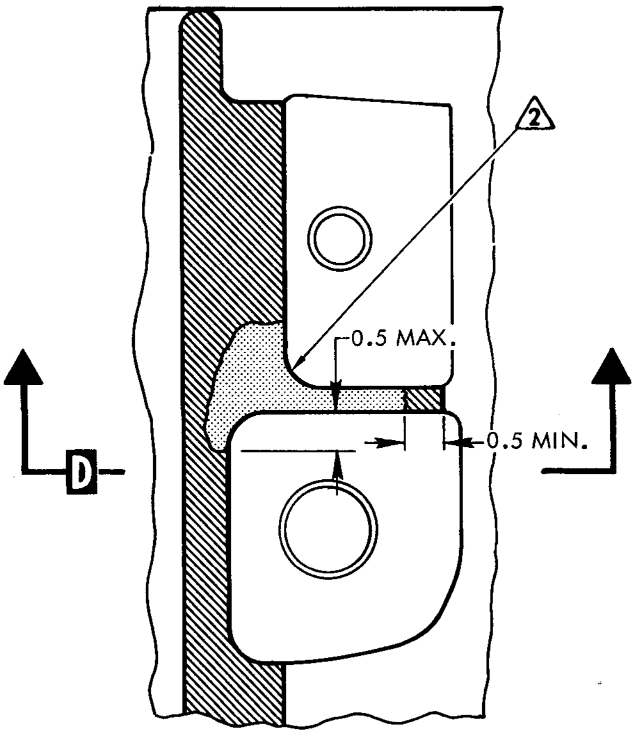




A
TOP SURFACE



B



SECTION C 1

STEP 1

GAIN ACCESS TO THE LANDING GEAR ACTUATOR AND AUXILIARY SPAR OUTBOARD SUPPORT FITTING (DETAIL B)

NOTE

1 DETAIL C IS A SECTION THROUGH THE SLOTS SHOWING TOTAL PERMISSABLE DAMAGE TO THE JW 60 FITTING.

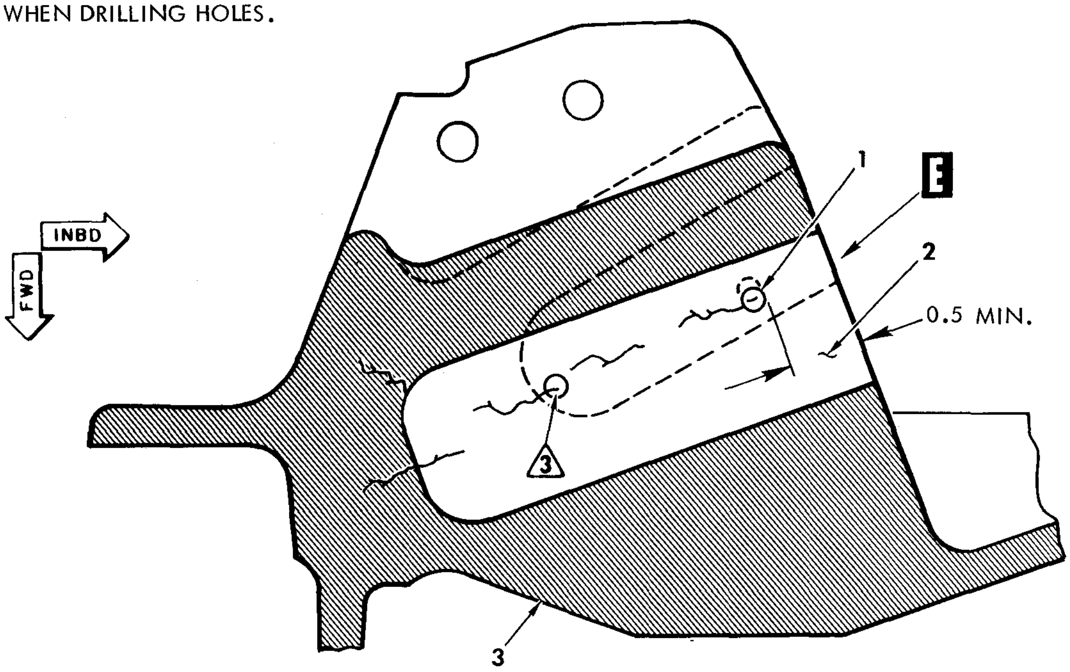
2 CRACKS IN THIS AREA CANNOT BE SEEN DUE TO THE AUXILIARY SPAR INSTALLATION.

JS-1-2-X0/6-5928E-1

Figure 59-28E. Landing Gear Actuator and Auxiliary Spar Outboard Support Fitting (JW60) Repair (Sheet 1 of 2)

STEP 2

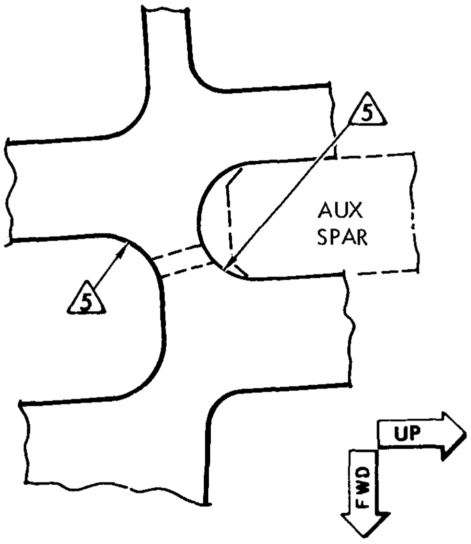
STOP DRILL A NO. 20 HOLE AT THE END OF THE MOST INBOARD CRACK (1). IF CRACK PROPAGATES INBOARD, CONTINUE STOP DRILLING UNTIL A 0.5 MIN. EDGE DISTANCE (2) IS REACHED. IF CRACK EXTENDS FURTHER, REMOVE AND REPLACE THE JW60 FITTING (3). USE A STRIP OF STEEL UNDER THE BOTTOM END OF THE SPAR TO PROTECT IT WHEN DRILLING HOLES.



SECTION D △4

STEP 3

REFINISH REWORKED PARTS WITH ALODINE 1200S. APPLY TWO COATS OF TT-P-1757 ZINC CHROMATE PRIMER. REFER TO SECTION 51.8.



E

NOTE

△3 DO NOT STOP DRILL OUTBOARD PAST THIS POINT.

△4 SECTION D SHOWS EXTENT OF PERMISSIBLE DAMAGE BETWEEN SLOTS

△5 AVOID STOP DRILLING CLOSE TO THE VERTICAL WALLS.

JS-I-2-X0/2-5928E-2

Figure 59-28E. Landing Gear Actuator and Auxiliary Spar Outboard Support Fitting (JW60) Repair (Sheet 2 of 2)

CHAPTER 59.3
EMPENNAGE GROUP

DESCRIPTION 59-71

REPAIRS 59-71

REPLACEMENT OF ELEVATOR AND RUDDER TRAILING EDGE ASSEMBLIES 59-71

FIGURES

59-29 Horizontal Stabilizer Front Beam Cap Repair (Outboard to Box Beam Station 46) 59-73
59-30 Horizontal Stabilizer Front Beam Cap Repair (Box Beam Stations 46 to 88) 59-75
59-31 Horizontal Stabilizer Trailing Edge Tee Repair 59-77
59-31A Elevator Boost Bellcrank Repair for Oversize Bearing and Bolt Holes 59-78

59.3.1 DESCRIPTION. See Chapter 55 for a complete description of the empennage and empennage structure. Chapter 55 also contains detail component material and identification figures to aid in establishing the applicable repair figures in Sections 59.3 and 59.8.

59.3.2 REPAIRS. Figures 59-29 through 59-31A give specific repairs and figures 59-49 through 59-81 negligible damage limits and typical repairs applicable to the empennage.

59.3.3 REPLACEMENT OF ELEVATOR AND RUDDER TRAILING EDGE ASSEMBLIES. Allowable rivets for the installation of the elevator and rudder trailing edge assemblies are listed below.

HOLE DIAMETER	APPROVED FASTENERS
0.141 to 0.144	NAS1739E4-X CR3242-4-X CR3552P-4-X
0.156 to 0.159	CR3522P-5-X CR3212-5-X CR3222-5-X
0.173 to 0.176	NAS1739E5-X CR3242-5-X CR3252-5-X CR3552P-5-X

Note

1. The smallest diameter rivet, using the above noted hole size, shall be installed.
2. Indiscriminate oversize installation is not permitted.
3. These rivets shall be installed in countersunk holes with the heads of the rivets protruding 0.002 to 0.004 inches.
4. Holes for blind rivets should be carefully drilled. A light interference fit is desirable for rivets in this application.

NOTE

△1 DRILL LEADING EDGE SCREW HOLES THROUGH FILLER AND REPAIR MEMBERS.

△2 REATTACH THE MS21063L3-12 GANG CHANNEL WITH THE REPAIR RIVETS.

△3 MAKE FILLER SAME THICKNESS AS EXISTING CAP.

△4 USE SAME TYPE AND SIZE FASTENER AT EXISTING SPACING FOR THIS ROW. A MINIMUM OF 6 FASTENERS ARE REQUIRED EACH SIDE OF THE CUTOUT. USE ADDITIONAL FASTENERS AS NEEDED TO MAINTAIN UNIFORM LENGTH OF REPAIR MEMBERS.

△5 FILLER NOT REQUIRED IF FLANGE IS ONLY CRACKED. REMOVE CRACK BY DRILLING 3/16-INCH DIAMETER OR LARGER HOLE AND PENETRANT INSPECT.

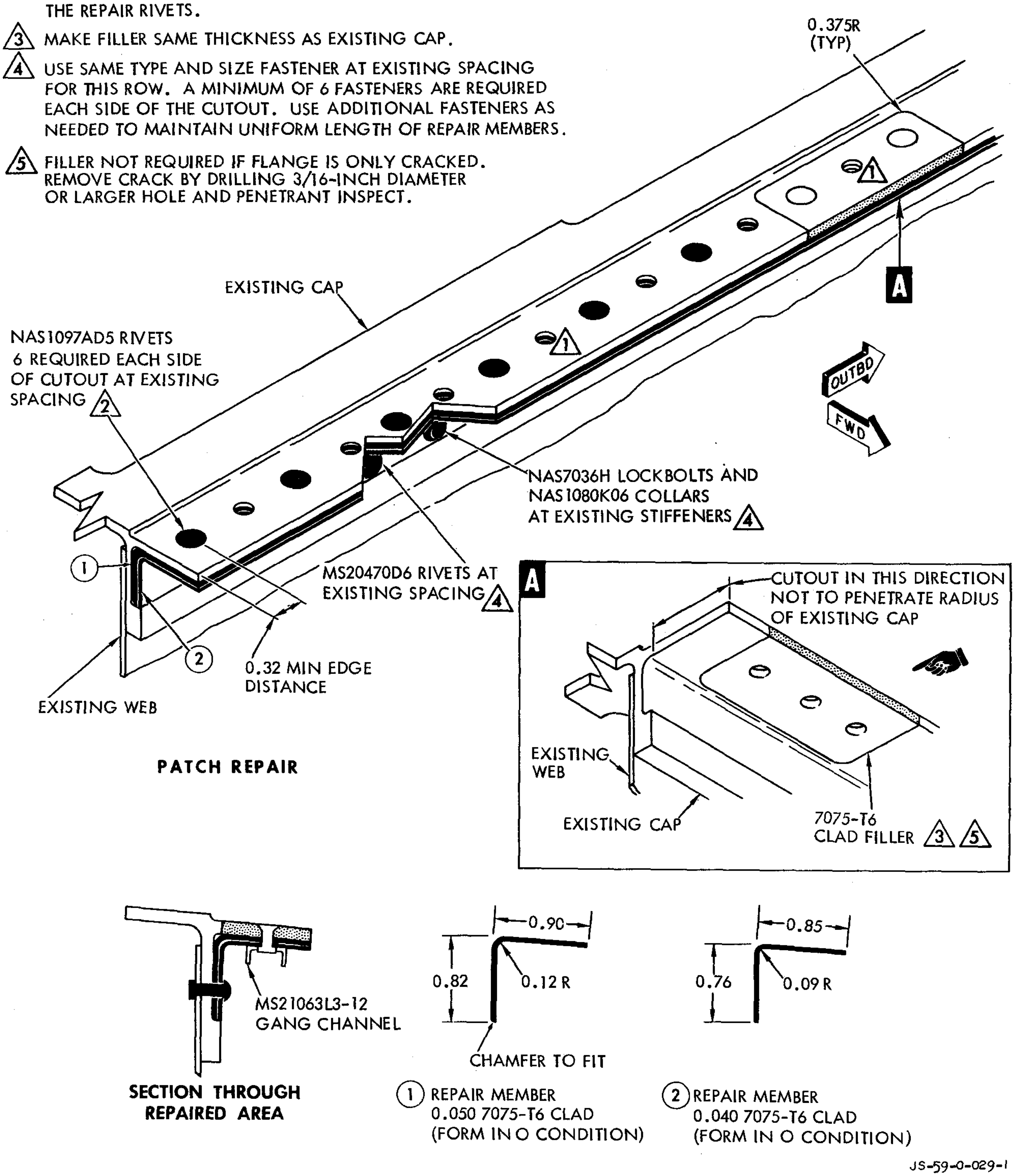
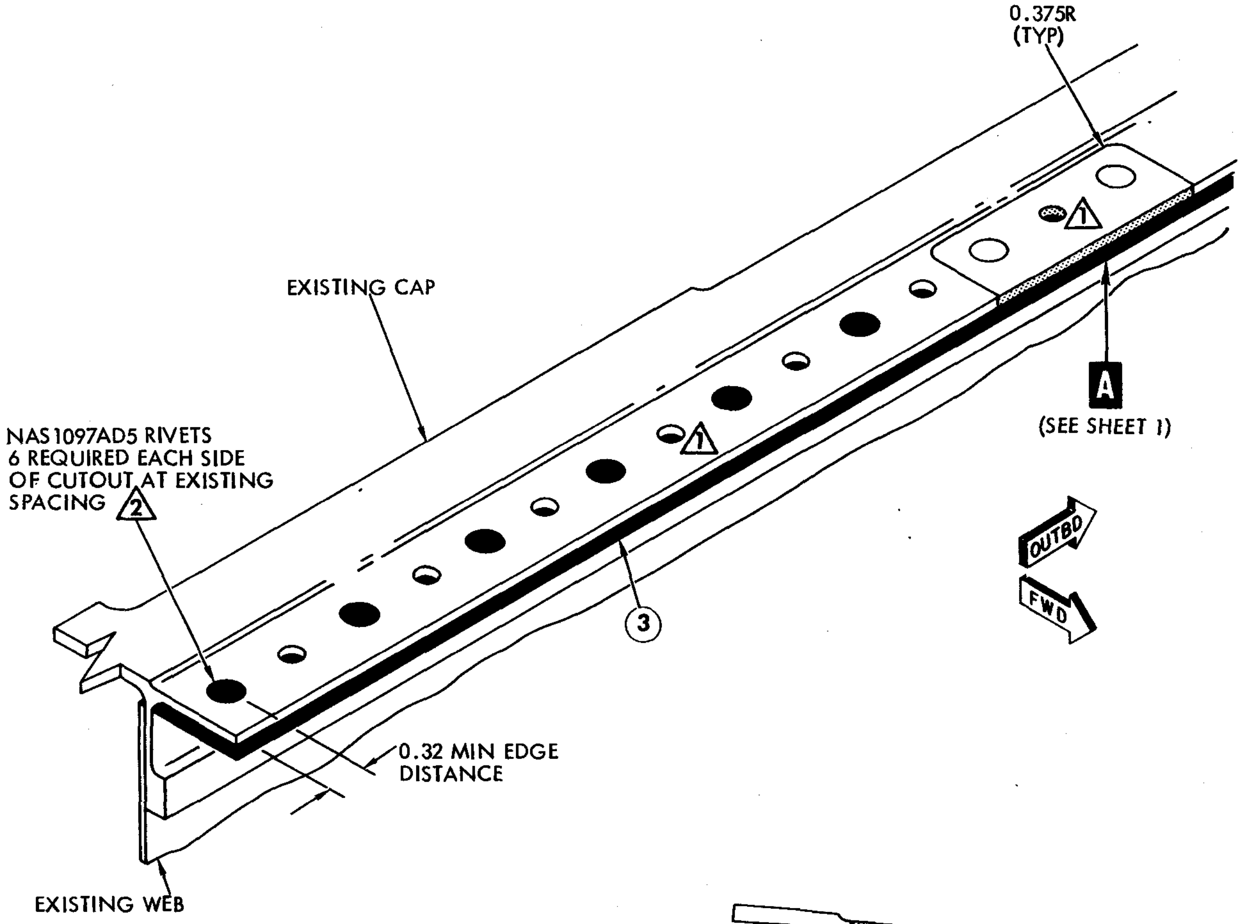
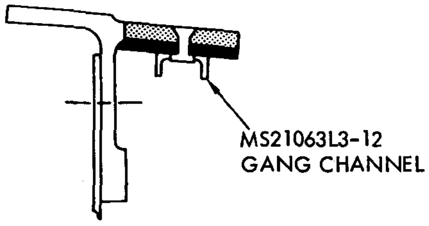

0.375R (TYP)
△1
A
EXISTING CAP
NAS1097AD5 RIVETS 6 REQUIRED EACH SIDE OF CUTOUT AT EXISTING SPACING △2
△1
OUTBD
FWD
NAS7036H LOCKBOLTS AND NAS1080K06 COLLARS AT EXISTING STIFFENERS △4
①
MS20470D6 RIVETS AT EXISTING SPACING △4
②
0.32 MIN EDGE DISTANCE
EXISTING WEB
PATCH REPAIR
A
CUTOUT IN THIS DIRECTION NOT TO PENETRATE RADIUS OF EXISTING CAP
EXISTING WEB
EXISTING CAP
7075-T6 CLAD FILLER △3 △5
MS21063L3-12 GANG CHANNEL
SECTION THROUGH REPAIRED AREA
0.90
0.82
0.12 R
CHAMFER TO FIT
① REPAIR MEMBER 0.050 7075-T6 CLAD (FORM IN O CONDITION)
0.85
0.76
0.09 R
② REPAIR MEMBER 0.040 7075-T6 CLAD (FORM IN O CONDITION)
JS-59-0-029-1

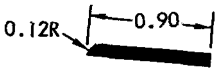
Figure 59-29. Horizontal Stabilizer Front Beam Cap Repair (Outboard to Box Beam Station 46) (Sheet 1 of 2)

OPTIONAL REPAIR - USE ONLY IF CLEARANCE CANNOT BE OBTAINED FOR REINSTALLATION OF GANG CHANNELS USING ANGLE REPAIR.





SECTION THROUGH REPAIRED AREA



③ REPAIR MEMBER
0.156 7075-T6
EXTRUSION OR SHEET

JS-1-2-X0/2-5929-2

Figure 59-29. Horizontal Stabilizer Front Beam Cap Repair (Outboard to Box Beam Station 46) (Sheet 2 of 2)

NOTE

△1 DRILL LEADING EDGE SCREW HOLES THROUGH FILLER AND REPAIR MEMBER.

△2 REATTACH THE MS21063L3-12 GANG CHANNEL WITH THE REPAIR RIVETS.

△3 MAKE FILLER SAME THICKNESS AS EXISTING CAP.

△4 USE SAME TYPE AND SIZE FASTENER AT EXISTING SPACING FOR THIS ROW. A MINIMUM OF 5 FASTENERS ARE REQUIRED EACH SIDE OF THE CUTOUT. USE ADDITIONAL FASTENERS AS NEEDED TO MAINTAIN UNIFORM LENGTH OF REPAIR MEMBER.

△5 FILLER NOT REQUIRED IF FLANGE IS ONLY CRACKED. REMOVE CRACK BY DRILLING 3/16-INCH DIAMETER OR LARGER HOLE AND PENETRANT INSPECT.

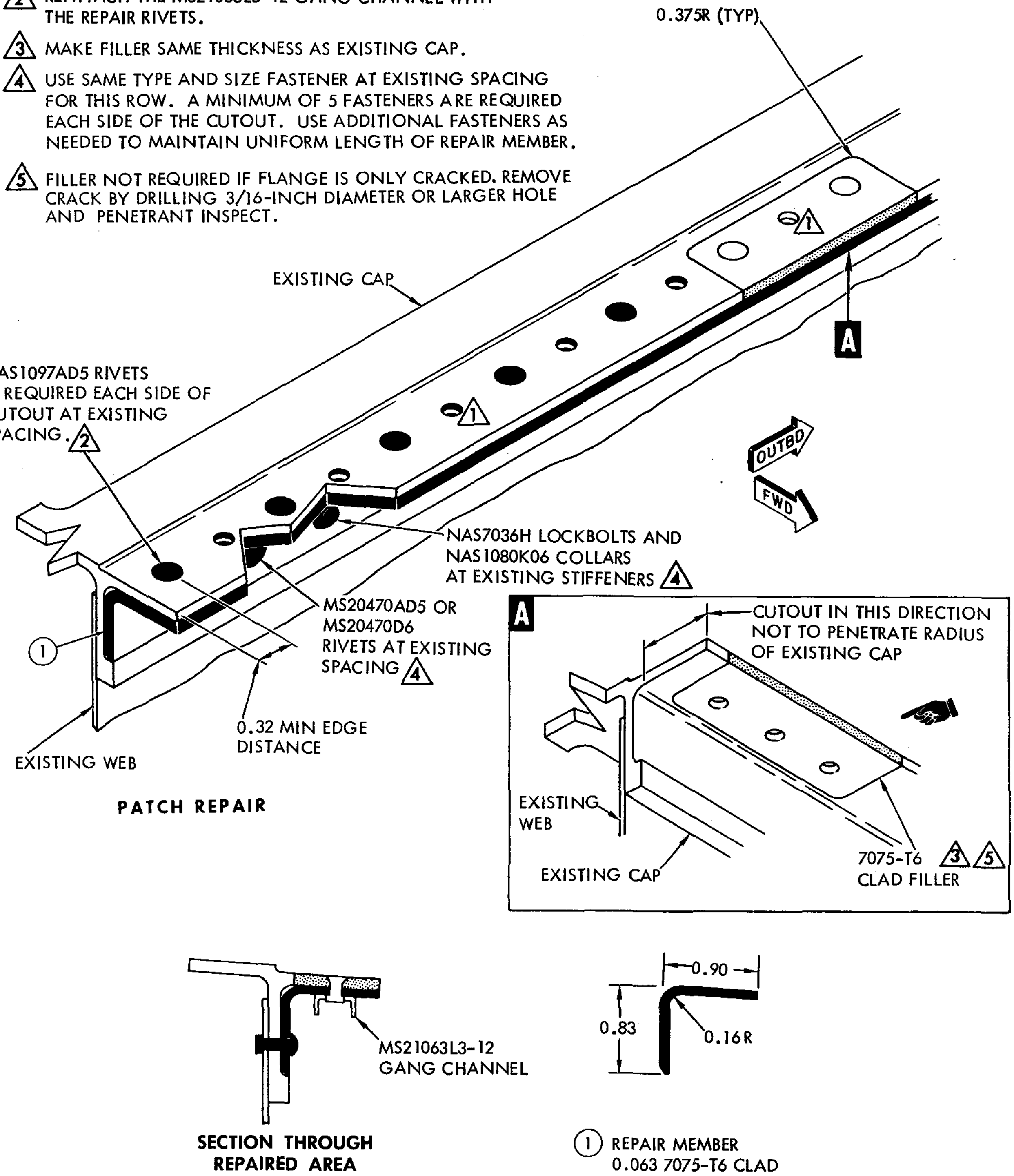

0.375R (TYP)
1
EXISTING CAP
A
NAS1097AD5 RIVETS
5 REQUIRED EACH SIDE OF CUTOUT AT EXISTING SPACING. 2
1
OUTBD
FWD
NAS7036H LOCKBOLTS AND NAS1080K06 COLLARS AT EXISTING STIFFENERS 4
MS20470AD5 OR MS20470D6 RIVETS AT EXISTING SPACING 4
1
0.32 MIN EDGE DISTANCE
EXISTING WEB
PATCH REPAIR
A
CUTOUT IN THIS DIRECTION NOT TO PENETRATE RADIUS OF EXISTING CAP
EXISTING WEB
EXISTING CAP
7075-T6 CLAD FILLER 3 5
MS21063L3-12 GANG CHANNEL
SECTION THROUGH REPAIRED AREA
0.90
0.83
0.16R
1 REPAIR MEMBER
0.063 7075-T6 CLAD
(FORM IN O CONDITION)
JS-59-0-030-1

Figure 59-30. Horizontal Stabilizer Front Beam Cap Repair (Box Beam Stations 46 to 88) (Sheet 1 of 2)

OPTIONAL REPAIR - USE ONLY IF CLEARANCE CANNOT BE OBTAINED FOR REINSTALLATION OF GANG CHANNELS USING ANGLE REPAIR.

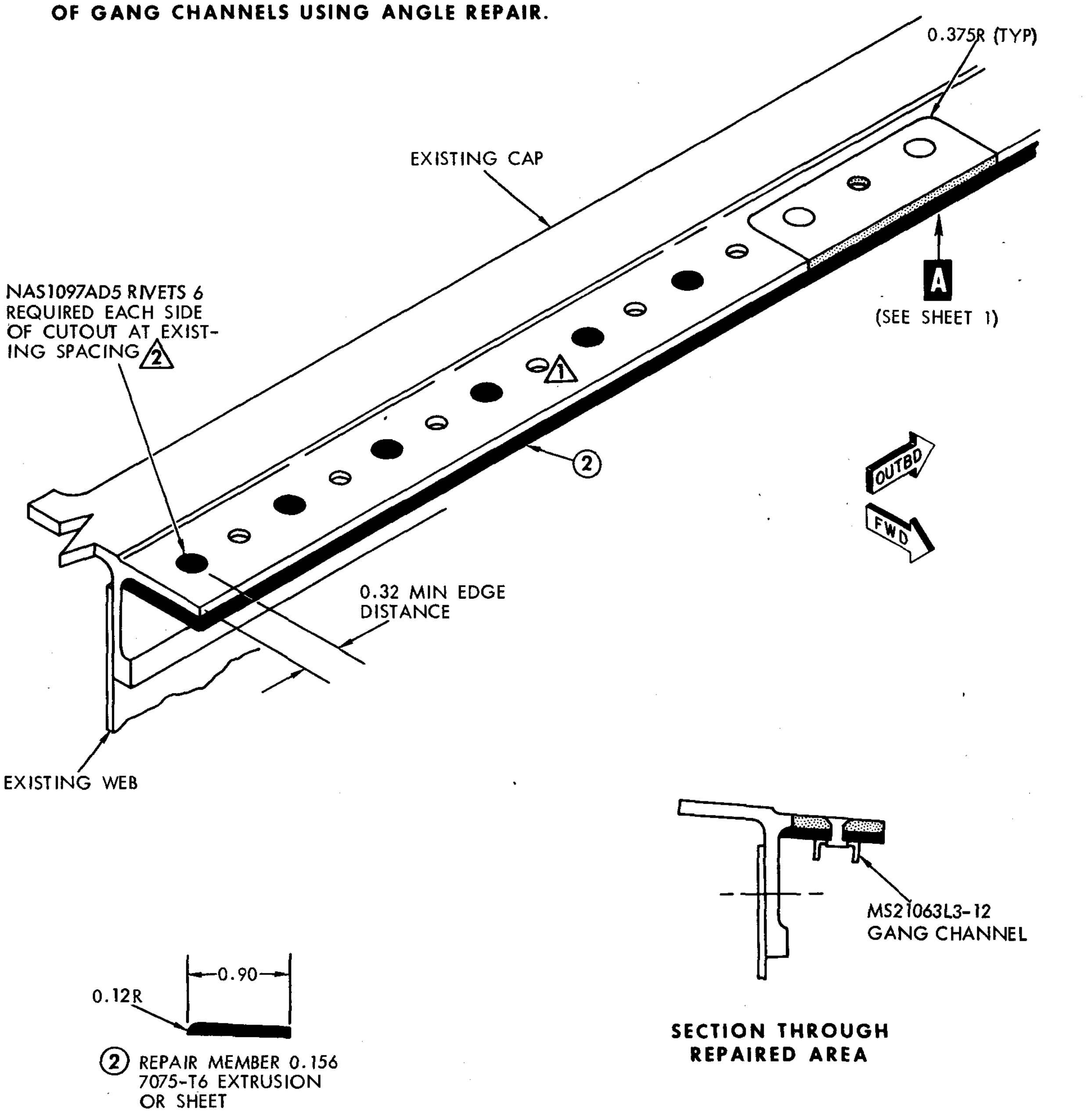
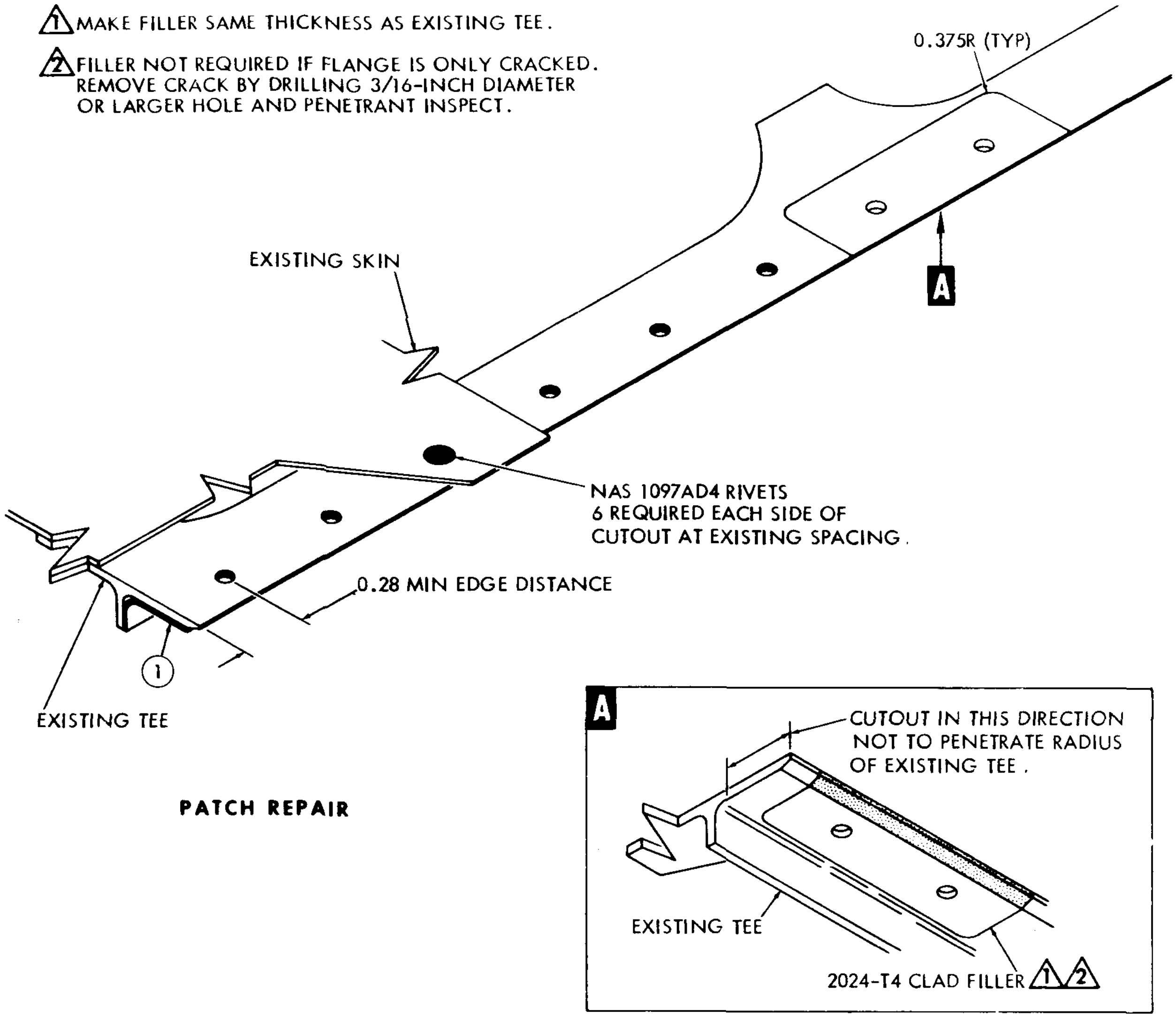

0.375R (TYP)
EXISTING CAP
NAS1097AD5 RIVETS 6 REQUIRED EACH SIDE OF CUTOUT AT EXISTING SPACING 2
1
A
(SEE SHEET 1)
2
OUTBD
FWD
0.32 MIN EDGE DISTANCE
EXISTING WEB
MS21063L3-12 GANG CHANNEL
0.90
0.12R
② REPAIR MEMBER 0.156 7075-T6 EXTRUSION OR SHEET
SECTION THROUGH REPAIRED AREA
JS-1-2-X0/2-5930-2

Figure 59-30. Horizontal Stabilizer Front Beam Cap Repair (Box Beam Stations 46 to 88) (Sheet 2 of 2)

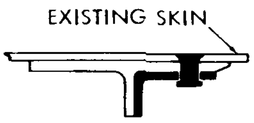
NOTE

⚠1 MAKE FILLER SAME THICKNESS AS EXISTING TEE.

⚠2 FILLER NOT REQUIRED IF FLANGE IS ONLY CRACKED. REMOVE CRACK BY DRILLING 3/16-INCH DIAMETER OR LARGER HOLE AND PENETRANT INSPECT.



PATCH REPAIR



SECTION THROUGH REPAIRED AREA

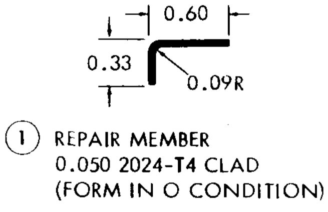
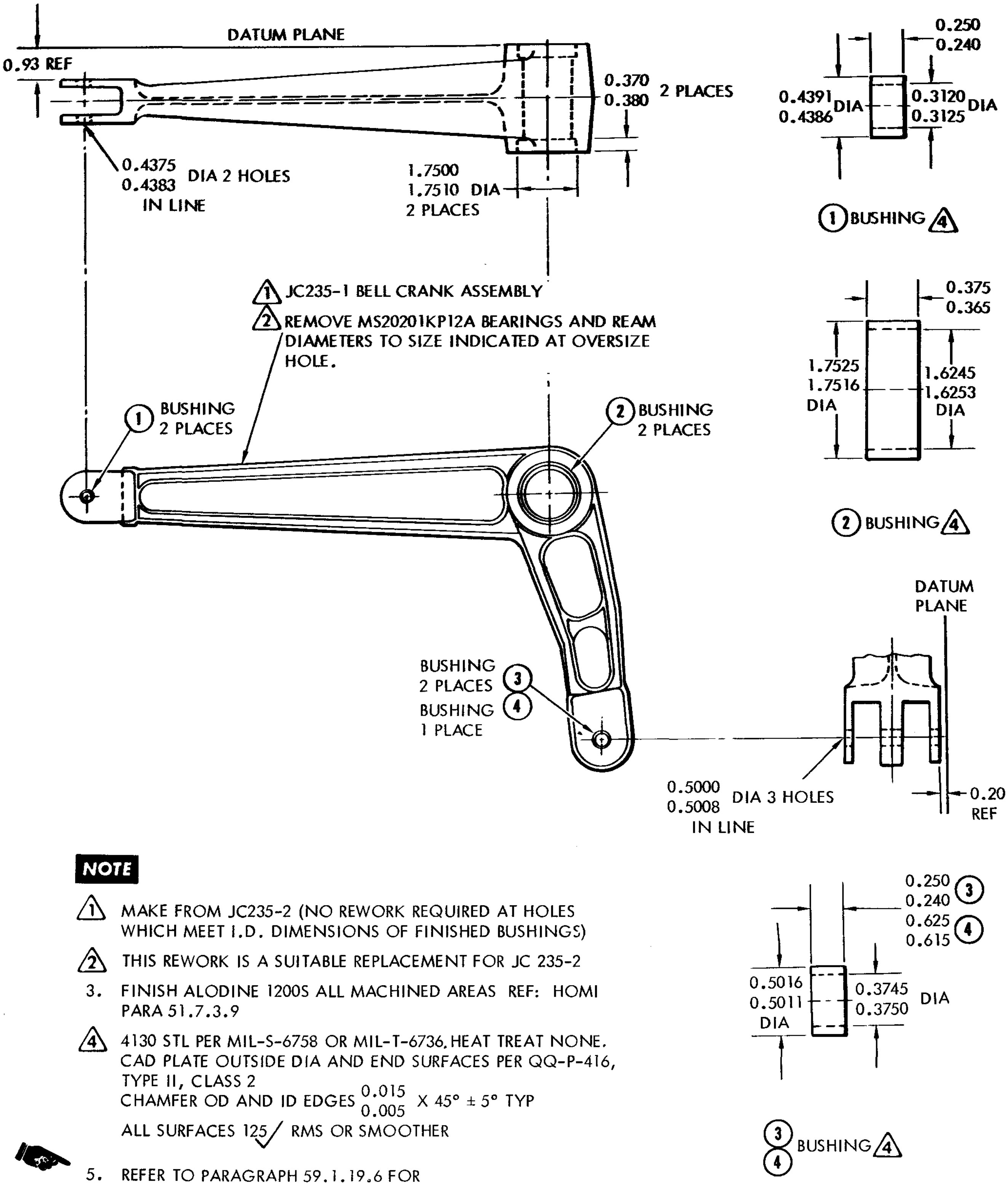


Figure 59-31. Horizontal Stabilizer Trailing Edge Tee Repair



△1 JC235-1 BELL CRANK ASSEMBLY

△2 REMOVE MS20201KP12A BEARINGS AND REAM DIAMETERS TO SIZE INDICATED AT OVERSIZE HOLE.

NOTE

△1 MAKE FROM JC235-2 (NO REWORK REQUIRED AT HOLES WHICH MEET I.D. DIMENSIONS OF FINISHED BUSHINGS)

△2 THIS REWORK IS A SUITABLE REPLACEMENT FOR JC 235-2

3. FINISH ALODINE 1200S ALL MACHINED AREAS REF: HOMI PARA 51.7.3.9

△4 4130 STL PER MIL-S-6758 OR MIL-T-6736, HEAT TREAT NONE. CAD PLATE OUTSIDE DIA AND END SURFACES PER QQ-P-416, TYPE II, CLASS 2
CHAMFER OD AND ID EDGES $^{0.015}_{0.005}$ X 45° ± 5° TYP
ALL SURFACES 125√ RMS OR SMOOTHER

5. REFER TO PARAGRAPH 59.1.19.6 FOR BUSHING INSTALLATION.

Figure 59-31A. Elevator Boost Bellcrank Repair for Oversize Bearing and Bolt Holes

SECTION 59.4

FUSELAGE GROUP

Contents Page
FUSELAGE STRUCTURAL REPAIRS 59-79
Pressure Skin Scratch Repair 59-79
Classification of Pressurized Skin Damage 59-79
Anti-Corrosion Control for Skin Repair 59-80

RADOME REPAIR 59-80
Restrictions 59-80
Critical Radome Areas 59-80
Electrical Test Deviations 59-80
Repair Materials 59-80
Repair Procedures 59-80
Replacement of Radome Metal Conductor Strips 59-80

RADOME RAIN EROSION-RESISTANT AND ANTISTATIC COATINGS 59-80

FIGURES
59-32 Fuselage Pressure Skin Scratch Limitations 59-81
59-33 Pressure Nose Shell Upper Longeron Repair 59-82
59-34 Pressure Nose Shell Lower Longeron Repair 59-83
59-35 Electronic Gear Support Structure Lower Cap Repair 59-84
59-36 Fuselage Ring Z Segment Repair 59-85
59-37 Fuselage Ring L Segment Repair 59-86
59-38 BL9 Longeron Repair (Fuselage Stations 283 to 296) 59-87
59-39 Top Longeron Repair (Fuselage Stations 277 to 397, and Fuselage Stations 502 to 539) 59-88
59-40 Top Longeron Repair (Fuselage Stations 403 to 495) 59-89
59-41 30-Degree Longeron Angle Repair (Fuselage Stations 277 to 350) 59-90
59-42 WL 73 Longeron Sill Repair (Fuselage Stations 283 to 410) 59-91
59-43 Fuselage Station 570 Bulkhead Circumferential Stiffener Repair 59-92
59-44 Fuselage Station 570 Bulkhead Radial Stiffener 59-93
59-45 Lower 28-Degree Longeron Repair (Fuselage Stations 608 to 682) 59-94
59-46 Deleted
59-47 Deleted
59-48 Deleted

59.4.1 FUSELAGE STRUCTURAL REPAIRS. Specific repairs to fuselage group members are referenced in the major assembly illustrations listed in figure 59-1. Refer to Section 59.1 for classification of damage and types of repairs. Refer to Section 59.8 for negligible damage limits and typical repairs.

59. 4. 1. 1 PRESSURE SKIN SCRATCH REPAIR. All scratches located by inspection must be reworked. Unworked pressure skin scratches result in stress concentrations which increase the likelihood of subsequent failure. After the scratch is removed, the depth of the reworked area must be determined in accordance with the instructions in Section 59. 8. If the depth is within the limits shown in figure 59-32, the damage is negligible and is regarded as Class A damage only. If the limits shown in figure 59-32 are exceeded, the damage is Class B, and a repair doubler must be installed, per Section 59. 8 to reinforce the weakened area.

59.4.1.2 CLASSIFICATION OF PRESSURIZED SKIN DAMAGE. Damage classification is defined as follows:

a. Class A damage

   1. Scratches which have a length and depth (after rework) that falls below the curves shown in figure 59-32.

   2. Tool marks or deformations which have been cold-worked to a radius of at least five times the thickness of the skin panel, and the maximum length and depth of the tool mark or deformation falls below the applicable curve shown in figure 59-32.

b. Class B damage

   1. Scratches which have a length and depth (after rework) that falls on or above the applicable curve shown in figure 59-32.

59.4.1.2 (Cont).

2. Tool marks or deformations which cannot be cold-worked to a radius of at least five times the thickness of the skin panel, regardless of their length and depth.

59.4.1.3 ANTI-CORROSION CONTROL FOR SKIN REPAIR. All reworked and repaired pressure skins must be given the proper anti-corrosion treatment. Refer to paragraph 51.7.2.1 for the steps to be followed to prevent corrosion to repairs.

59.4.2 RADOME REPAIR.

59.4.2.1 RESTRICTIONS. Repairable damage is limited to the removal and replacement of rain erosion and anti-static coating and to the repair of scars, abrasions, scratches, or erosion penetrating no more than one glass cloth face ply. Damaged radomes that are not within this repair limitation should be removed and returned to the manufacturer for the necessary repairs or replacement.

59.4.2.2 CRITICAL RADOME AREAS. The entire area of the nose and doppler radomes is critical. Before repairing damage to either radome make an investigation to determine whether the repair will make an electrical test necessary. See paragraph 59.4.2.3.

59.4.2.3 ELECTRICAL TEST DEVIATIONS. Electrical testing of a repaired radome will not be required if the radome meets either of the following conditions:

a. If the repair of the damaged area requires the application of less than a 0.005 inch thickness of resin for the repair of surface abrasion.

b. If the total area of the repair does not exceed one percent of the area of the aperture of the radar system antenna.

Note

If a radome requires an electrical test, send it to the manufacturer for repairs and tests.

59.4.2.4 REPAIR MATERIALS. The following materials are required for the repair of damaged radomes.

a. The resin to be used for repair is an epoxy resin conforming to MIL-R-9300 such as Epon 828, Shell Chemical Corporation. This resin can be made to cure at different temperatures by adjusting the amount of catalyst (curing agent D, Shell Chemical Corporation). Catalyze and cure the resin in accordance with the manufacturer's instructions.

Note

Use the usual precautions when mixing resins, catalysts, promoters, etc. These materials are very reactive and can be a dangerous source of fire and/or explosion.

b. Colored commercial grade cellophane to be used over the uncured resin. The cellophane reduces excess air and provides a smooth surface against which the resin may cure.

59.4.2.5 REPAIR PROCEDURES. Repair scars, scratches, surface abrasions, or minor rain erosion penetrating through no more than one glass cloth face ply as follows:

a. Remove the nose cone section or the doppler radome from the aircraft.

b. Clean the damaged area thoroughly using a clean cloth saturated with methylethylketone (MEK), or other approved cleaning agent.

CAUTION

Use extreme care in sanding to avoid further damage to the radome fairing.

c. Sand the damaged area lightly using No. 280 grit sandpaper. Clean the sanded surface thoroughly with MEK. Do not allow MEK or other solvents to dry on the radome surface. Wipe solvents dry with clean dry cloths. This will prevent the solvents from inhibiting the cure of the resin.

d. Apply one or two coats (depending on the severity of the damage) of the catalyzed resin to the sanded surface. Be sure that enough resin is applied to the sanded area to enable the area to fair with the adjoining unsanded area.

e. Apply to the resin coated surface a sheet of colored cellophane extending two or three inches beyond and all around the coated area. The cellophane will reduce excess air, and it will provide a smooth surface against which the resin may cure.

f. Tape the cellophane in place, and work out all air bubbles and excessive resin with the hand or a rubber squeegee. Subject the resin coated area to curing temperatures as specified by the resin manufacturer's instructions.

Note

Repair will not be satisfactory if the resin cures or sets with air trapped under the cellophane or in the resin.

g. After the resin has cured, remove the cellophane and remove any excess resin by wet sanding with No. 280 grit sandpaper. Remove grit and resin dust with MEK, and wipe dry with clean cloths.

59.4.2.6 REPLACEMENT OF RADOME METAL CONDUCTOR STRIPS.

a. Sand away the glass fabric patch over the forward end of the existing strap using No. 280 grit sandpaper.

59.4.2.6 (Cont).

Remove completely the remainder of the old strip and sand the location smooth. Remove the screw and washer at the aft end of the strip and clean inside surface of washer and screw head for good electrical contact.

b. The cadmium plated end of the new conductor strip should be free of cement and held in place by the screw and counter-sunk washer at the aft end. The remaining portion of the strip should be aligned with the position of the strip removed in step a and cemented to the radome using Scotchweld 2216 A/B epoxy cement or equal. The strip for the top of the radome is 17.00 inches long, the bottom is 16.50 inches long and each side is 6.00 inches long. All cadmium plated ends should be clean and under the counter-sunk washer at the aft end.

c. **Using the materials and procedure specified in paragraph 59.4.2.4, apply a one ply 0.5 X 1 inch glass fabric patch over the forward end of the strip so that it will cover the first 0.2 to 0.25 inch of the strip. After curing, the edges of this patch should be feathered into the radome.**

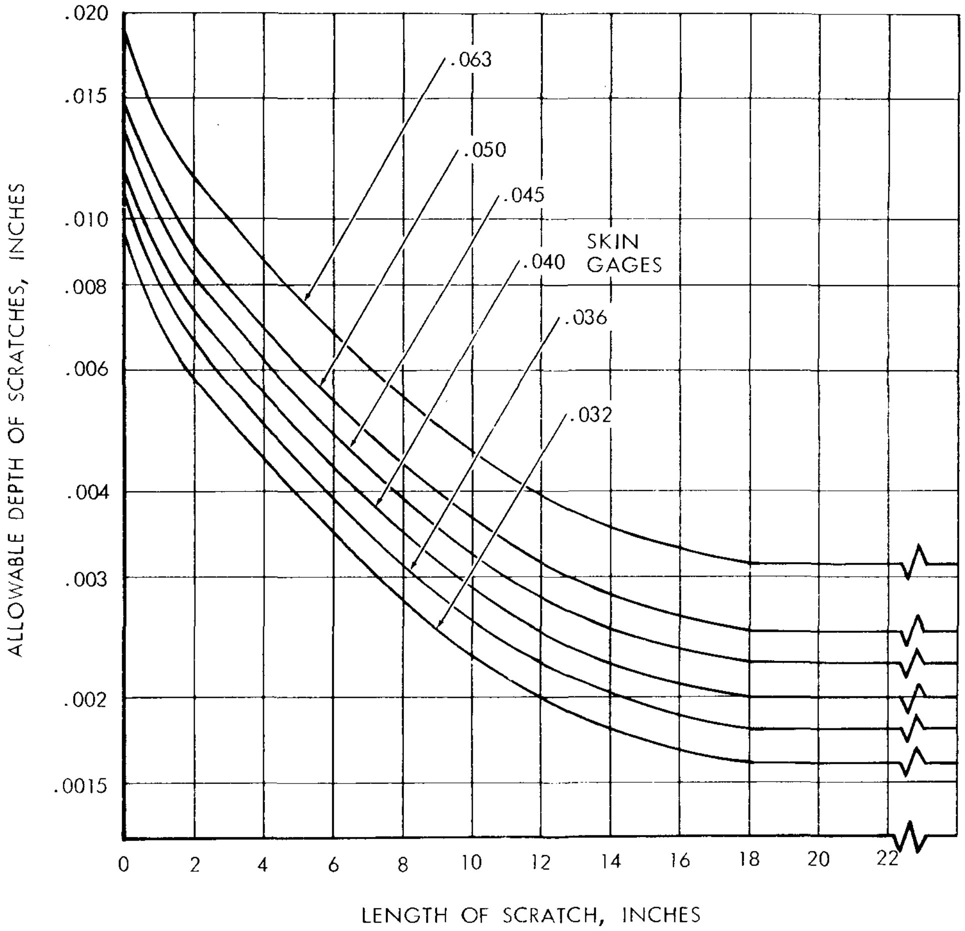
d. Tighten the screw and washer at the aft end without allowing the washer to rotate on the strap.

e. Touch up radome finish as required. (See paragraph 51.8.4.3.)

59.4.3 RADOME RAIN EROSION-RESISTANT AND ANTI-STATIC COATINGS. See paragraph 51.8.4.3 for instructions in applying rain erosion-resistant and anti-static coatings to the repaired area.

INSTRUCTIONS:

LOCATE ACROSS THE BOTTOM OF THE GRAPH THE LENGTH, IN INCHES, OF POSITIVE PENETRATION OF THE SCRATCH. EXTEND A VERTICAL LINE UP FROM THIS POINT UNTIL IT INTERSECTS A HORIZONTAL LINE EXTENDED FROM A POINT ON THE LEFT SIDE OF THE GRAPH, REPRESENTING THE DEPTH, IN INCHES, OF THE SCRATCH. IF THE POINT OF INTERSECTION OF THESE TWO LINES IS BELOW THE CURVE FOR THE PARTICULAR GAGE OF MATERIAL IN QUESTION, THE SCRATCH IS A CLASS "A" SCRATCH. IF THE INTERSECTION POINT IS ABOVE THE APPLICABLE CURVE, THE SCRATCH IS A CLASS "B" SCRATCH.



NOTE

1. THESE CURVES APPLY REGARDLESS OF SCRATCH DIRECTION.
2. THESE CURVES APPLY FOR 2024-T3 AND 7075-T6 MATERIAL.
3. THESE CURVES APPLY TO SCRATCHES ONLY, AND DO NOT CONSIDER COLD WORKING OF THE MATERIAL DUE TO OVER-DRIVING OF THE RIVET GUN OR BUCKING BAR

Figure 59-32. Fuselage Pressure Skin Scratch Limitations

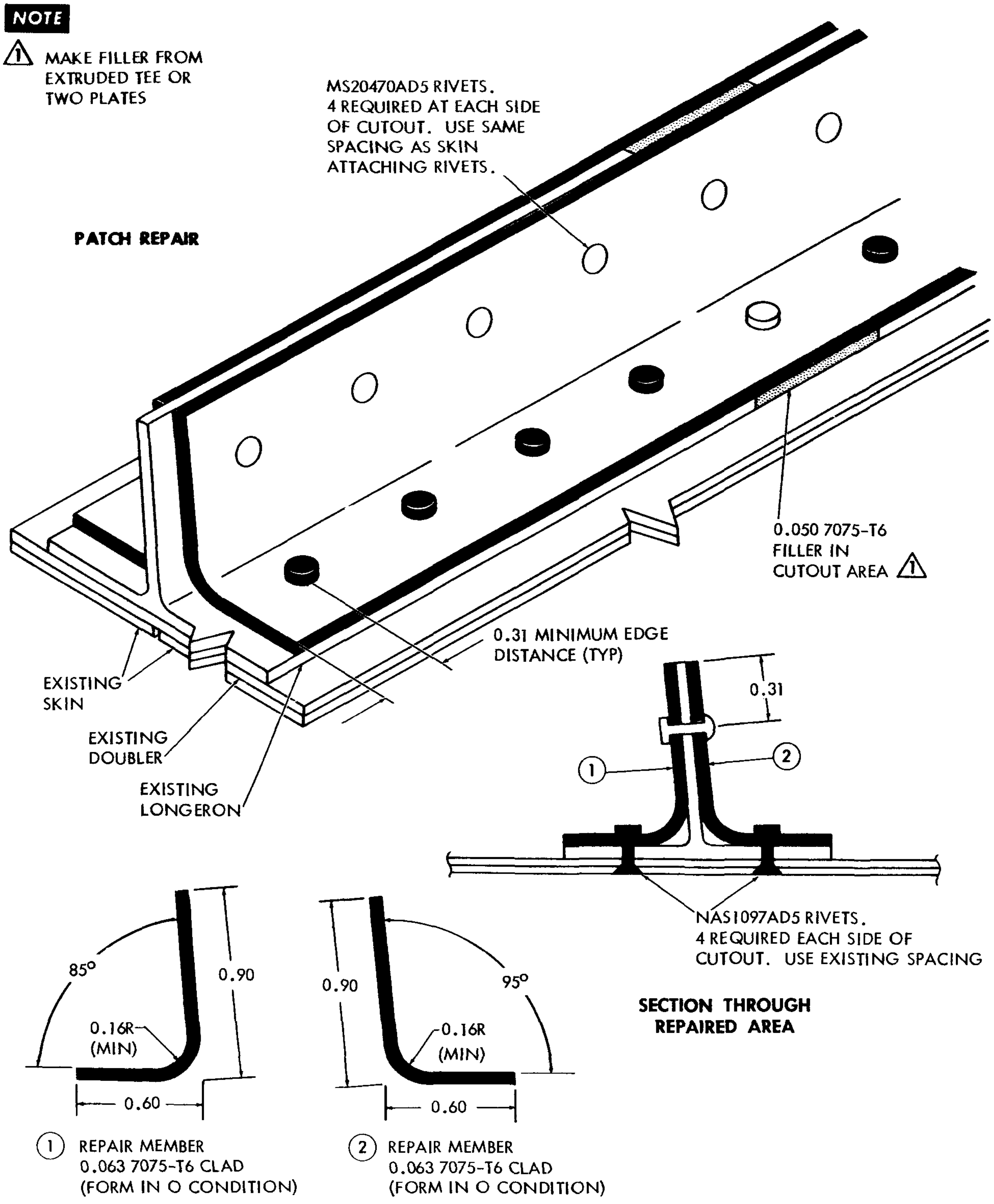

NOTE
⚠1 MAKE FILLER FROM EXTRUDED TEE OR TWO PLATES
PATCH REPAIR
MS20470AD5 RIVETS. 4 REQUIRED AT EACH SIDE OF CUTOUT. USE SAME SPACING AS SKIN ATTACHING RIVETS.
0.050 7075-T6 FILLER IN CUTOUT AREA ⚠1
0.31 MINIMUM EDGE DISTANCE (TYP)
EXISTING SKIN
EXISTING DOUBLER
EXISTING LONGERON
0.31
①
②
NAS1097AD5 RIVETS. 4 REQUIRED EACH SIDE OF CUTOUT. USE EXISTING SPACING
SECTION THROUGH REPAIRED AREA
85°
0.90
0.16R (MIN)
0.60
0.90
95°
0.16R (MIN)
0.60
① REPAIR MEMBER 0.063 7075-T6 CLAD (FORM IN O CONDITION)
② REPAIR MEMBER 0.063 7075-T6 CLAD (FORM IN O CONDITION)

Figure 59-33. Pressure Nose Shell Upper Longeron Repair

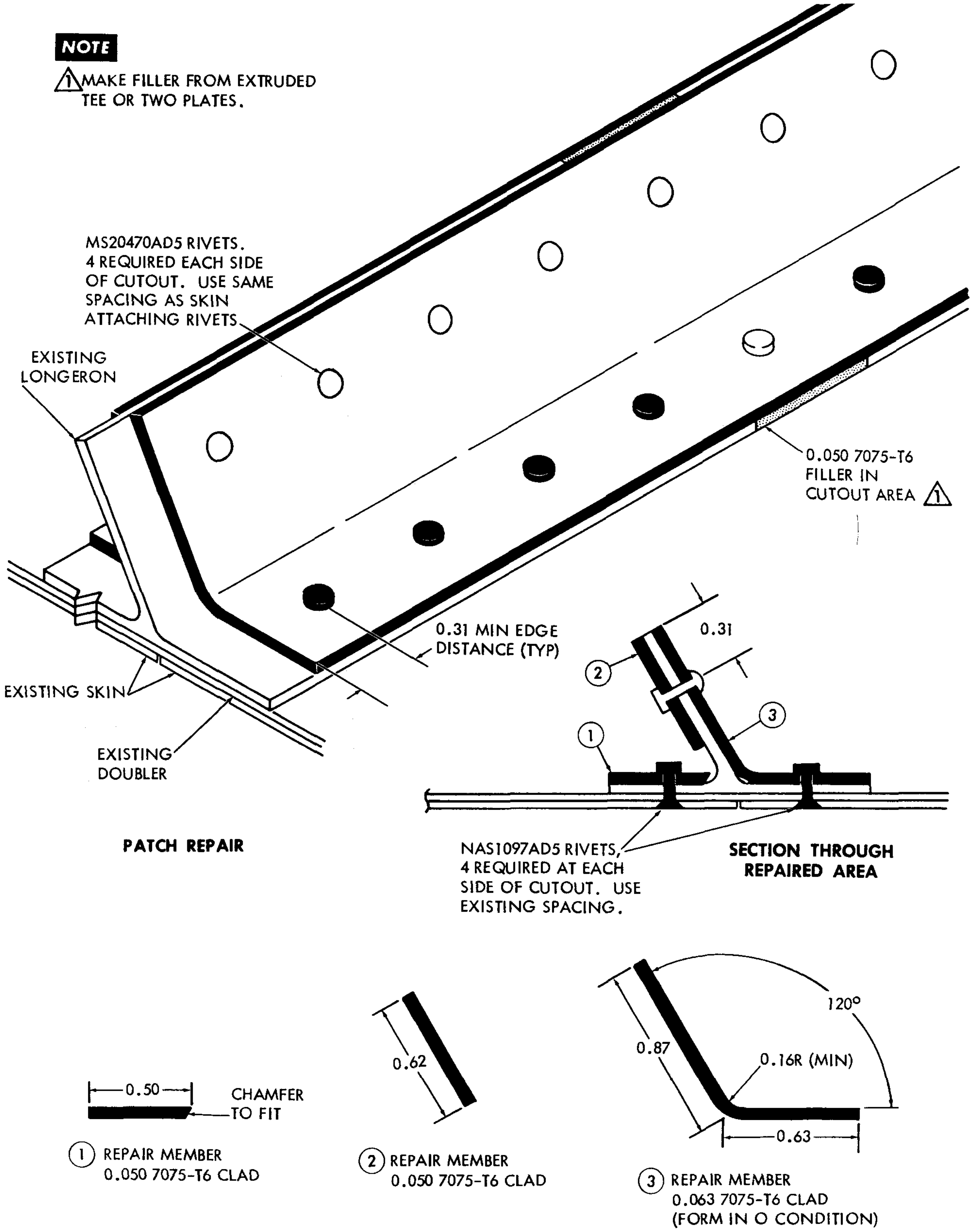

NOTE
1 MAKE FILLER FROM EXTRUDED TEE OR TWO PLATES.
MS20470AD5 RIVETS. 4 REQUIRED EACH SIDE OF CUTOUT. USE SAME SPACING AS SKIN ATTACHING RIVETS
EXISTING LONGERON
0.050 7075-T6 FILLER IN CUTOUT AREA 1
0.31 MIN EDGE DISTANCE (TYP)
EXISTING SKIN
EXISTING DOUBLER
PATCH REPAIR
0.31
2
3
1
NAS1097AD5 RIVETS, 4 REQUIRED AT EACH SIDE OF CUTOUT. USE EXISTING SPACING.
SECTION THROUGH REPAIRED AREA
0.50
CHAMFER TO FIT
1 REPAIR MEMBER 0.050 7075-T6 CLAD
0.62
2 REPAIR MEMBER 0.050 7075-T6 CLAD
0.87
120°
0.16R (MIN)
0.63
3 REPAIR MEMBER 0.063 7075-T6 CLAD (FORM IN O CONDITION)

Figure 59-34. **Pressure Nose Shell Lower Longeron Repair**

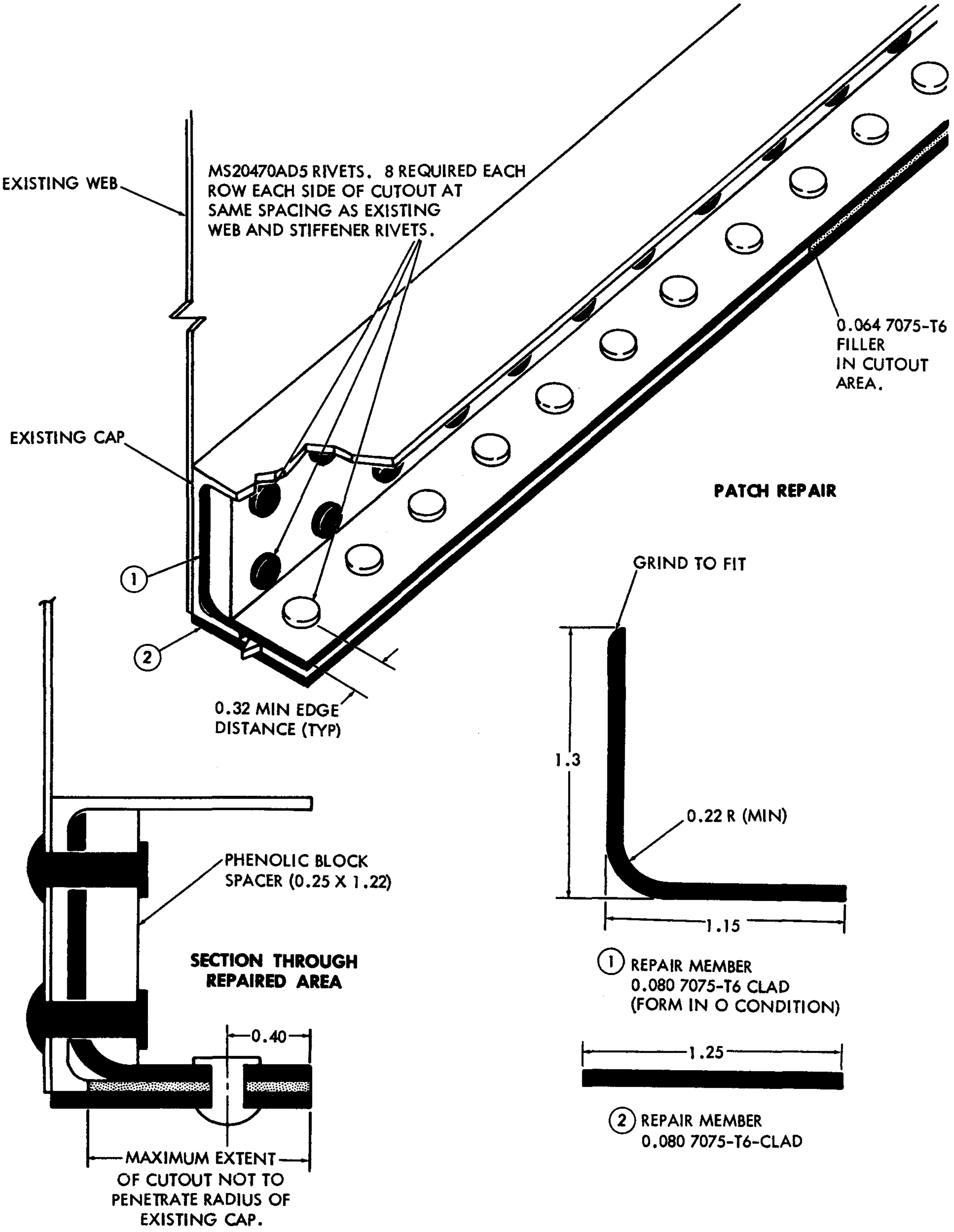

EXISTING WEB
MS20470AD5 RIVETS. 8 REQUIRED EACH ROW EACH SIDE OF CUTOUT AT SAME SPACING AS EXISTING WEB AND STIFFENER RIVETS.
0.064 7075-T6 FILLER IN CUTOUT AREA.
EXISTING CAP
PATCH REPAIR
①
②
0.32 MIN EDGE DISTANCE (TYP)
GRIND TO FIT
1.3
0.22 R (MIN)
1.15
① REPAIR MEMBER 0.080 7075-T6 CLAD (FORM IN O CONDITION)
1.25
② REPAIR MEMBER 0.080 7075-T6-CLAD
PHENOLIC BLOCK SPACER (0.25 X 1.22)
SECTION THROUGH REPAIRED AREA
0.40
MAXIMUM EXTENT OF CUTOUT NOT TO PENETRATE RADIUS OF EXISTING CAP.

Figure 59-35 Electronic Gear Support Structure Lower Cap Repair

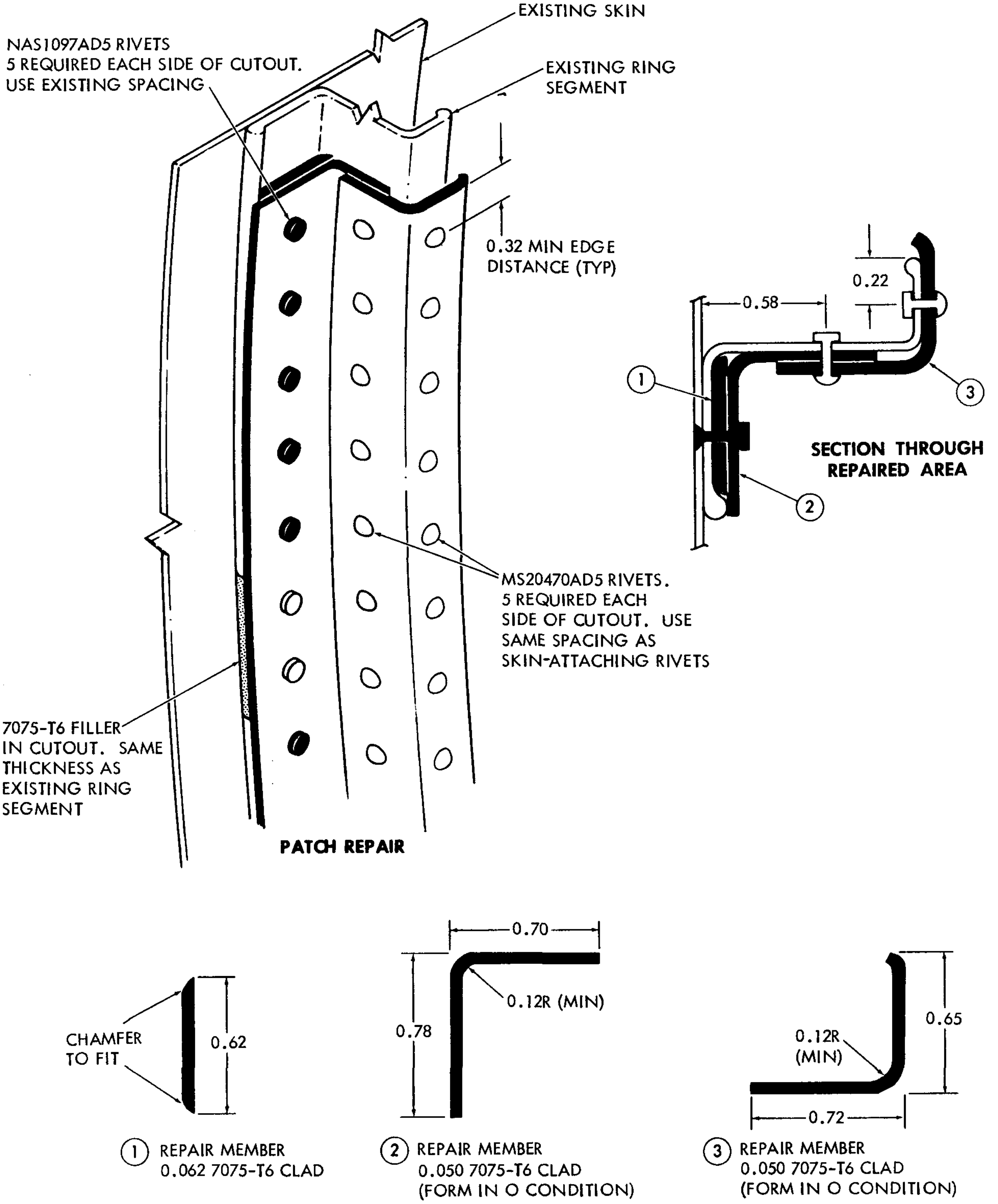


Figure 59-36. ***Fuselage Ring "Z" Segment Repair***

Refer to Figure 59-61 for Fuselage Ring "L" Segment Repair

Figure 59-37. Fuselage Ring "L" Segment Repair

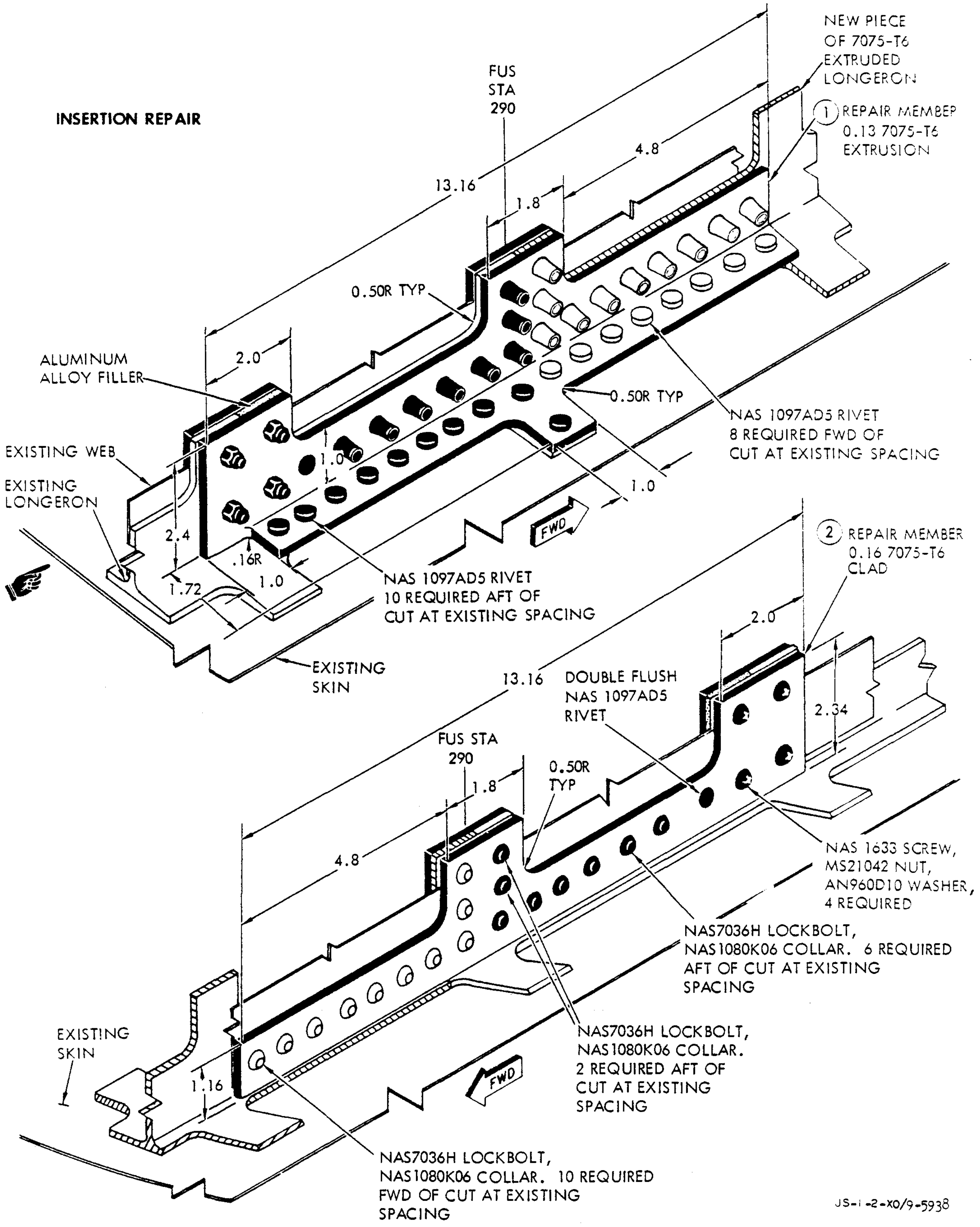
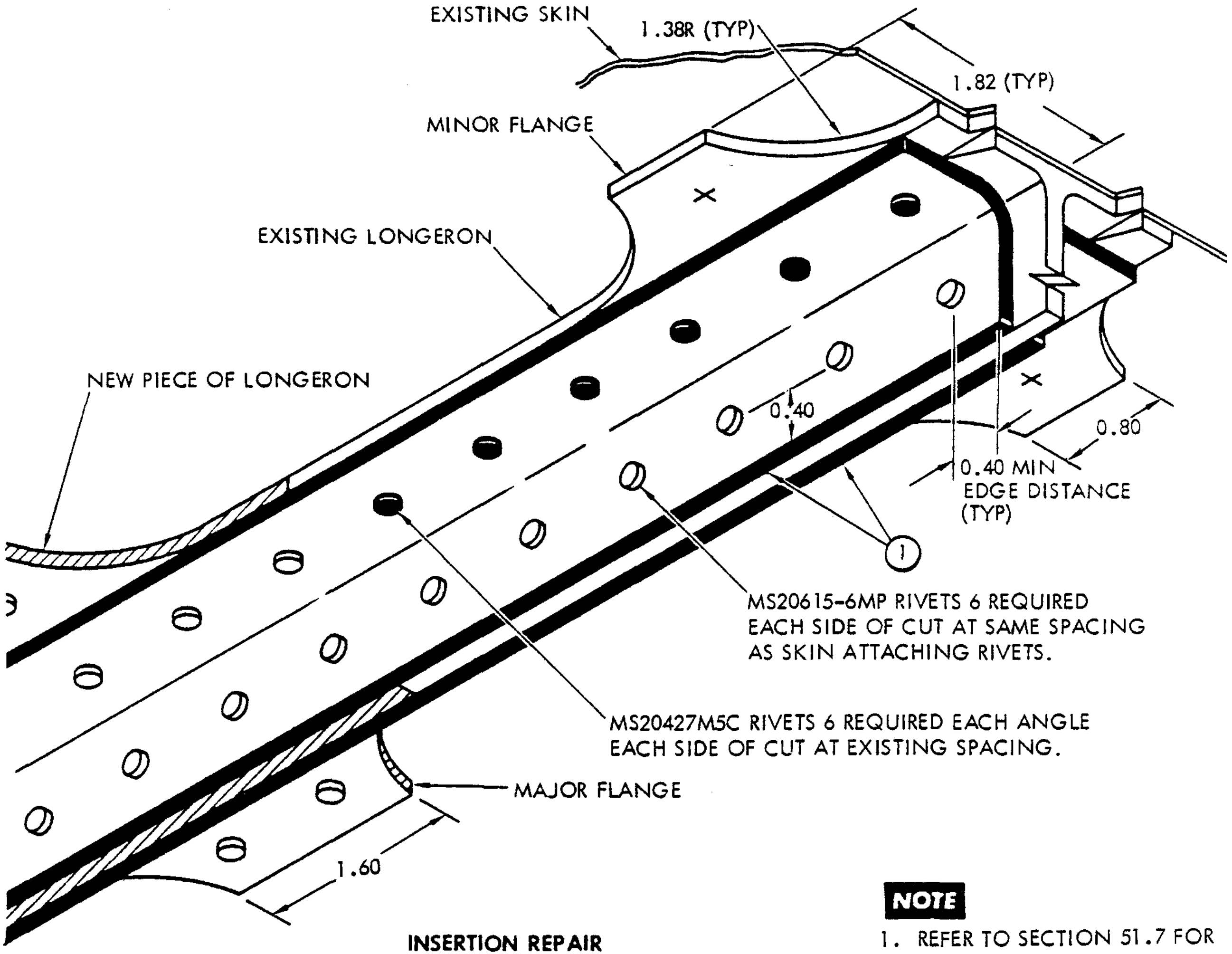

INSERTION REPAIR
FUS STA 290
NEW PIECE OF 7075-T6 EXTRUDED LONGERON
① REPAIR MEMBER 0.13 7075-T6 EXTRUSION
4.8
13.16
1.8
0.50R TYP
2.0
ALUMINUM ALLOY FILLER
0.50R TYP
NAS 1097AD5 RIVET 8 REQUIRED FWD OF CUT AT EXISTING SPACING
EXISTING WEB
1.0
EXISTING LONGERON
1.0
FWD
2.4
.16R
1.72
1.0
NAS 1097AD5 RIVET 10 REQUIRED AFT OF CUT AT EXISTING SPACING
EXISTING SKIN
② REPAIR MEMBER 0.16 7075-T6 CLAD
2.0
13.16
DOUBLE FLUSH NAS 1097AD5 RIVET
2.34
FUS STA 290
1.8
0.50R TYP
4.8
NAS 1633 SCREW, MS21042 NUT, AN960D10 WASHER, 4 REQUIRED
NAS7036H LOCKBOLT, NAS1080K06 COLLAR. 6 REQUIRED AFT OF CUT AT EXISTING SPACING
EXISTING SKIN
NAS7036H LOCKBOLT, NAS1080K06 COLLAR. 2 REQUIRED AFT OF CUT AT EXISTING SPACING
1.16
FWD
NAS7036H LOCKBOLT, NAS1080K06 COLLAR. 10 REQUIRED FWD OF CUT AT EXISTING SPACING
JS-1-2-X0/9-5938

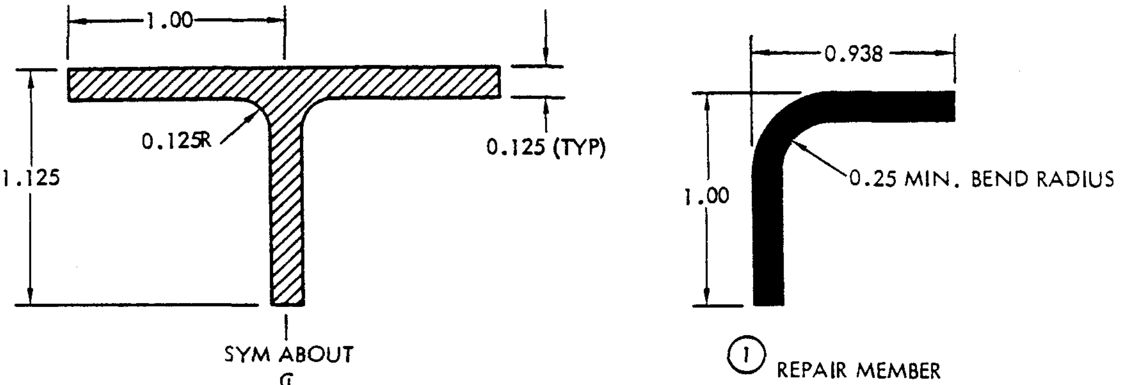
Figure 59-38. BL 9 Longeron Repair (Fuselage Stations 283 to 296)


EXISTING SKIN
1.38R (TYP)
1.82 (TYP)
MINOR FLANGE
EXISTING LONGERON
NEW PIECE OF LONGERON
0.40
0.80
0.40 MIN
EDGE DISTANCE
(TYP)
①
MS20615-6MP RIVETS 6 REQUIRED
EACH SIDE OF CUT AT SAME SPACING
AS SKIN ATTACHING RIVETS.
MS20427M5C RIVETS 6 REQUIRED EACH ANGLE
EACH SIDE OF CUT AT EXISTING SPACING.
MAJOR FLANGE
1.60

INSERTION REPAIR

NOTE

1. REFER TO SECTION 51.7 FOR CORROSION PREVENTION PRECAUTIONS FOR DISSIMILAR METALS.

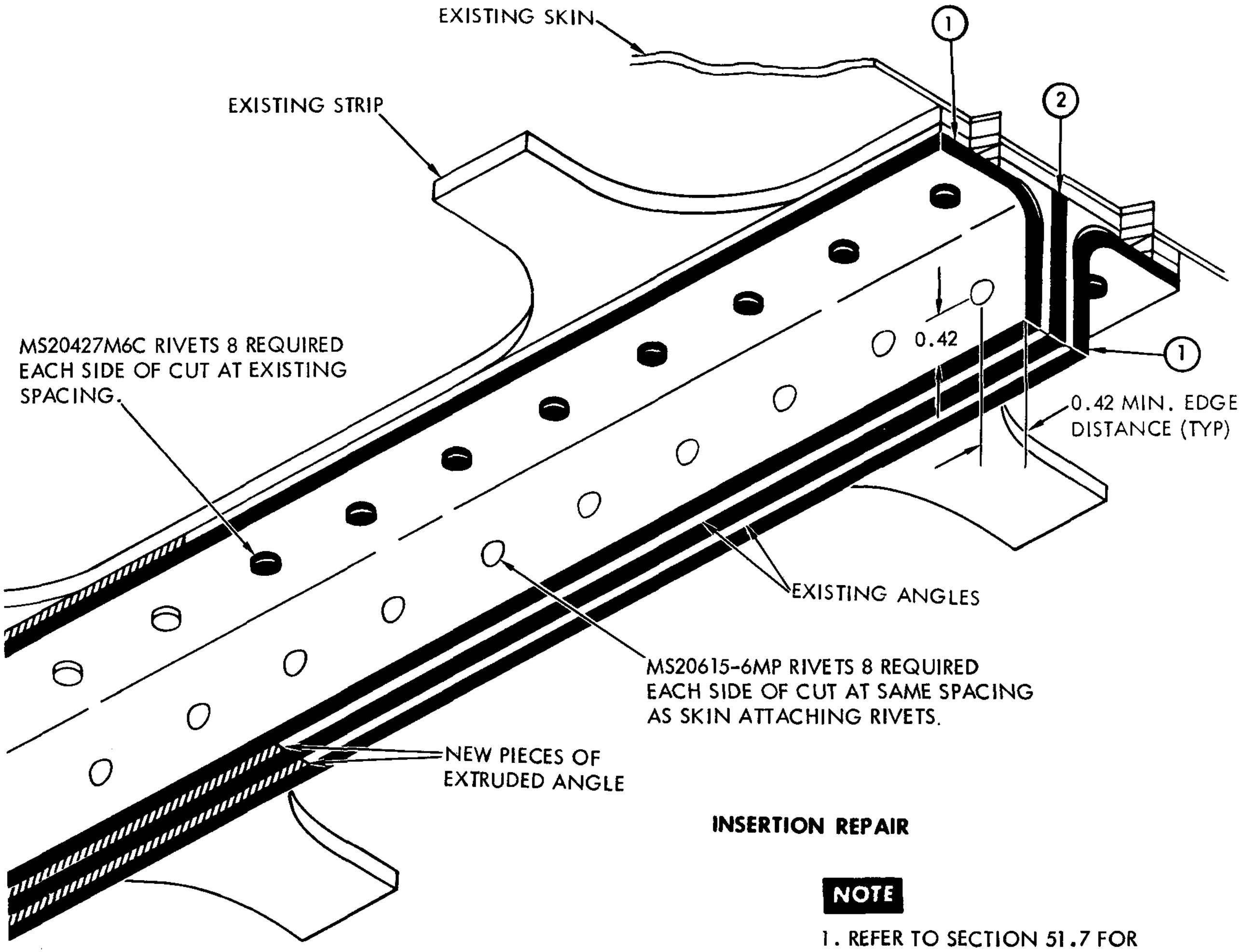


NEW PIECE OF 7075-T6 EXTRUSION
AT SECTION BETWEEN FRAMES.

REPAIR MEMBER
(TWO REQUIRED)
0.125 4130 STEEL
CONDITION N

JS-1-2-X0/2-5939

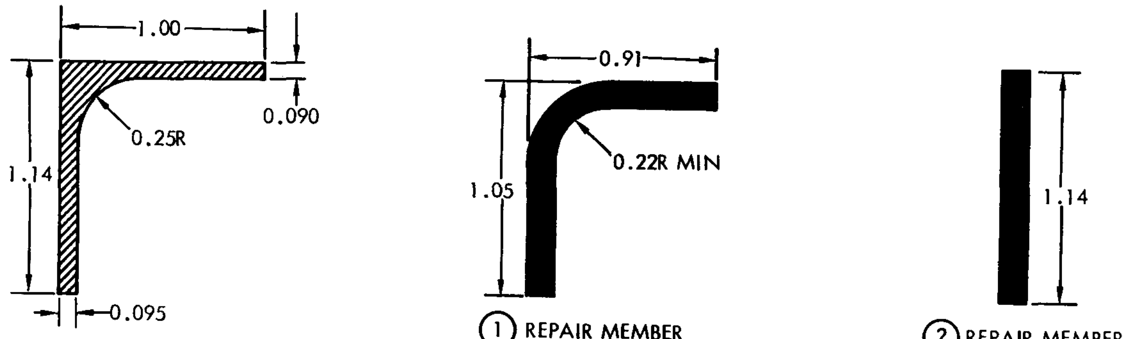
Figure 59-39. Top Longeron Repair (Fuselage Stations 277 to 397, and Fuselage Stations 502 to 539)


EXISTING SKIN
EXISTING STRIP
①
②
MS20427M6C RIVETS 8 REQUIRED EACH SIDE OF CUT AT EXISTING SPACING.
0.42
①
0.42 MIN. EDGE DISTANCE (TYP)
EXISTING ANGLES
MS20615-6MP RIVETS 8 REQUIRED EACH SIDE OF CUT AT SAME SPACING AS SKIN ATTACHING RIVETS.
NEW PIECES OF EXTRUDED ANGLE

INSERTION REPAIR

NOTE

1. REFER TO SECTION 51.7 FOR FOR CORROSION PREVENTION PRECAUTIONS FOR DISSIMILAR METAL.



NEW PIECE OF 7075-T6 EXTRUDED ANGLE

① REPAIR MEMBER
(2 REQUIRED)
0.112 4130 STEEL,
CONDITION N

② REPAIR MEMBER
0.125 7075-T6
PLATE

JS-1-2-X0/2-5940

Figure 59-40. Top Longeron Repair (Fuselage Stations 403 to 495)

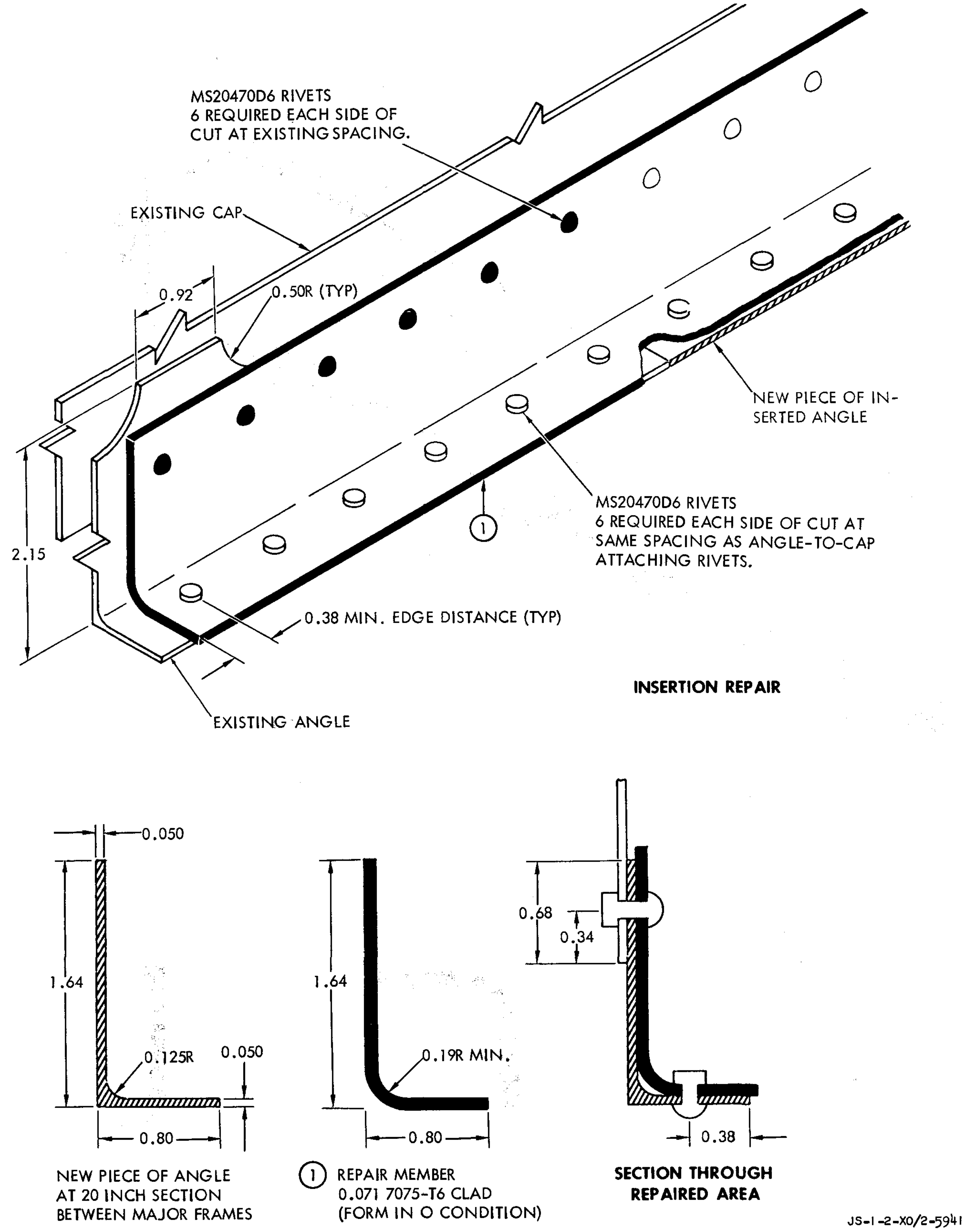
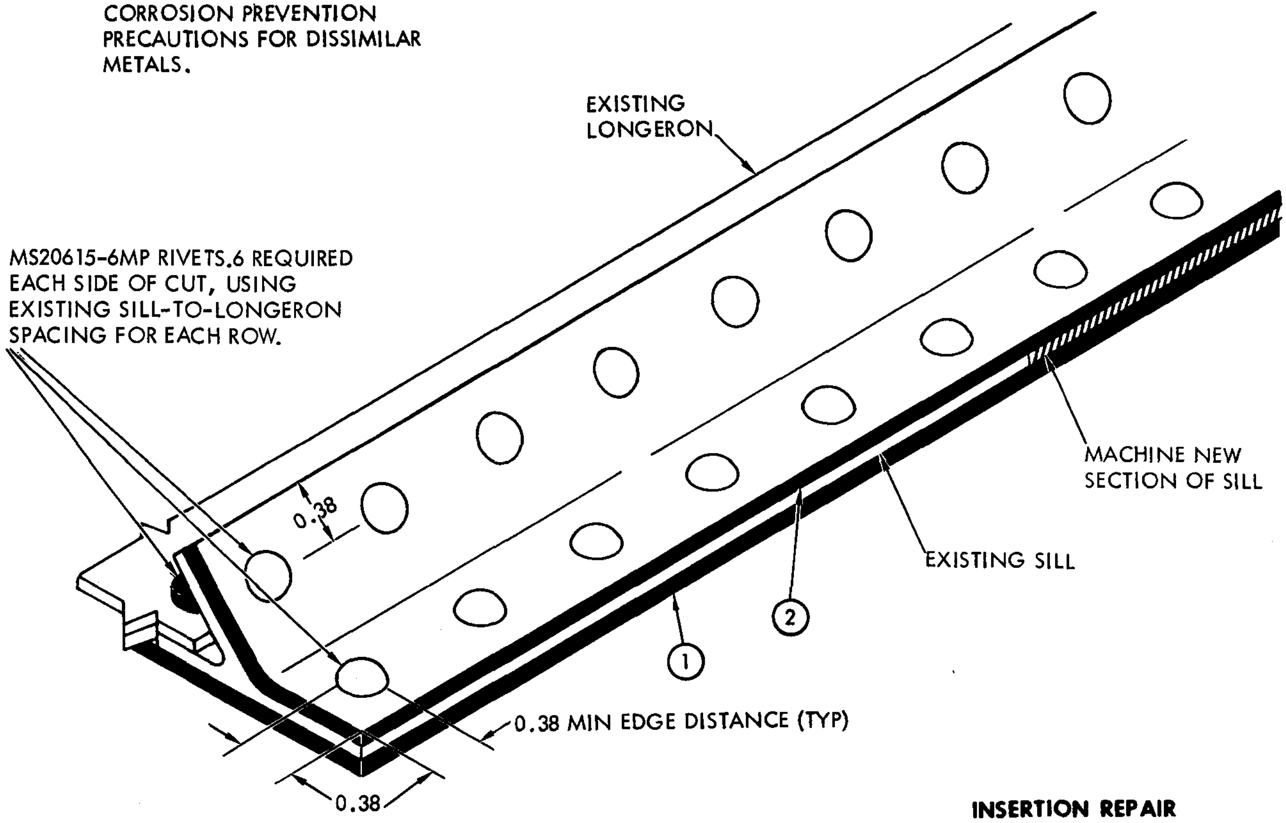

MS20470D6 RIVETS
6 REQUIRED EACH SIDE OF CUT AT EXISTING SPACING.
EXISTING CAP
0.92
0.50R (TYP)
NEW PIECE OF INSERTED ANGLE
2.15
1
MS20470D6 RIVETS
6 REQUIRED EACH SIDE OF CUT AT SAME SPACING AS ANGLE-TO-CAP ATTACHING RIVETS.
0.38 MIN. EDGE DISTANCE (TYP)
EXISTING ANGLE
INSERTION REPAIR
0.050
1.64
0.125R
0.050
0.80
NEW PIECE OF ANGLE AT 20 INCH SECTION BETWEEN MAJOR FRAMES
1.64
0.19R MIN.
0.80
① REPAIR MEMBER 0.071 7075-T6 CLAD (FORM IN O CONDITION)
0.68
0.34
0.38
SECTION THROUGH REPAIRED AREA
JS-1-2-XO/2-5941

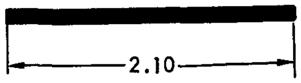
Figure 59-41. 30-Degree Longeron Angle Repair (Fuselage Stations 277 to 350)

NOTE

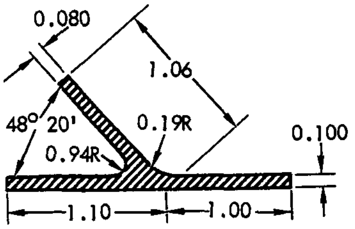
1. REFER TO SECTION 51.7 FOR CORROSION PREVENTION PRECAUTIONS FOR DISSIMILAR METALS.


EXISTING LONGERON
MS20615-6MP RIVETS.6 REQUIRED EACH SIDE OF CUT, USING EXISTING SILL-TO-LONGERON SPACING FOR EACH ROW.
0.38
MACHINE NEW SECTION OF SILL
EXISTING SILL
②
①
0.38 MIN EDGE DISTANCE (TYP)
0.38

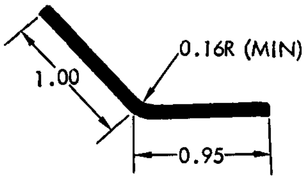
INSERTION REPAIR



① REPAIR MEMBER
0.100 4130 STEEL
CONDITION N



NEW PIECE OF SILL
AT 20 INCH SECTION
BETWEEN MAJOR FRAMES



② REPAIR MEMBER
0.080 4130 STEEL
CONDITION N

JS-1-2-X0/2-5942

Figure 59-42. WL 73 Longeron Sill Repair (Fuselage Stations 283 to 410)

NOTE

PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4.

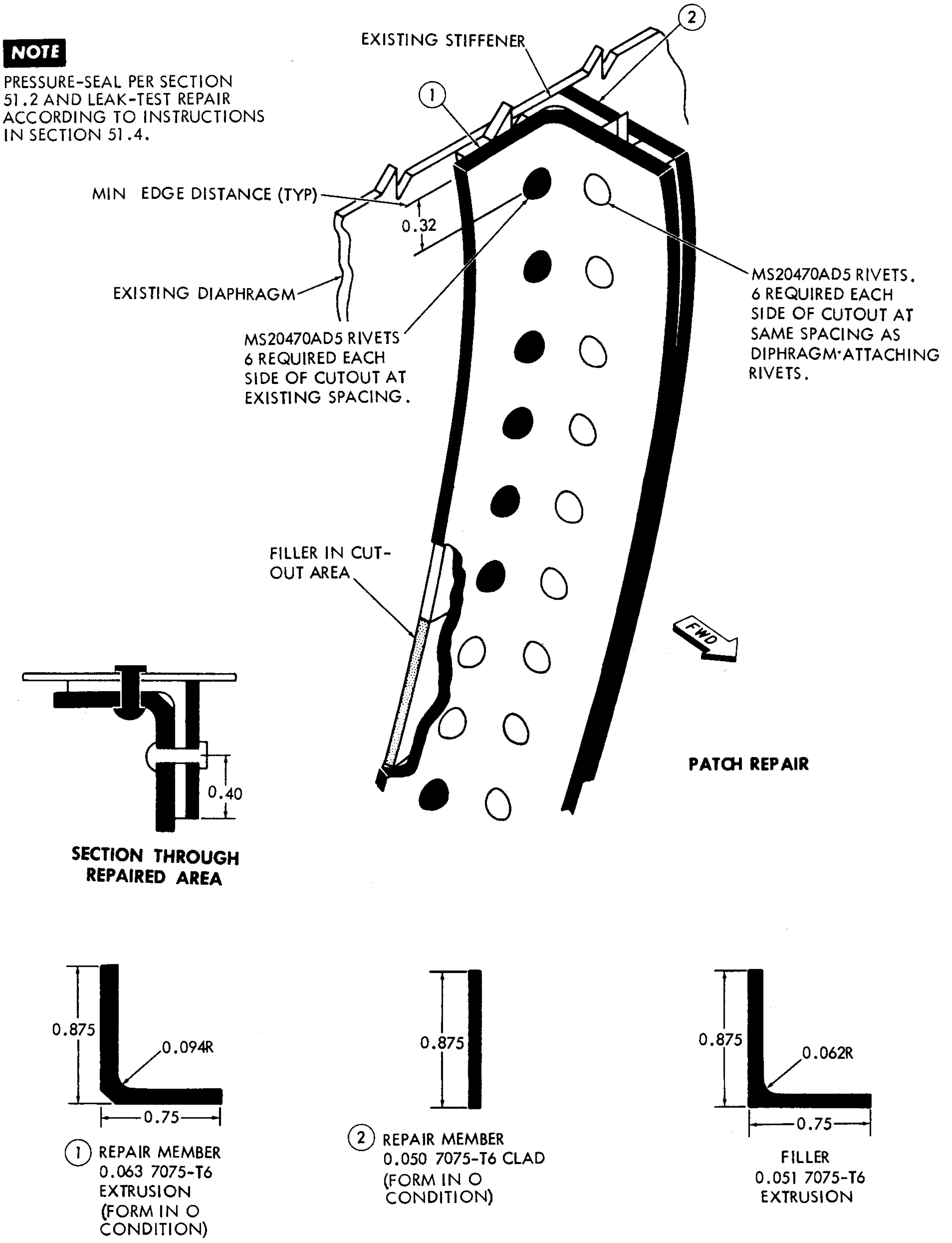

EXISTING STIFFENER
②
①
MIN EDGE DISTANCE (TYP)
0.32
EXISTING DIAPHRAGM
MS20470AD5 RIVETS 6 REQUIRED EACH SIDE OF CUTOUT AT EXISTING SPACING.
MS20470AD5 RIVETS. 6 REQUIRED EACH SIDE OF CUTOUT AT SAME SPACING AS DIPHRAGM·ATTACHING RIVETS.
FILLER IN CUT-OUT AREA
FWD
PATCH REPAIR
0.40
SECTION THROUGH REPAIRED AREA
0.875
0.094R
0.75
① REPAIR MEMBER 0.063 7075-T6 EXTRUSION (FORM IN O CONDITION)
0.875
② REPAIR MEMBER 0.050 7075-T6 CLAD (FORM IN O CONDITION)
0.875
0.062R
0.75
FILLER 0.051 7075-T6 EXTRUSION

Figure 59-43. Fuselage Station 570 Bulkhead Circumferential Stiffener Repair

NOTE

1. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4.

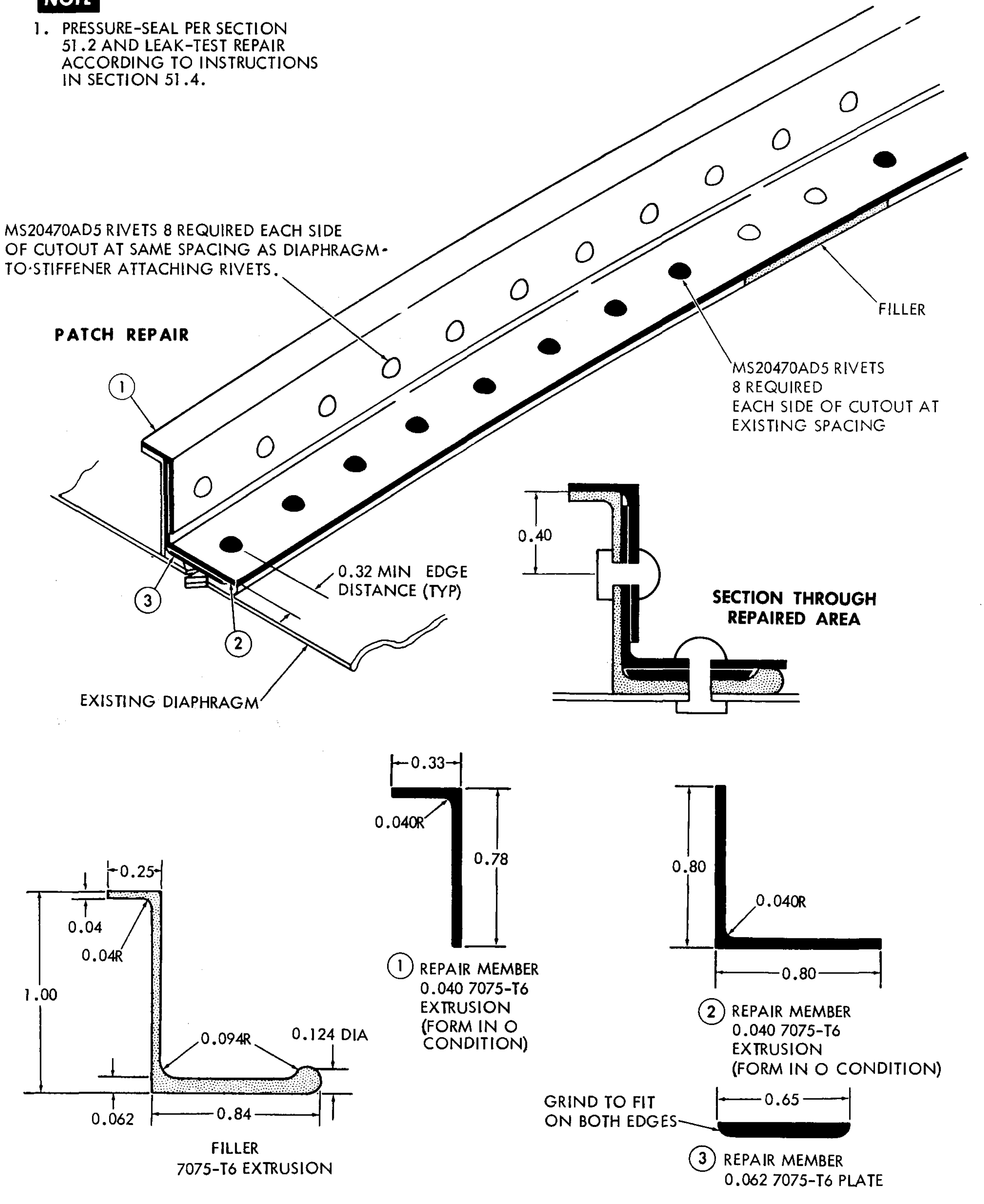
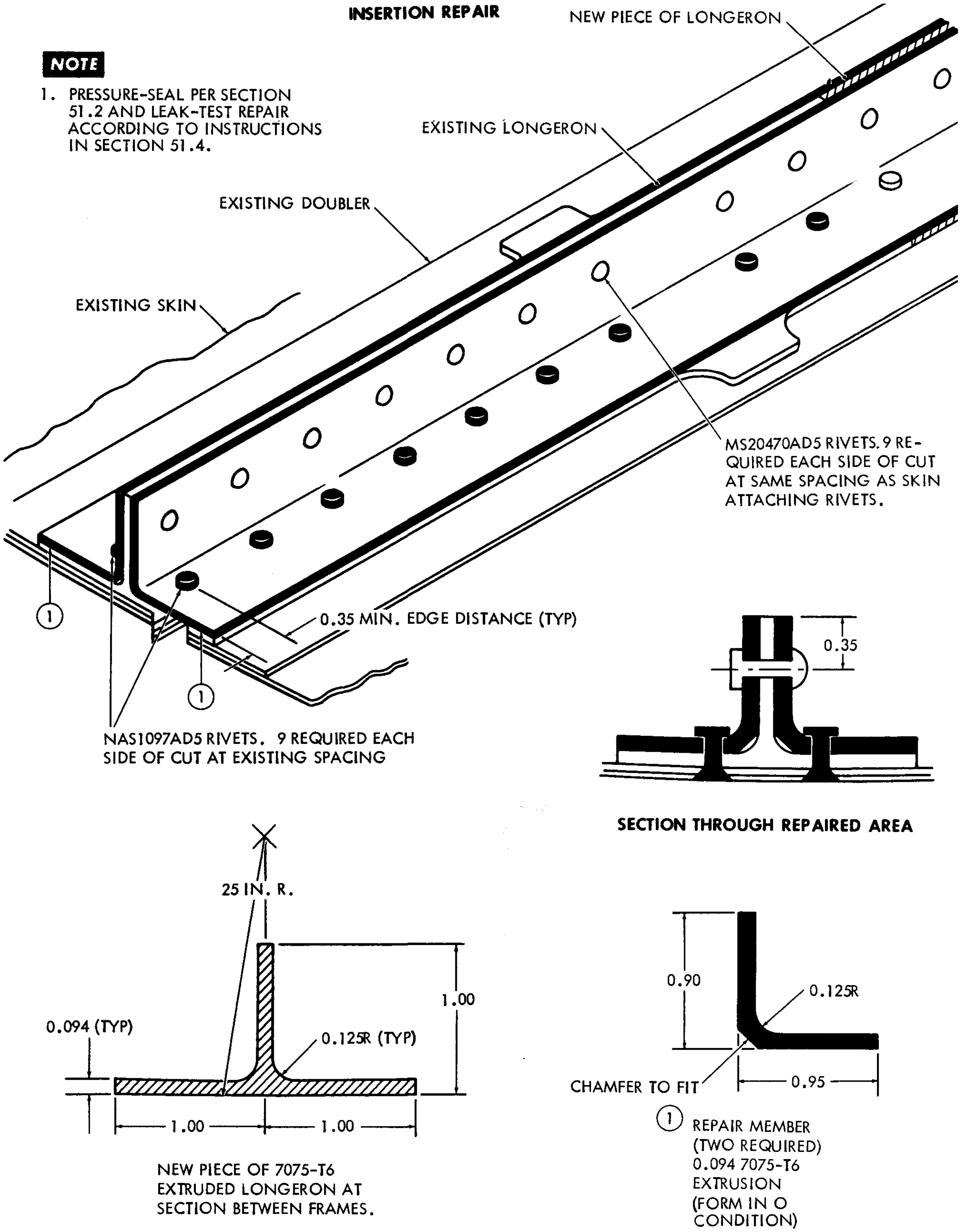


Figure 59-44. Fuselage Station 570 Bulkhead Radial Stiffener Repair



NOTE

1. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4.

INSERTION REPAIR
NEW PIECE OF LONGERON
EXISTING LONGERON
EXISTING DOUBLER
EXISTING SKIN
MS20470AD5 RIVETS. 9 REQUIRED EACH SIDE OF CUT AT SAME SPACING AS SKIN ATTACHING RIVETS.
0.35 MIN. EDGE DISTANCE (TYP)
①
①
NAS1097AD5 RIVETS. 9 REQUIRED EACH SIDE OF CUT AT EXISTING SPACING
0.35
SECTION THROUGH REPAIRED AREA
25 IN. R.
1.00
0.094 (TYP)
0.125R (TYP)
1.00
1.00
NEW PIECE OF 7075-T6 EXTRUDED LONGERON AT SECTION BETWEEN FRAMES.
0.90
0.125R
CHAMFER TO FIT
0.95
① REPAIR MEMBER (TWO REQUIRED) 0.094 7075-T6 EXTRUSION (FORM IN O CONDITION)

Figure 59-45. Lower 28 Degree Longeron Repair (Fuselage Stations 608 to 682)

Page 59-95/59-96 deleted.

SECTION 59.5

LANDING GEAR GROUP

Contents Page

GENERAL 59-97

REPAIRS 59-97

MAIN LANDING GEAR CYLINDER REPAIR 59-97

NOSE LANDING GEAR CYLINDER REWORK 59-97

59.5.1 GENERAL. See Section 32.0 for a complete description of the landing gear and landing gear doors. See Section 52.5 for the detail component material and identification figures to aid in establishing the detail part and the applicable repair figures in section 59.8.

59.5.2 REPAIRS. Refer to figures in Section 59.8 for negligible damage and typical repairs applicable to the landing gear doors.

59.5.3 MAIN LANDING GEAR CYLINDER REPAIR. If corrosion occurs in the lower end of the main landing gear cylinders at the lower bearing contact area, blend out corroded spots locally with Alodine 1200S nitric acid solution together with 400 grit abrasive cloth. Material removal shall be limited as follows:

a. Maximum depth -0.005 inch
b. Maximum diameter - 1.000 inch
c. No more than 4 places in any plane perpendicular (cross section) to the cylinder center line.

Treat reworked surfaces per paragraph 51.7.3.9.

If the above limits do not permit complete cleanup of corrosion, see figure 59-80, for permissible rework of the cylinder. This rework requires a new lower bearing shown in figure 59-80.

59.5.4 NOSE LANDING GEAR CYLINDER REWORK. If corrosion occurs in the lower end of the nose landing gear cylinder at the lower bearing contact area, blend out corroded spots locally with Alodine 1200S nitric acid solution together with 400 grit abrasive cloth. Material removal shall be limited as follows:

a. Maximum depth - 0.005 inch
b. Maximum diameter - 1.00 inch
c. No more than 4 places in any plane perpendicular (cross section) to the cylinder centerline.

Treat reworked surfaces per paragraph 51.7.3.9.

If the above limits do not permit complete cleanup of corrosion, see figure 59-81 for permissible rework of the cylinder. This rework requires a new lower bearing shown in figure 59-81.

SECTION 59.6

NACELLES

Contents Page

GENERAL 59-99

REPAIRS 59-99

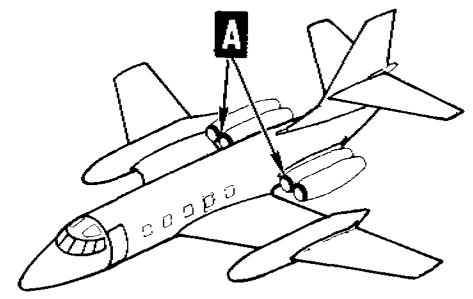
FIGURES

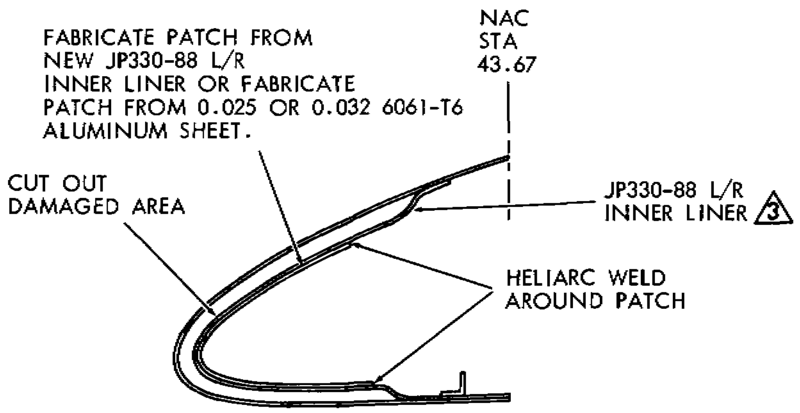
59-46 Nose Lip Assembly Air Outlet Inner Liner Repair 59-101

59-47 Engine Forward Inlet Duct Splitter Vane Crack Limitations and Repair 59-102

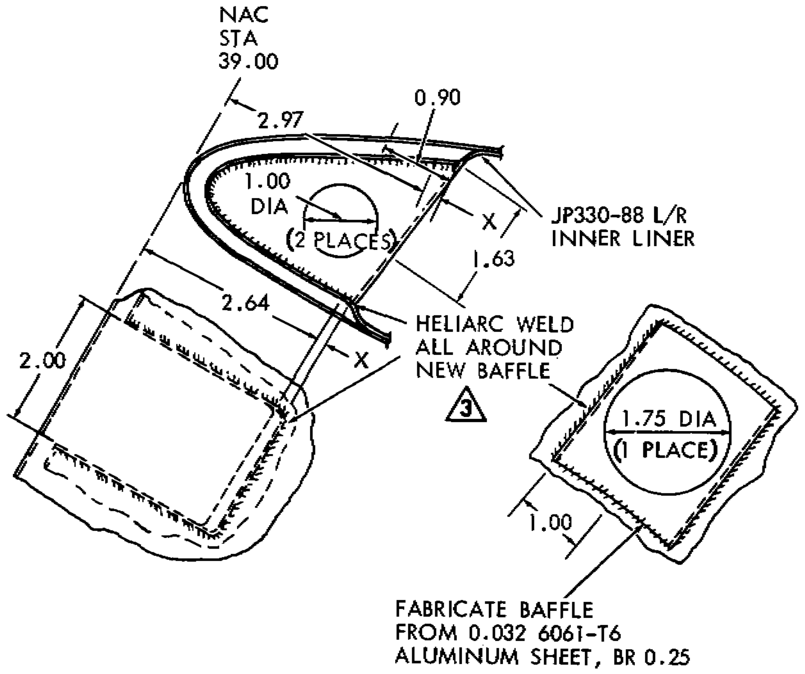
59.6.1 GENERAL. See Chapter 54 for a complete description of the nacelles and nacelle structure. Chapter 54 also contains detail component material and identification figures to aid in establishing the applicable repair figures in Section 59.8. Also, refer to Chapter 78 for thrust reverser description and minor repair.

59.6.2 REPAIRS. Figures 59-49 through 59-66 give negligible damage limits and typical repairs applicable to the pylon, pylon fairing, and nacelles. See figures 78-13, 78-14, 78-15, and 78-17 for thrust reverser minor repairs.




FABRICATE PATCH FROM NEW JP330-88 L/R INNER LINER OR FABRICATE PATCH FROM 0.025 OR 0.032 6061-T6 ALUMINUM SHEET.
NAC STA 43.67
CUT OUT DAMAGED AREA
JP330-88 L/R INNER LINER △3
HELIARC WELD AROUND PATCH

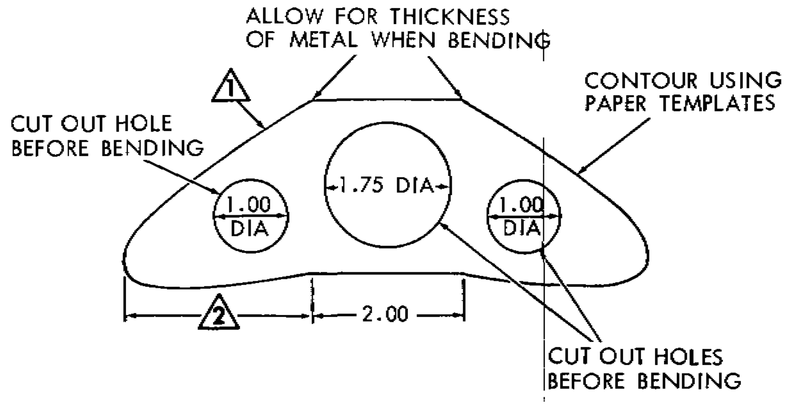
A



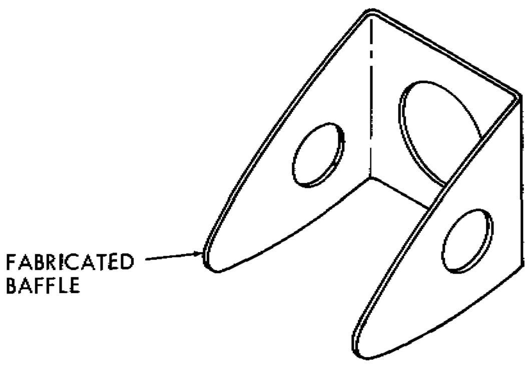
B

PROCEDURE FOR REPAIR OF EROSION DAMAGE TO NOSE LIP ASSEMBLIES BY ADDING A REINFORCING BAFFLE TO STRENGTHEN UNSUPPORTED EDGES OF AIR OUTLET INNER LINER ASSEMBLIES.

1. REMOVE THE NOSE LIP ASSEMBLY FROM THE AIRPLANE.
2. IF DAMAGE TO THE JP330-88 L/R INNER LINER IS NOT EXTENSIVE ENOUGH TO REQUIRE A CONTOURED SHEET, CUT OUT THE DAMAGED AREA AND FABRICATE A PATCH FROM 0.025 OR 0.032 INCH 6061-T6 ALUMINUM SHEET TO COVER THE DAMAGED AREA. SECURE THE PATCH BY HELIARC WELDING. USE A5356 WELDING ROD. SEE DETAIL A.
3. WHEN DAMAGE TO THE JP330-88 L/R INNER LINER IS EXTENSIVE ENOUGH TO REQUIRE A CONTOURED SHEET, CUT OUT THE DAMAGED AREA AND FABRICATE A PATCH FROM A NEW JP330-88 L/R INNER LINER TO COVER THE DAMAGED AREA. SECURE THE PATCH BY HELIARC WELDING. USE A5356 WELDING ROD. SEE DETAIL A.
4. MAKE A PAPER REINFORCING BAFFLE TEMPLATE TO FIT THE INNER LINER CUTOUT CONTOUR. SEE DETAILS B AND C.
5. LAYOUT A SLIGHTLY OVERSIZE REINFORCING BAFFLE ON 0.032 INCH 6061-T6 ALUMINUM SHEET. CUT OUT THREE HOLES. FILE THE BAFFLE TO FIT AFTER BENDING. SEE DETAILS B AND C.


ALLOW FOR THICKNESS OF METAL WHEN BENDING
△1
CONTOUR USING PAPER TEMPLATES
CUT OUT HOLE BEFORE BENDING
1.00 DIA
1.75 DIA
1.00 DIA
△2
2.00
CUT OUT HOLES BEFORE BENDING

C



D

6. BEND THE REINFORCING BAFFLE TO FIT THE INNER CUTOUT. SEE DETAILS B AND D. CUT OUT UNSUPPORTED EDGE OF INNER LINER TO THE DISTANCE SHOWN BY DIMENSION "X." SEE DETAIL B.
7. SECURE THE REINFORCING BAFFLE TO THE INNER LINER BY HELIARC WELDING AROUND THE ENTIRE BAFFLE. USE A5356 WELDING ROD.
8. RE-INSTALL THE NOSE LIP ASSEMBLY ON THE AIRPLANE.

NOTE

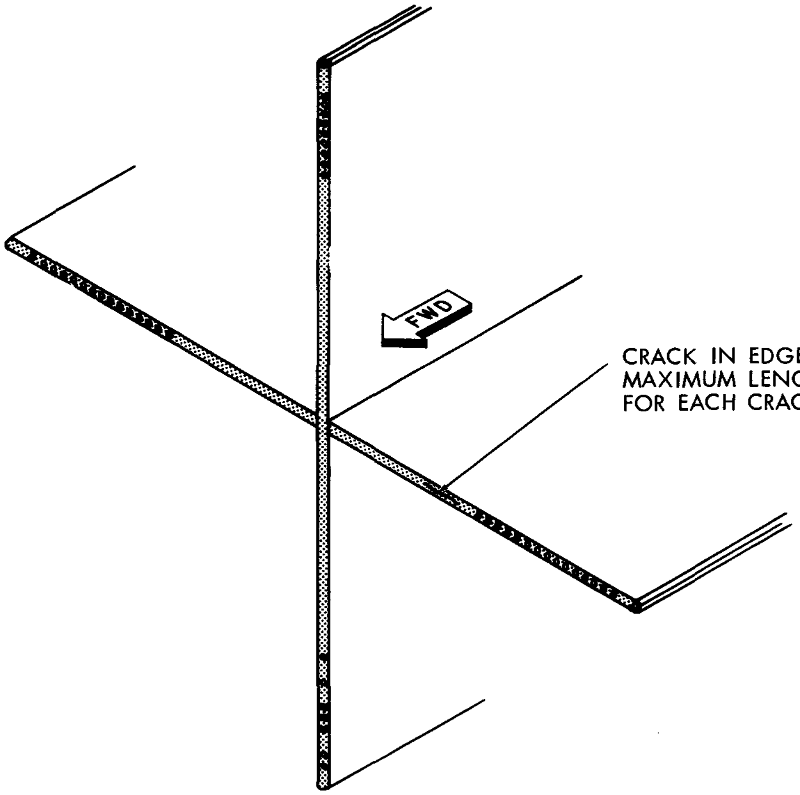
△1 FABRICATE BAFFLE SLIGHTLY OVERSIZE AND FILE TO FIT.

△2 DIMENSION AS SHOWN IN DETAIL B AND AS REQUIRED BY PAPER TEMPLATE.

△3 MATERIAL IN AREA OF WELDS ACCEPTABLE IN WELDED CONDITION.

X – INCREASE CUTOUT SIZE AS REQUIRED TO REMOVE UNSUPPORTED EDGE BEFORE FABRICATING PAPER TEMPLATE.

Figure 59-46. Nose Lip Assembly Air Outlet Inner Liner Repair



CRACK IN EDGE WELD - MAXIMUM LENGTH ALLOWED FOR EACH CRACK IS 2.0 INCHES

CRACK LIMITS AT PANEL FORWARD EDGE

Figure 59-47. Engine Forward Inlet Duct Splitter Vane Crack Limitations and Repair

(Figure 59-48 Deleted.)

SECTION 59.7

DAMAGE DUE TO LANDING GEAR FAILURE

Contents Page

DESCRIPTION .59-103
General .59-103
Nose Landing Gear Retracted (Forward Collapse) .59-103
Nose Landing Gear Aft Collapse .59-103
Both Main Landing Gear Retracted (Inward Collapse) .59-103
Both Main Landing Gear Outward Collapse .59-103
All Landing Gear Retracted .59-104
Tire Blowout .59-104

59.7.1 DESCRIPTION.

59.7.2 GENERAL. Damage to the airframe can result if the landing gear is collapsed when the aircraft is at rest or is being towed, or if the landing gear is overloaded when the aircraft is landed, or if the aircraft is landed with the landing gear retracted. The extent of damage varies with the ground speed, the sinking rate, and the attitude of the aircraft on contact with the ground. The following paragraphs give the type of landing gear failure, the areas of the aircraft likely to be damaged, and the reference to the applicable component repair index illustrations. Reference should first be made to Section 59-1, (which describes repair capabilities) before investigating specific damage.

59.7.3 NOSE LANDING GEAR RETRACTED (FORWARD COLLAPSE). Investigate damage to the following areas:

a. Fuselage pressure nose skin and structure, fuselage stations 94 to 158 (see figures 53-1 and 53-3).

b. Forward fuselage skin and structure below waterline 73, fuselage stations 158 to 270.5 (see figures 53-3, 53-4, and 53-5).

c. Nose landing gear doors (see figure 52-12).

d. Nose landing gear (see figure 52-11).

59.7.4 NOSE LANDING GEAR AFT COLLAPSE. Investigate damage to the following areas:

a. Fuselage pressure nose skin and structure, fuselage stations 94 to 158 (see figures 53-1 and 53-3).

b. Forward fuselage skin and structure below waterline 73, fuselage stations 158 to 270.5 (see figures 53-3, 53-4, and 53-5).

c. Nose landing gear doors (see figure 52-12).

d. Center fuselage skin and structure below waterline 73, fuselage stations 270.5 to 330 (see figures 53-3, 53-6, 53-7, and 53-8).

e. Nose landing gear (see figure 52-11).

59.7.5 BOTH MAIN LANDING GEARS RETRACTED (INWARD COLLAPSE). Investigate damage to the following areas:

a. Aft fuselage skin and structure, fuselage stations 570 to 800 (see figures 53-3 and 53-9).

b. Center fuselage skin and structure below waterline 83, fuselage stations 450 to 500 (see figures 53-3, 53-6, 53-7, and 53-8).

c. Left and right wing tips (see figures 57-2 and 57-3).

d. Left and right external wing tanks and fillets, tank stations 149 to 278 (see figures 57-7 and 57-9).

e. Left and right ailerons (see figure 57-6).

f. Left and right wing trailing edge flaps (see figure 57-8).

g. Left and right wing skin and structure, wing stations 154 to 318 (see figures 57-2 and 57-3).

h. Left and right main landing gear doors (see figure 52-13).

i. Left and right main landing gear (see figure 52-11).

j. Wing attach fittings (see figures 57-3 and 53-7 and 53-8).

59.7.6 BOTH MAIN LANDING GEARS OUTWARD COLLAPSE. Investigate damage to the following areas:

a. Aft fuselage skin and structure, fuselage stations 570 to 800 (see figures 53-3 and 53-9).

b. Center fuselage skin and structure below waterline 83, fuselage stations 450 to 500 (see figures 53-3, 53-6, 53-7, and 53-8).

59.7.6 (Cont).

c. Left and right wing tips (see figures 57-2 and 57-3).

d. Left and right external wing tanks and fillets, tank stations 149 to 278 (see figures 57-7 and 57-9).

e. Left and right ailerons (see figure 57-6).

f. Left and right wing trailing edge flaps (see figure 57-8).

g. Left and right wing skin and structure, wing station 41.5 to 318 (see figures 57-2 and 57-3).

h. Left and right main landing gear doors (see figure 52-13).

i. Left and right main landing gear (see figure 52-11).

59.7.7 ALL LANDING GEAR RETRACTED. Investigate damage to the following areas:

a. Forward fuselage skin and structure below waterline 73, fuselage stations 164 to 270.5 (see figures 53-3, 53-4, and 53-5).

b. Nose landing gear doors (see figure 52-12).

c. Nose landing gear (see figure 52-11).

d. Center fuselage skin and structure below waterline 73, fuselage stations 270.5 to 570 (see figures 53-3, 53-6, 53-7, and 53-8).

e. Aft fuselage skin and structure, fuselage stations 570 to 800 (see figures 53-6 and 53-9).

f. Left and right wing tips (see figures 57-2 and 57-3).

g. Left and right external wing tanks and fillets, tank stations 0 to 278 (see figures 57-7 and 57-9).

h. Left and right ailerons (see figure 57-6).

i. Left and right wing-to-fuselage fillets (see figure 57-7).

j. Left and right wing skin and structure, wing stations 41.5 to 318 (see figures 57-2 and 57-3).

k. Left and right main landing gear doors (see figure 52-13).

l. Left and right main landing gear (see figure 52-11).

m. Left and right wing trailing edge flaps (see figure 57-8).

59.7.8 TIRE BLOWOUT. Inspect the airframe skin, landing gear doors, and exposed wheel well structure in the immediate vicinity of the blown tire for minor damage caused by strips and pieces of the tire hitting the airframe.

SECTION 59.8

TYPICAL REPAIRS AND APPLICATIONS

Contents Page

REWORK OF SKIN SCRATCHES 59-106
- Scratch Penetration Test for 2024 Clad Aluminum Alloy 59-106
- Scratch Penetration Test for 7075 Clad Aluminum Alloy 59-107
- Test Analysis 59-107
- Removing Skin Scratches 59-107
- Determining Scratch Depth 59-108
- Analysis of Reworked Area 59-108

LOCAL DEPRESSIONS OR DENTS 59-108
- Fabrication of Contour Tool 59-108
- Depth Measurement of Contoured Skin Surfaces 59-108
- Depth Measurement of Flat Skin Surfaces 59-108

NEGLIGIBLE DAMAGE LIMITS 59-108

TYPICAL REPAIRS 59-108
- Skin Repairs 59-108
- Web Repairs 59-108
- Formed and Extruded Member Repairs 59-108
- Lightening Hole Flange Repairs 59-108
- Flange Repairs 59-108
- Honeycomb Repairs 59-108
- Formed Aluminum Alloy Repairs 59-108
- Defective Spotweld Repairs 59-108
- Cracked Rivet Hole Repairs 59-109
- Tubing Surface Repairs 59-109
- Elevator Horn Fitting Repairs 59-109
- Main Landing Gear Cylinder Repairs 59-109
- Nose Landing Gear Cylinder Repairs 59-109

FIGURES

59-49 Negligible Damage Limits for Skin Panels and Bulkhead Webs 59-110
59-50 Negligible Damage Limits for Extruded Sections 59-112
59-51 Negligible Damage Limits for Formed Sections 59-113
59-52 Negligible Damage Limits for Beams, Ribs, and Bulkheads 59-114
59-53 Typical Fuselage Skin Scratch Repair at Frames and Cutouts 59-116
59-54 Fuselage Pressure Skin Scratch Repair in Unobstructed Areas 59-117
59-55 Typical Flush Skin Repair for Unobstructed Areas 59-119
59-56 Typical Flush Skin Repair at Panel Edges 59-121
59-57 Typical One-Time Ferry Flight Repair 59-122
59-58 Typical Temporary Skin Repair 59-124
59-59 Typical Web Repair 59-126
59-60 Damaged Web - Secondary Structure - Doubler Repair 59-128
59-61 Typical Fuselage Ring "L" Segment Repair 59-129
59-62 Typical Rib and Former Repairs 59-130
59-63 Typical Formed and Extruded "Z" Repair 59-133
59-64 Typical Extruded "T" Repair 59-134
59-65 Typical Extruded Angle Repair 59-135
59-66 Lightening Hole Flange Crack - Secondary Structure 59-136
59-67 Cracked Flange - Beaded Lightening Hole - Doubler Repair 59-137
59-68 Flange Bend Crack - Inside Doubler Repair 59-138

FIGURES

59-69 Flange Bend Crack - Outside Doubler Repair 59-139
59-70 Corner Crack - Formed Flange - Cut-Out Repair 59-139
59-71 Corner Crack - Formed Flange - Doubler Repair 59-140
59-72 Damaged Flange - Replacement Repair 59-140
59-73 Honeycomb Repair, JL371 Door 59-141
59-74 Typical Honeycomb Panel Repair 59-142
59-75 Typical Repair for Formed Aluminum Alloy Sections 59-143
59-76 Defective Spotweld - Rivet Repair 59-144
59-77 Cracked Rivet Hole - Plug Repair 59-145
59-78 Surface Damage - Tubing 59-145
59-79 Elevator Horn Fitting (JE190-1) Repair 59-146
59-80 Main Landing Gear Cylinder Rework 59-147
59-81 Nose Landing Gear Cylinder Rework 59-149

59.8.1 REWORK OF SKIN SCRATCHES. All scratches must be located by inspection and must be reworked. Unreworked scratches result in stress concentration which increases the likelihood of failure. This is of particular importance when the skin is subjected to the added loads of pressurization.

59.8.1.1 SCRATCH PENETRATION TEST FOR 2024 CLAD ALUMINUM ALLOY. Use the following procedure to determine if a scratch has penetrated through the cladding on 2024 material.

WARNING

Use impervious clothing, gloves, face shield and other appropriate protective clothing necessary to prevent any possibility of skin contact with any chemical or solution used in this procedure. Avoid breathing the vapors or fumes from the materials used in this procedure, and provide adequate ventilation.

a. Cool or warm the fuselage skin in the area of the scratch to bring its temperature between 15.6°C (60°F) and 26.7°C(80°F). Higher temperatures will cause the chemicals to act too fast, and lower temperatures will cause the chemicals to act too slowly or not at all.

b. Apply two strips of masking tape parallel to the scratch, about 3/16 inch apart, with the scratch between them. Mask any joints or rivets on which the chemicals might spill accidentally.

c. Clean the surface to be tested with trichlorethylene or an equivalent grease-removing solvent.

d. Prepare a caustic soda solution by dissolving 2 or 3 ounces of caustic soda (lye) flakes in 1/2 pint of water, or 1 pound of flakes to 1/2 gallon of water. An alternate method of mixing the solution is to obtain a mixture with a specific gravity between 1.22 and 1.32, inclusive. Use glass containers for mixing and storing.

WARNING

Caustic soda flakes and solutions are very harmful to the human skin. In case of accidental contact, immediately flood the affected area with water. If the eyes are affected, get medical aid immediately. Wear rubber gloves to protect the hands.

e. Using an eyedropper, place a drop of the caustic soda solution on the scratch area to be tested. Allow the solution to act on the aluminum skin for 60 seconds.

CAUTION

If the caustic soda solution enters any joint or rivet hole, remove the rivets and open the joint enough to flush the area with clean water, and dry. Apply anti-corrosion treatment according to the instructions in Section 51.7 replace the rivets and reseal, if required.

f. Any exposed area of 2024 core material will appear dark in contrast to the aluminum cladding, which will be brightened by the caustic soda solution.

g. If there is penetration through the cladding material, mark its length on a piece of the tape beside the scratch.

h. Flush the area with clean water.

i. Refer to the "Test Analysis" paragraph 59.8.1.3.

CAUTION

If the scratch is not to be treated immediately as outlined in paragraph 59.8.1.3, perform the anti-corrosion treatment according to the instructions in Section 51.7.

Page 59-106A/59-106B Deleted.

59.8.1.2 SCRATCH PENETRATION TEST FOR 7075 CLAD ALUMINUM ALLOY. Use the following procedure to determine if a scratch has penetrated through the cladding on 7075 material.

a. Cool or warm the skin in the area of the scratch to bring its temperature between 15.6°C (60°F) and 26.7°C (80°F). Higher temperature will cause the chemicals to act too fast and lower temperatures will cause the chemicals to act too slowly or not at all.

b. Apply two strips of masking tape parallel to the scratch, about 3 16 inch apart with the scratch between them. Mask any joints or rivets on which the chemicals might spill accidentally.

c. Clean the surface to be tested with trichlorethylene or an equivalent grease-removing solvent.

d. Using a cotton swab (throat swab or equivalent) or eyedropper, moisten the scratch area with concentrated hydrochloric acid. Cover the full length of the scratch, and make sure the acid is well down into the scratch.

WARNING

Hydrochloric acid is very harmful to human skin. In case of personal contact, immediately flood the affected area with water. If the eyes are affected, obtain medical aid immediately. Wear rubber gloves to protect the hands.

CAUTION

If the acid enters any joint or rivet hole, remove the rivets and open the joint enough to flush the area with clean water, and dry. Apply anti-corrosion treatment according to the instructions in Section 51.7; replace the rivets and reseal, if necessary.

e. Allow the acid to act for about 10 seconds; then wipe dry with a clean, soft cloth.

f. Any exposed area of the 7075 material will be dark, while the color of the aluminum cladding material will not be affected.

g. If there is penetration through the cladding material, mark its length on a piece of the tape beside the scratch.

h. Flush the area with clean water.

i. Refer to paragraph 59.8.1.3 for analysis of the test.

CAUTION

If the scratch is not to be treated immediately as directed in the test analysis, perform the anti-corrosion treatment according to the instructions in Section 51.7.

59.8.1.3 TEST ANALYSIS. If the scratch does not penetrate the cladding material, it may either be left as it is or be removed. (See paragraph 59.8.1.4.) If the scratch is removed, the test for penetration must be repeated. (See paragraph 59.8.1.1 or 59.8.1.2.)

CAUTION

Whether the scratch is smoothed out or not, the skin in the scratched area must be given the proper anti-corrosion treatment. (See Section 51.7.)

If the scratch does penetrate the cladding material, proceed as follows:

a. Smooth out the scratch as outlined in paragraph 59.8.1.4.

b. Determine the depth of the scratch as outlined in paragraph 59.8.1.5

c. See figure 59-32 and paragraph 59.4.1.2 for classification and disposition of the scratch.

d. Perform the proper anti-corrosion treatment as outlined in Section 51.7.

59.8.1.4 REMOVING SKIN SCRATCHES. Remove skin scratches with an impregnated rubber grinding wheel. Use a wheel 3 inches in diameter, 0.18 inch thick with a 0.09 inch radius on the edges, and impregnated with No. 70-fine grit. Observe the following precautions while smoothing out skin scratches:

a. Smooth to the bottom of the scratch and no deeper. Removal of more skin material than is necessary could lead to an unnecessary repair.

b. Check the 0.09 radius on the wheel frequently. Dress the wheel as necessary to maintain the radius. Use of the wheel tends to flatten it; this will cause unnecessary removal of skin material.

c. Use the wheel intermittently to prevent the skin from overheating. Overheating of the skin will cause it to lose the required strength.

d. When smoothing out a scratch that crosses or ends at a rivet, do not smooth into the dimpled or countersunk area. As a precaution, do not remove the rivet until the smoothing is completed.

59.8.1.5 DETERMINING SCRATCH DEPTH. After smoothing out the scratch, lay a straight edge across the smoothed area, and measure the depth with a depth micrometer.

CAUTION

The depth of the scratch is the distance from the original surface of the skin to the bottom of the scratch. Therefore, do not use an area that has previously been smoothed to support the straightedge. A dangerously thin spot could result without detection.

59.8.1.6 ANALYSIS OF REWORKED AREA. After determining the scratch depth and length, see figure 59-49 for negligible damage limits. If the scratch is negligible, the reworked area must be given the proper anticorrosion and refinishing treatment in accordance with the instructions in Section 51.7. If the scratch exceeds the negligible damage limits, make repairs in accordance with instructions in the applicable structural sections of this manual or in accordance with the typical skin repairs given in this section.

59.8.2 LOCAL DEPRESSIONS OR DENTS. Procedures for measuring the depths of local depressions or dents in contoured and flat skin surfaces are necessary to determine if the negligible damage limits of the depressions or dents are exceeded.

59.8.2.1 FABRICATION OF CONTOUR TOOL. Make a spline 0.09-inch thick, 0.50-inch wide, and 18-inches long (minimum) from a flexible plastic material. The spline is to aid in positioning a depth micrometer to obtain an accurate measurement of the depth of a local depression or dent.

59.8.2.2 DEPTH MEASUREMENT OF CONTOURED SKIN SURFACES. Use the following procedure to measure the depths of local depressions or dents on contoured skin surfaces:

a. Bend the spline flush against the contoured surface adjacent to the local depression or dent to be measured.

b. Measure the maximum depth of the local depression or dent with the depth micrometer held perpendicular to the contoured surface of the spline.

c. See figure 59-49 for permissible negligible damage limits.

59.8.2.3 DEPTH MEASUREMENT OF FLAT SKIN SURFACES. Use the following procedure to measure the depths of local depressions or dents on flat skin surfaces:

a. Place the spline flush against the flat surface adjacent to the local depression or dent being measured.

b. Measure the maximum depth of the local depression or dent with the depth micrometer held perpendicular to the surface of the spline.

c. See figure 59-49 for permissible negligible damage limits.

59.8.3 NEGLIGIBLE DAMAGE LIMITS. Typical negligible damage limits are given for skin panels, bulkhead webs, extruded members, and formed members in this section. (See figures 59-49 through 59-52.) These limits determine the maximum extent of damage permitted to a structural member or panel that can be corrected without adding a repair member. Specific negligible damage data will be found in the applicable structural group section. If required, the specific data will take precedence over the typical negligible damage data.

59.8.4 TYPICAL REPAIRS. Typical repairs may be accomplished individually, or may be combined with specific repairs for a major repair situation. Technique and material variations are permissible only to aid fabrication or installation of the typical repair, but the original strength and usefulness of the affected structural member or panel must be maintained.

59.8.4.1 SKIN REPAIRS. Typical permanent, temporary, and one-time ferry-flight skin repairs are given in this section. (See figures 59-53 through 59-58.) Temporary or one-time ferry-flight repairs must be used only at the discretion of the aircraft manufacturer. Temporary repairs must be replaced with permanent repairs at the next periodic inspection of the aircraft. One-time ferry-flight repairs must be replaced with permanent repairs before the aircraft is returned to service.

59.8.4.2 WEB REPAIRS. See figure 59-59 for typical web repairs.

59.8.4.3 FORMED AND EXTRUDED MEMBER REPAIRS. See figures 59-61 through 59-65 for typical formed and extruded member repairs.

59.8.4.4 LIGHTENING HOLE FLANGE REPAIRS. See figures 59-66 and 59-67 for typical lightening hole flange repairs.

59.8.4.5 FLANGE REPAIRS. See figures 59-68 through 59-72 for typical flange repairs.

59.8.4.6 HONEYCOMB REPAIRS. See figures 59-73 and 59-74 for typical honeycomb repairs

59.8.4.7 FORMED ALUMINUM ALLOY REPAIRS. See figure 59-75 for typical formed aluminum alloy repairs.

59.8.4.8 DEFECTIVE SPOTWELD REPAIRS. See figure 59-76 for typical spotweld repairs.

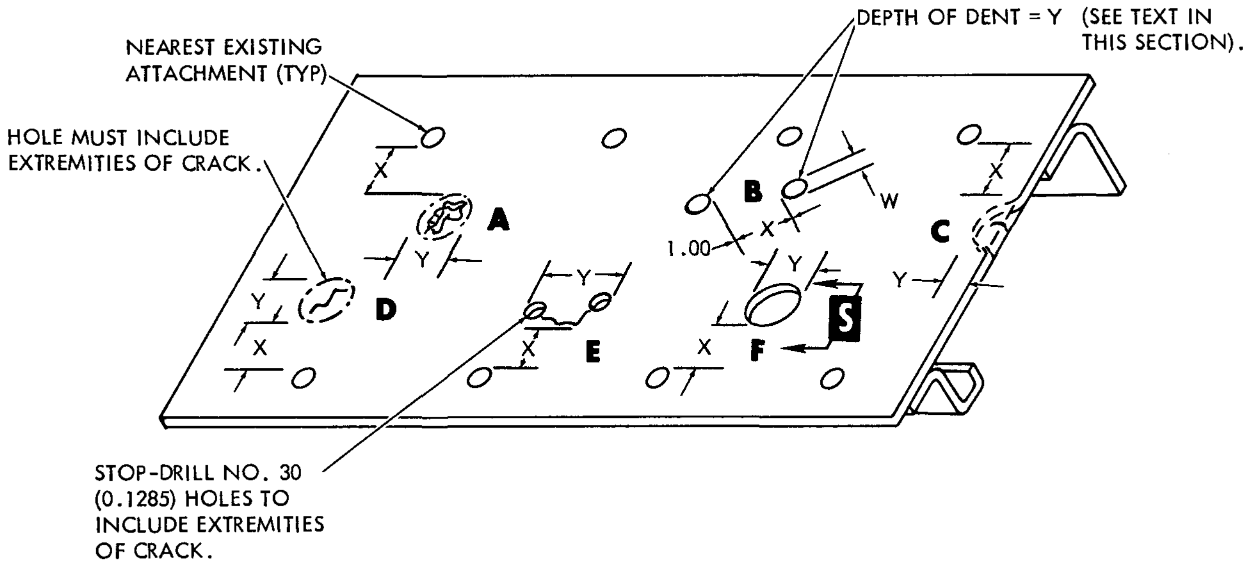
59.8.4.9 CRACKED RIVET HOLE REPAIRS. See figure 59-77 for typical cracked rivet repairs.

59.8.4.10 TUBING SURFACE REPAIRS. See figure 59-78 for typical tubing surface repairs.

59.8.4.11 ELEVATOR HORN FITTING REPAIRS. See figure 59-79 for elevator horn fitting repair.

59.8.4.12 MAIN LANDING GEAR CYLINDER REPAIRS. See figure 59-80 for typical main landing gear cylinder repair.

59.8.4.13 NOSE LANDING GEAR CYLINDER REPAIRS. See figure 59-81 for typical nose landing gear cylinder repair.


DEPTH OF DENT = Y (SEE TEXT IN THIS SECTION).
NEAREST EXISTING ATTACHMENT (TYP)
HOLE MUST INCLUDE EXTREMITIES OF CRACK.
X
A
Y
B
W
X
1.00
C
Y
Y
D
Y
X
E
X
X
F
S
STOP-DRILL NO. 30 (0.1285) HOLES TO INCLUDE EXTREMITIES OF CRACK.

CODE

1. "X" IS MINIMUM DISTANCE FROM EDGE OF ATTACHMENTS OR FROM EDGE OF ADJACENT DAMAGE.
2. "Y" IS MAXIMUM EXTENT OF CLEANED-UP DAMAGE.
3. "Z" IS MINIMUM THICKNESS OF ORIGINAL SKIN.
4. "W" IS THE MAXIMUM DIAMETER, WIDTH, OR LENGTH OF LOCAL DEPRESSION OR DENTS.

TYPE OF DAMAGE

A = SKIN PUNCTURE CLEANED UP TO A ROUND HOLE.

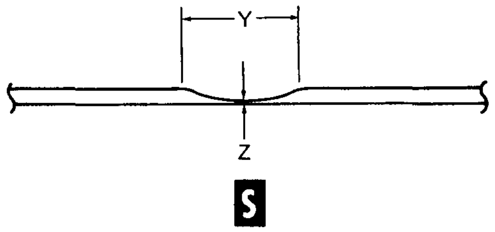
B = SMOOTH DENT OR DEPRESSION FREE FROM ABRASIONS, SCRATCHES, CRACKS, OR SHARP BENDS.

C = DAMAGE AT EDGE OF SHEET CLEANED UP TO A SMOOTH CONTOUR.

D = CRACK CLEANED UP TO A ROUND HOLE.

E = CRACK STOP-DRILLED AT EXTREMITIES.

F = CORRODED AREA CLEANED UP TO A ROUND HOLE EXCEPT AS NOTED IN VIEW **S**



WHERE "Z" IS NOT ZERO FOR A COMPONENT IN THE TABLE ON SHEET 2 OF THIS FIGURE, REMOVE CORROSION CAREFULLY AND SMOOTH OUT THE AREA, LEAVING NO SHARP EDGES, SCRATCHES, OR CUTS. MAKE DEPTH MEASUREMENTS WITH A SPLINE AND DEPTH MICROMETER. (SEE TEXT IN THIS SECTION.)

Figure 59-49. Negligible Damage Limits for Skin Panels and Bulkhead Webs (Sheet 1 of 2)

ALLOWABLE EXTENT OF DAMAGE

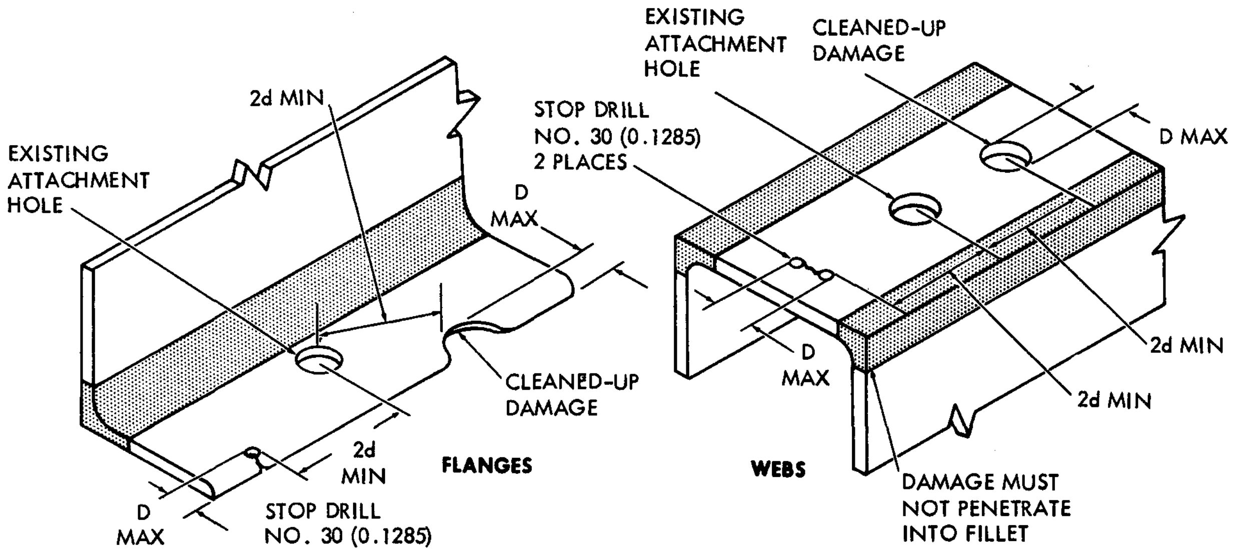
LOCATION OF DAMAGE	TYPE OF DAMAGE												
	A		B		C		D		E		F		
	X	Y	W	Y	X	Y	X	Y	X	Y	X	Y	Z
WING-TO-FUSELAGE FAIRINGS △1	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
WING-TO-TANK FAIRINGS △1	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
WING LEADING EDGE △2	NONE		1.25	0.08	NONE		NONE		NONE		NONE		
WING LEADING EDGE FLAP △2	NONE		1.25	0.08	NONE		NONE		NONE		NONE		
EXTERNAL WING TANK △3	1.5	1.5	1.25	0.06	1.0	0.75	1.5	1.5	1.5	1.5	1.5	1.5	0.0
WING BOX SECTION	NONE		1.0	0.06	NONE		NONE		NONE		1.5	1.0	△4
WING TRAILING EDGE	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
WING TIP △1	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
WING TRAILING EDGE FLAPS	1.5	1.5	1.25	0.06	1.0	0.75	1.5	1.5	1.5	1.5	1.5	1.5	0.0
AILERON	1.5	1.5	1.25	0.06	1.0	0.75	1.5	1.5	1.5	1.5	1.5	1.5	0.0
EMPENNAGE FAIRINGS △5	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
HORIZONTAL STABILIZER LEADING EDGE △2	NONE		1.25	0.08	NONE		NONE		NONE		NONE		
HORIZONTAL STABILIZER BOX SECTION	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
HORIZONTAL STABILIZER TRAILING EDGE	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
HORIZONTAL STABILIZER TIP	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
ELEVATOR	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
VERTICAL STABILIZER LEADING EDGE △2	NONE		1.25	0.08	NONE		NONE		NONE		NONE		
VERTICAL STABILIZER BOX SECTION	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
VERTICAL STABILIZER TRAILING EDGE	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
VERTICAL STABILIZER TIP	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
RUDDER	1.5	1.5	1.25	0.06	1.0	0.75	1.5	1.5	1.5	1.5	1.5	1.5	0.0
RADOME	NONE		NONE		NONE		NONE		NONE		NONE		
PRESSURIZED FUSELAGE NOSE SHELL (FUS STA 115 TO 164)	NONE		1.00	0.04	NONE		NONE		NONE		NONE		
PRESSURIZED FORWARD FUSELAGE (FUS STA 164 TO 270)	NONE		1.00	0.04	NONE		NONE		NONE		NONE		
PRESSURIZED CENTER FUSELAGE (FUS STA 270 TO 570)	NONE		1.00	0.04	NONE		NONE		NONE		NONE		
UNPRESSURIZED AFT FUSELAGE (FUS STA 570 TO 760)	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
UNPRESSURIZED FUSELAGE TAIL CONE (FUS STA 760 TO 800)	1.5	1.0	1.25	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
PRESSURIZED FUSELAGE MAIN ENTRANCE DOOR	NONE		1.00	0.04	NONE		NONE		NONE		NONE		
PRESSURIZED FUSELAGE ACCESS PANELS	NONE		1.00	0.04	NONE		NONE		NONE		NONE		
UNPRESSURIZED FUSELAGE ACCESS PANELS	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
NOSE LANDING GEAR DOORS	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
MAIN LANDING GEAR DOORS	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
NACELLE PYLONS	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0
NACELLES	1.5	1.0	1.5	0.06	1.0	0.75	1.5	1.0	1.5	1.0	1.5	1.0	0.0

NOTE

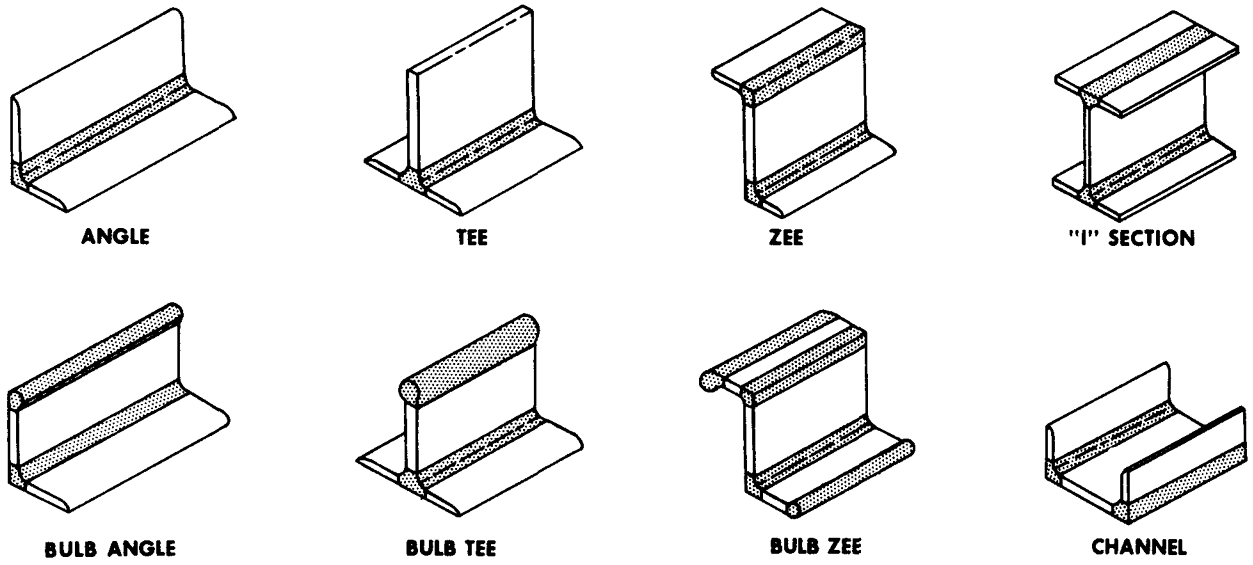
△1 NO "B" DAMAGE ALLOWED IN FIBERGLAS.
△2 DOES NOT APPLY TO DE-ICER BOOTS.
△3 ONLY "B" DAMAGE PERMITTED IN FUEL SECTIONS.
△4 "Z" DIMENSION, (REMAINING THICKNESS), NOT TO BE LESS THAN 50 PERCENT OF REFERENCE THICKNESS (t_{sk}) AT THE SUBJECT STATION. SEE FIGURES SEE FIGURES 57-4 AND 57-5
△5 DOES NOT APPLY TO FUEL JETTISON STRUCTURE.

6. ALTHOUGH THE TABLE PERMITS HOLES IN THE EXTERIOR COVERING OF THE AIRCRAFT, IT IS RECOMMENDED THAT ALL SUCH HOLES BE COVERED WITH PATCHES TO AVOID THE ENTRY OF MOISTURE AND FOREIGN MATERIAL INTO THE AIRCRAFT INTERIOR AND TO AVOID ADVERSE AERODYNAMIC EFFECTS.
7. BECAUSE OF THE COMPLEXITY OF MODERN AIRCRAFT AND THE HIGHLY INDIVIDUAL NATURE OF MOST DAMAGE SITUATIONS, EACH SPECIFIC CASE OF AIRFRAME REPAIR SHOULD BE CONSIDERED FIRST IN TERMS OF ITS INDIVIDUAL REQUIREMENTS AND PARTICULARS. FOR THIS REASON, THE INFORMATION CONTAINED HEREIN IS OF A VERY GENERAL NATURE AND REFERS ONLY TO MINOR CASES OF RELATIVELY TYPICAL DAMAGE. IF DAMAGE IS MORE SEVERE THAN IN THOSE CASES GIVEN IN THIS MANUAL, CONSULT A F.A.A. DESIGNATED ENGINEERING REPRESENTATIVE (D.E.R.).

Figure 59-49. Negligible Damage Limits for Skin Panels and Bulkhead Webs (Sheet 2 of 2)



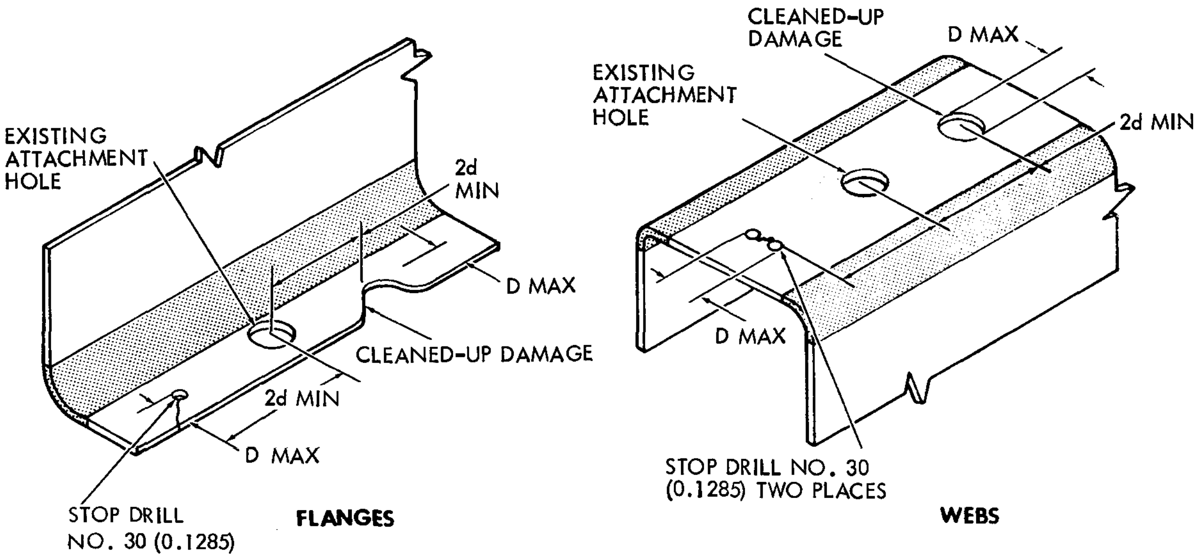
THE ABOVE DAMAGE RESTRICTIONS APPLY TO THE SECTIONS SHOWN BELOW
NO DAMAGE PERMITTED IN AREAS INDICATED ▨



NOTE

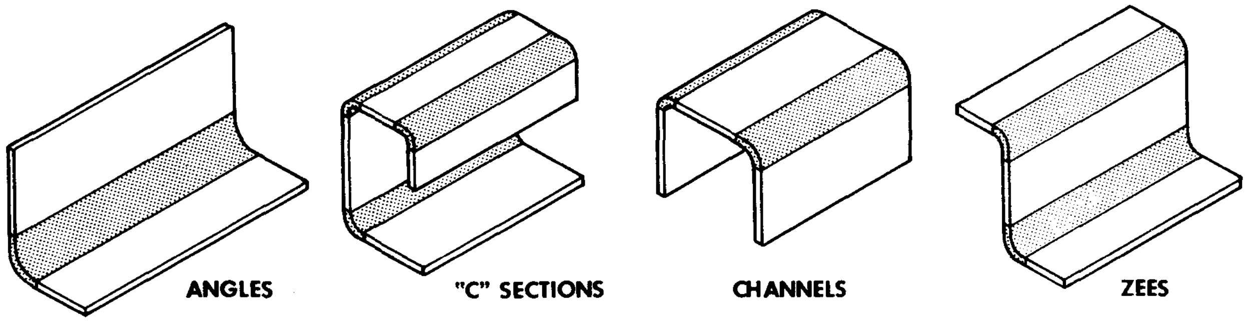
1. DAMAGE PERMITTED TO ONLY ONE COMPONENT IN ANY ONE CROSS SECTION. (COMPONENTS ARE UNSHADED LENGTHS).
2. NO REDUCTION IN SECTION PERMITTED IN COMPONENTS WHERE ORIGINAL DOES NOT CONTAIN ATTACHMENT HOLES.
3. ALL DAMAGE SHALL BE CLEANED UP TO A SMOOTH RADIUS OR STOP-DRILLED AS SHOWN.
4. "D" DENOTES THE DIAMETER OF THE LARGEST EXISTING ATTACHMENT HOLE IN THE DAMAGED COMPONENT.
5. "d" DENOTES THE DIAMETER OF THE ATTACHMENT HOLE ADJACENT TO THE DAMAGE.
6. DENTS PERMITTED TO A DEPTH EQUAL TO "t" (THICKNESS OF THE MATERIAL).
7. NICKS AND SCRATCHES MUST BE SMOOTHED AND FAIRED INTO THE ADJACENT MATERIAL. REDUCTION IN CROSS-SECTIONAL AREA MUST NOT EXCEED D X t SQ IN.

Figure 59-50. Negligible Damage Limits for Extruded Sections

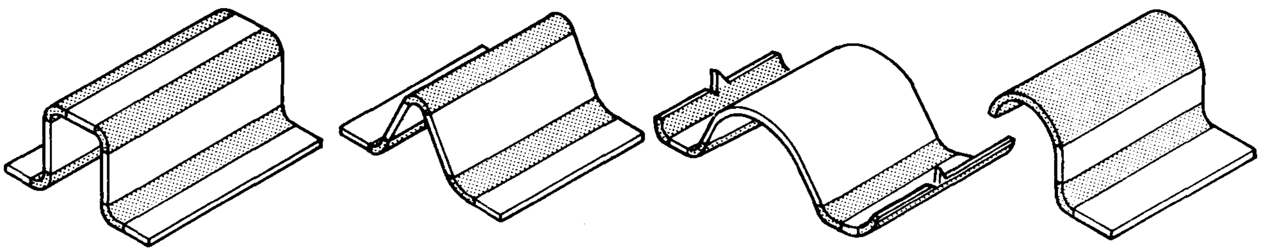


FLANGES **WEBS**

THE ABOVE DAMAGE RESTRICTIONS APPLY TO THE SECTIONS SHOWN BELOW.
NO DAMAGE PERMITTED IN AREAS INDICATED ▨



ANGLES **"C" SECTIONS** **CHANNELS** **ZEES**

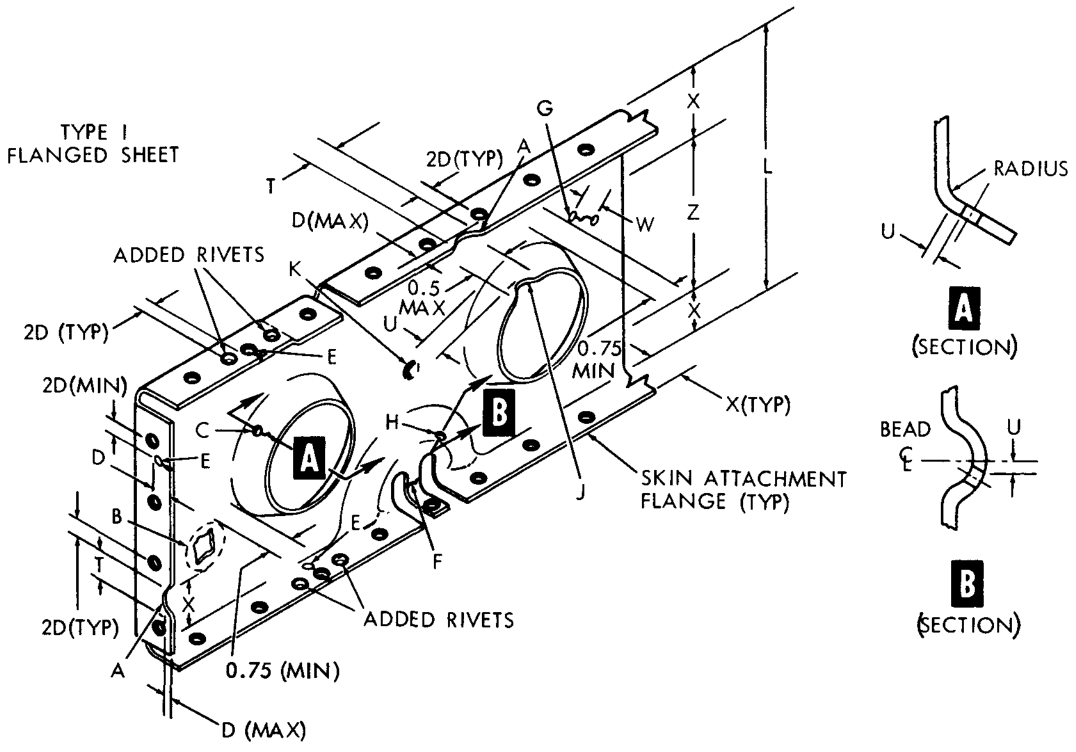


HAT SECTIONS **"V" SECTIONS** **CORRUGATIONS** **"J" SECTIONS**

NOTE

1. DAMAGE PERMITTED TO ONLY ONE COMPONENT (UNSHADED AREA) IN ANY ONE CROSS SECTION.
2. NO REDUCTION IN SECTION PERMITTED IN COMPONENTS WHERE ORIGINAL DOES NOT CONTAIN ATTACHMENT HOLES.
3. ALL DAMAGE SHALL BE CLEANED UP TO A SMOOTH RADIUS OR STOP-DRILLED AS SHOWN.
4. "D" DENOTES THE DIAMETER OF THE LARGEST EXISTING ATTACHMENT HOLE IN THE DAMAGED COMPONENT.
5. "d" DENOTES THE DIAMETER OF THE ATTACHMENT HOLE ADJACENT TO THE DAMAGE.
6. DENTS PERMITTED TO A DEPTH EQUAL TO "t" (THICKNESS OF THE MATERIAL).
7. NICKS AND SCRATCHES MUST BE SMOOTHED AND FAIRED INTO THE ADJACENT MATERIAL. REDUCTION IN CROSS SECTIONAL AREA MUST NOT EXCEED D X t SQ IN.

Figure 59-51. Negligible Damage Limits for Formed Sections



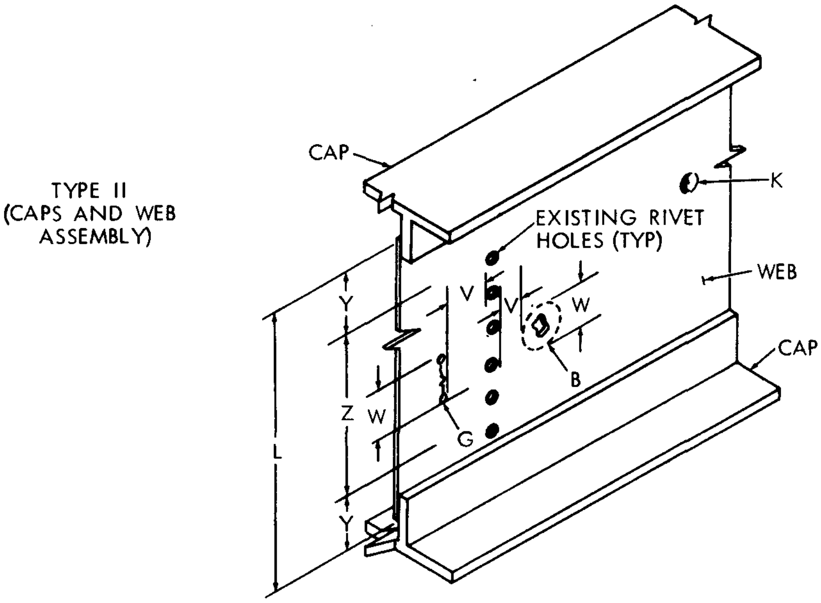


Figure 59-52. Negligible Damage Limits for Beams, Ribs, and Bulkheads (Sheet 1 of 2)

DAMAGE IDENTIFICATION AND DISPOSITION

A	NICK IN FLANGE EDGE - CLEAN TO A FAIRED OUTLINE.
B	PERFORATION IN WEB - CLEAN TO A SMOOTH HOLE.
C	CRACK IN LIGHTENING HOLE FLANGE - STOP-DRILL.
D	DIAMETER OF LARGEST ADJACENT ATTACHMENT
E	CRACK IN SKIN FLANGE OR END FLANGE - STOP-DRILL IF CRACK DOES NOT END AT EXISTING RIVET HOLE. CRACK FROM RIVET HOLE MUST NOT EXTEND INTO FLANGE RADIUS. INSERT AN ADDITIONAL RIVET ON EACH SIDE OF CRACK EXTENDING FROM EXISTING RIVET HOLES AS INDICATED. (CRACKS WITHIN 2D OF EXISTING RIVETS ARE TREATED AS IF FROM RIVET HOLE.)
F	CRACK IN ATTACHMENT CLIP - REPLACE CLIP WITH ONE OF SAME MATERIAL AND DIMENSIONS. ATTACH TO WEB WITH 3 OF SAME TYPE AND DIAMETER ATTACHMENTS AS ORIGINAL THROUGH CLIP USE 4D MINIMUM SPACING.
G	CRACK IN "EFFECTIVE WEB" (NOT ALLOWED IN "X" OR "Y") - STOP-DRILL BOTH ENDS
H	CRACK INTO BEAD OVER CUTOUT - STOP-DRILL.
J	NICK IN LIGHTENING HOLE FLANGE - CLEAN TO A FAIRED OUTLINE.
K	DENT IN WEB - TREAT AS B "PERFORATION IN WEB" IF DENT IS NOT FREE FROM SHARP CORNERS OR ABRASIONS AND/OR DEPTH OF DENT EXCEEDS WEB THICKNESS.

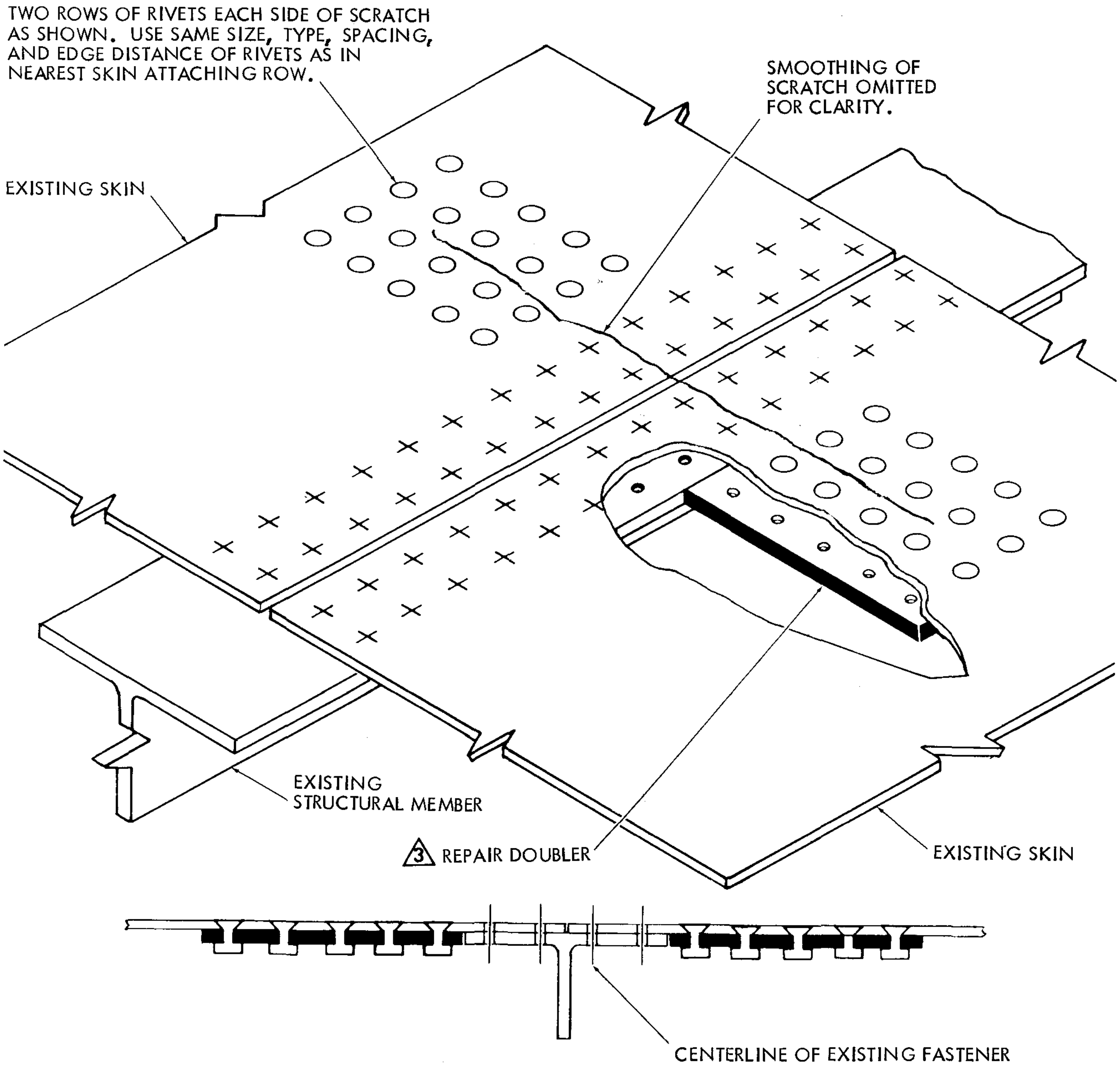
DIMENSIONS CODE

D	DIAMETER OF LARGEST ADJACENT ATTACHMENT.
L	WEB HEIGHT
T	EXISTING ATTACHMENT SPACING MINUS 4D. (MAXIMUM ALLOWED.)
U	0.10 FROM END OF RADIUS AS SHOWN IN SECTION A OR 0.10 FROM ℄ OF BEAD AS SHOWN IN SECTION B. (MINIMUM ALLOWED.)
V	SIX TIMES MAXIMUM DIMENSION OF DAMAGE (MAXIMUM ALLOWED.)
W	0.25 TIMES Z. (MAXIMUM ALLOWED.)
X	WIDTH OF SKIN ATTACHMENT FLANGE.
Y	30 TIMES WEB THICKNESS.
Z	EFFECTIVE WEB HEIGHT EQUALS L-2(X) FOR TYPE I AND L-2(Y) FOR TYPE II.

NOTE

1. TO BE CLASSED AS NEGLIGIBLE, ALL DAMAGE INDICATED MUST MEET THE LISTED DIMENSION AND LOCATION REQUIREMENTS AFTER CORRECTIVE TREATMENT.
2. DETERMINE THE CRACK EXTREMITIES BY PENETRANT INSPECTION (REFER TO SECTION 51.3) BEFORE STOP-DRILLING.
3. ALL CRACKS MUST BE STOP DRILLED. DRILL DIAMETER TO BE APPROXIMATELY 3 TIMES THICKNESS OF MATERIAL BUT NOT LESS THAN 0.1285 (NO. 30) NOR GREATER THAN 0.25.
4. ALL ADDITIONAL RIVETS TO BE SAME TYPE AND DIAMETER AS ADJACENT EXISTING RIVETS.
5. WEB PERFORATIONS AND CRACKS ARE LIMITED TO "EFFECTIVE WEB" LOCATIONS ONLY.
6. DAMAGE SUCH AS B OR G MUST BE AT LEAST AS PER DIMENSION "V" FROM SIMILAR DAMAGE.
7. DAMAGE SUCH AS A, C, OR E MUST BE AT LEAST 2.0 INCHES FROM ANY OTHER DAMAGE.
8. NOT APPLICABLE TO CENTER OR OUTER WING FRONT AND REAR BEAM ASSEMBLIES.

Figure 59-52. Negligible Damage Limits for Beams, Ribs, and Bulkheads (Sheet 2 of 2)

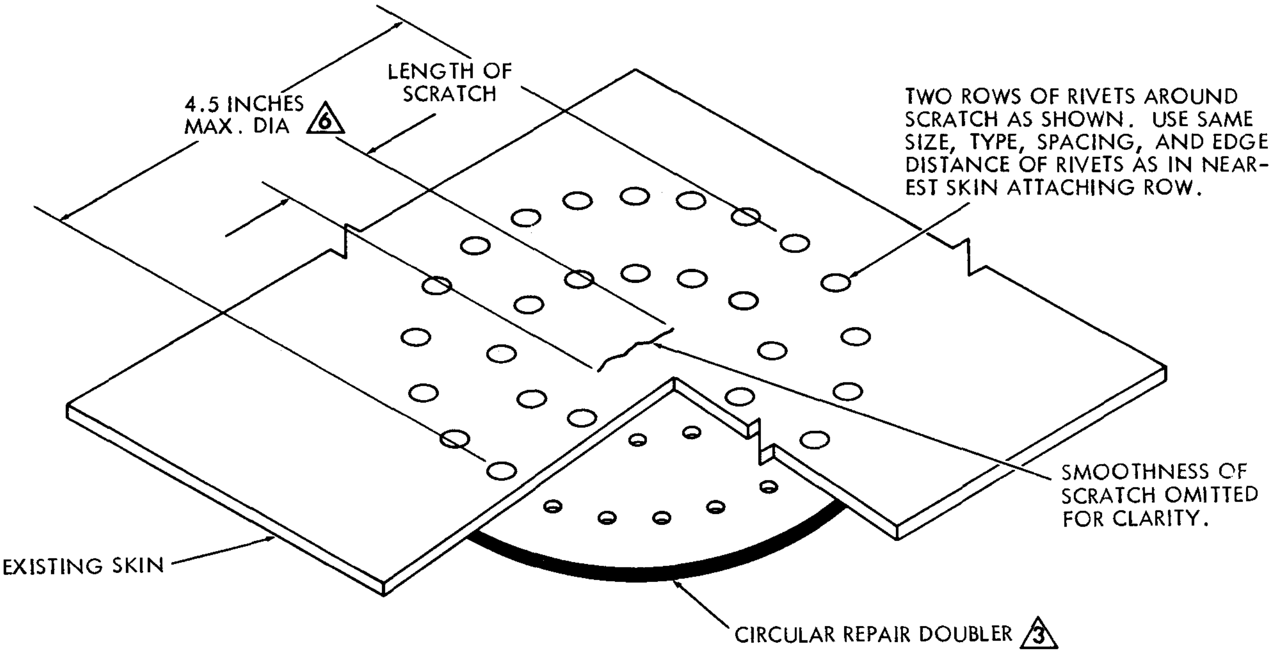

TWO ROWS OF RIVETS EACH SIDE OF SCRATCH AS SHOWN. USE SAME SIZE, TYPE, SPACING, AND EDGE DISTANCE OF RIVETS AS IN NEAREST SKIN ATTACHING ROW.
SMOOTHING OF SCRATCH OMITTED FOR CLARITY.
EXISTING SKIN
EXISTING STRUCTURAL MEMBER
3 REPAIR DOUBLER
EXISTING SKIN
CENTERLINE OF EXISTING FASTENER

SECTION THROUGH REPAIRED AREA

NOTE

1. DETERMINE IF SCRATCH IS CLASS A ACCORDING TO INSTRUCTIONS IN THIS SECTION.
2. WORK OUT THE SCRATCH ACCORDING TO INSTRUCTIONS IN THIS SECTION.
3. MAKE DOUBLERS OF 2024-T3 CLAD MATERIAL ONE THICKNESS HEAVIER THAN EXISTING SKIN, AND BUTT AGAINST STRUCTURAL MEMBER.
4. MINIMUM DISTANCE BETWEEN EDGE OF SMOOTHED AREA AND NEAREST ROW OF REPAIR RIVETS TO BE TWICE THE RIVET DIAMETER.
5. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4

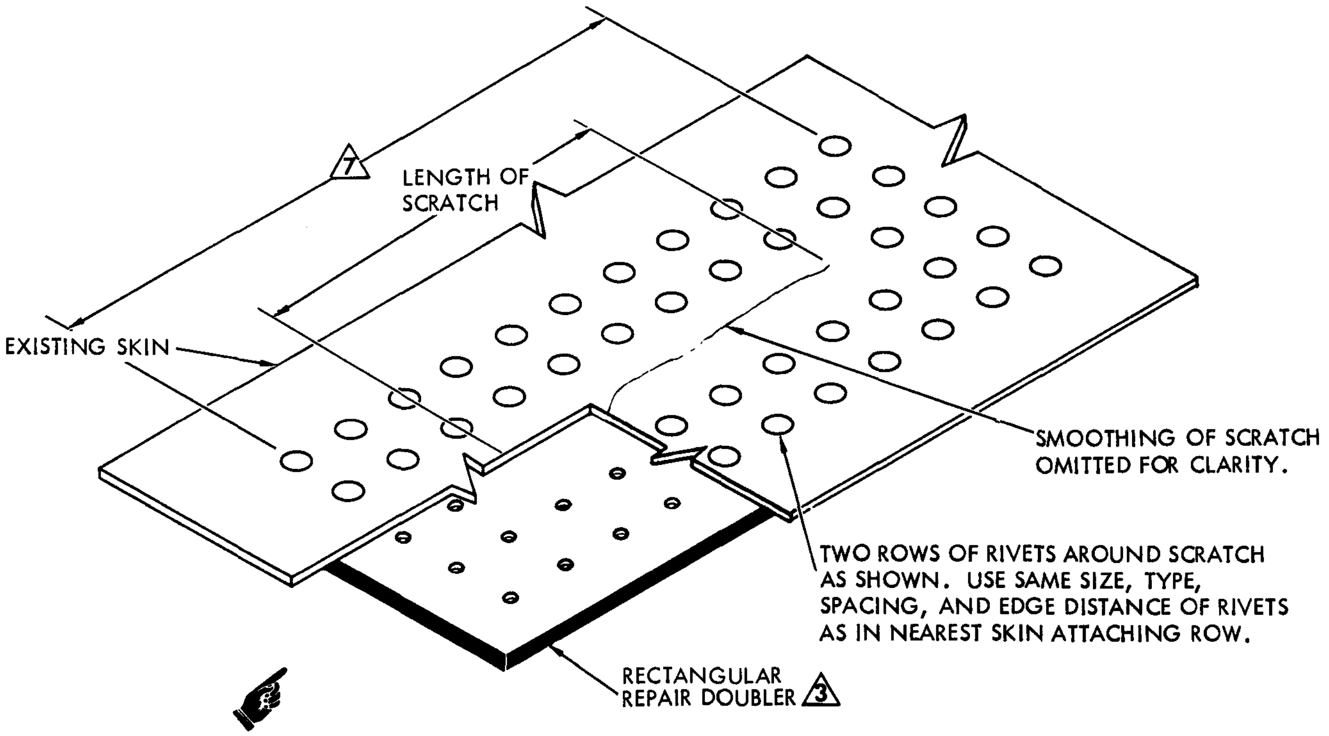
Figure 59-53. Typical Fuselage Skin Scratch Repair at Frames and Cutouts


4.5 INCHES MAX. DIA 6
LENGTH OF SCRATCH
TWO ROWS OF RIVETS AROUND SCRATCH AS SHOWN. USE SAME SIZE, TYPE, SPACING, AND EDGE DISTANCE OF RIVETS AS IN NEAREST SKIN ATTACHING ROW.
SMOOTHNESS OF SCRATCH OMITTED FOR CLARITY.
EXISTING SKIN
CIRCULAR REPAIR DOUBLER 3

NOTE

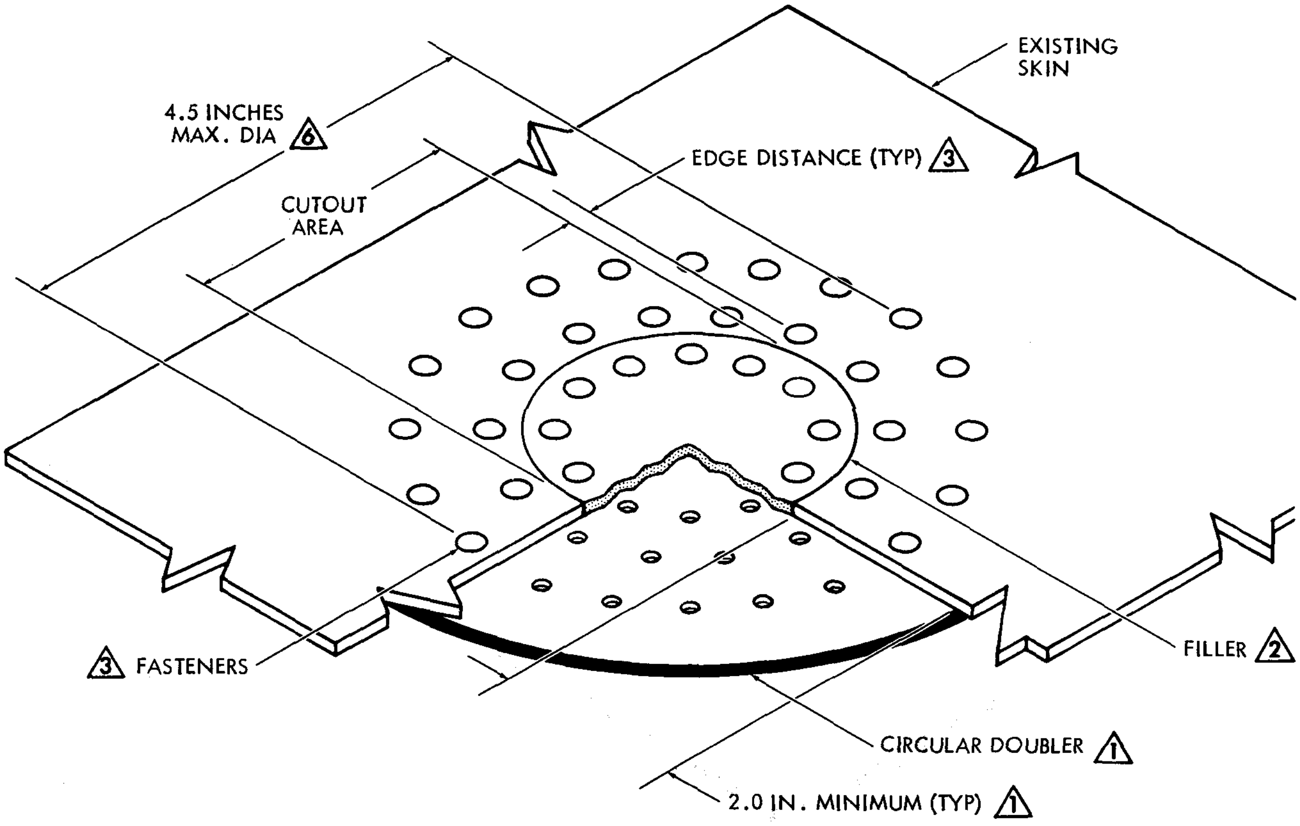
1. DETERMINE IF THE SCRATCH IS CLASS A ACCORDING TO INSTRUCTIONS IN SECTION 59.4.
2. WORK OUT SCRATCH ACCORDING TO INSTRUCTIONS IN THIS SECTION.
3. MAKE DOUBLER OF 2024-T3 CLAD MATERIAL ONE THICKNESS HEAVIER THAN EXISTING SKIN.
4. MINIMUM DISTANCE BETWEEN EDGE OF SMOOTHED AREA AND NEAREST ROW OF REPAIR RIVETS TO BE TWICE THE RIVET DIAMETER.
5. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4.
6. USE RECTANGULAR DOUBLER WHEN THIS DIMENSION EXCEEDS 4.5 INCHES.
7. USE CIRCULAR DOUBLER WHEN THIS DIMENSION IS 4.5 INCHES OR LESS.

Figure 59-54. Fuselage Pressure Skin Scratch Repair in Unobstructed Areas (Sheet 1 of 2)


7
LENGTH OF SCRATCH
EXISTING SKIN
SMOOTHING OF SCRATCH OMITTED FOR CLARITY.
TWO ROWS OF RIVETS AROUND SCRATCH AS SHOWN. USE SAME SIZE, TYPE, SPACING, AND EDGE DISTANCE OF RIVETS AS IN NEAREST SKIN ATTACHING ROW.
RECTANGULAR REPAIR DOUBLER 3

JS-1-2-X0/8-5954-2

Figure 59-54. Fuselage Pressure Skin Scratch Repair in Unobstructed Areas (Sheet 2 of 2)



NOTE

△1 MAKE DOUBLER OF SAME MATERIAL AS SKIN, BUT ONE THICKNESS HEAVIER. THE EDGE OF THE DOUBLER MUST NOT BE LESS THAN 2.0 INCHES FROM ANY PART OF THE CUTOUT AREA.

△2 MAKE FILLER OF SAME MATERIAL AND THICKNESS AS SKIN.

△3 FASTENERS USED TO ATTACH DOUBLER TO SKIN, AND FILLER TO DOUBLER, ARE TO BE THE SAME IN TYPE, SIZE, SPACING, DISTANCE BETWEEN ROWS, AND EDGE DISTANCE AS THE NEAREST ADJACENT SKIN SPLICE OF THE DAMAGED PANEL. USE AT LEAST TWO ROWS OF FASTENERS TO ATTACH THE DOUBLER TO THE SKIN, AND ONE ROW OF FASTENERS TO ATTACH THE FILLER TO THE DOUBLER.

4. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4 WHEN THE REPAIR IS APPLIED TO FUSELAGE PRESSURE SKIN.

5. FUEL-SEAL THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 28.2 WHEN THE REPAIR IS APPLIED TO EXTERNAL FUEL TANK SKIN. DO NOT USE THIS REPAIR ON INTEGRAL WING TANK PANELS.

△6 USE RECTANGULAR DOUBLER WHEN THIS DIMENSION EXCEEDS 4.5 INCHES.

△7 USE CIRCULAR DOUBLER WHEN THIS DIMENSION IS 4.5 INCHES OR LESS.

Figure 59-55. Typical Flush Skin Repair for Unobstructed Areas (Sheet 1 of 2)

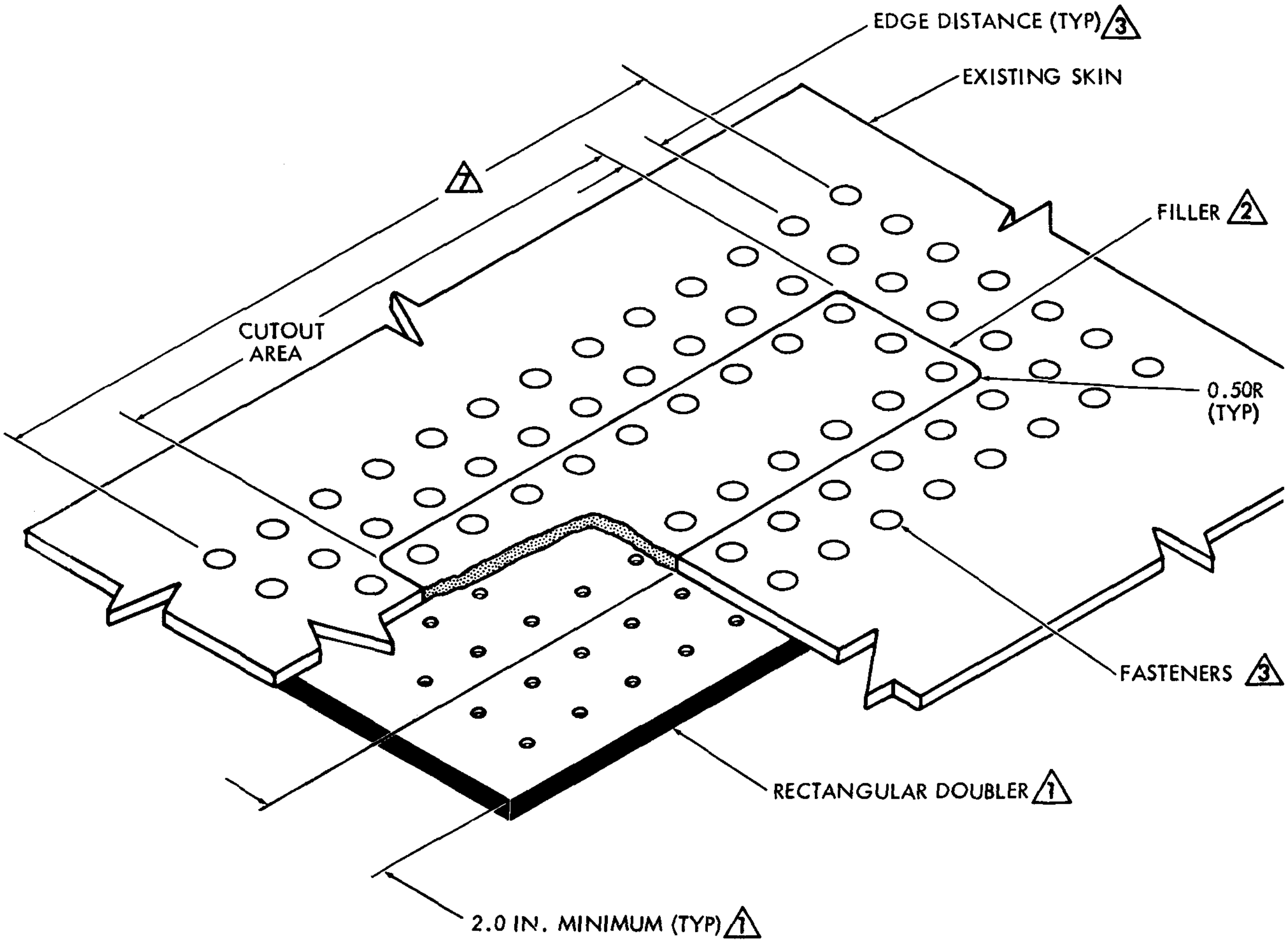
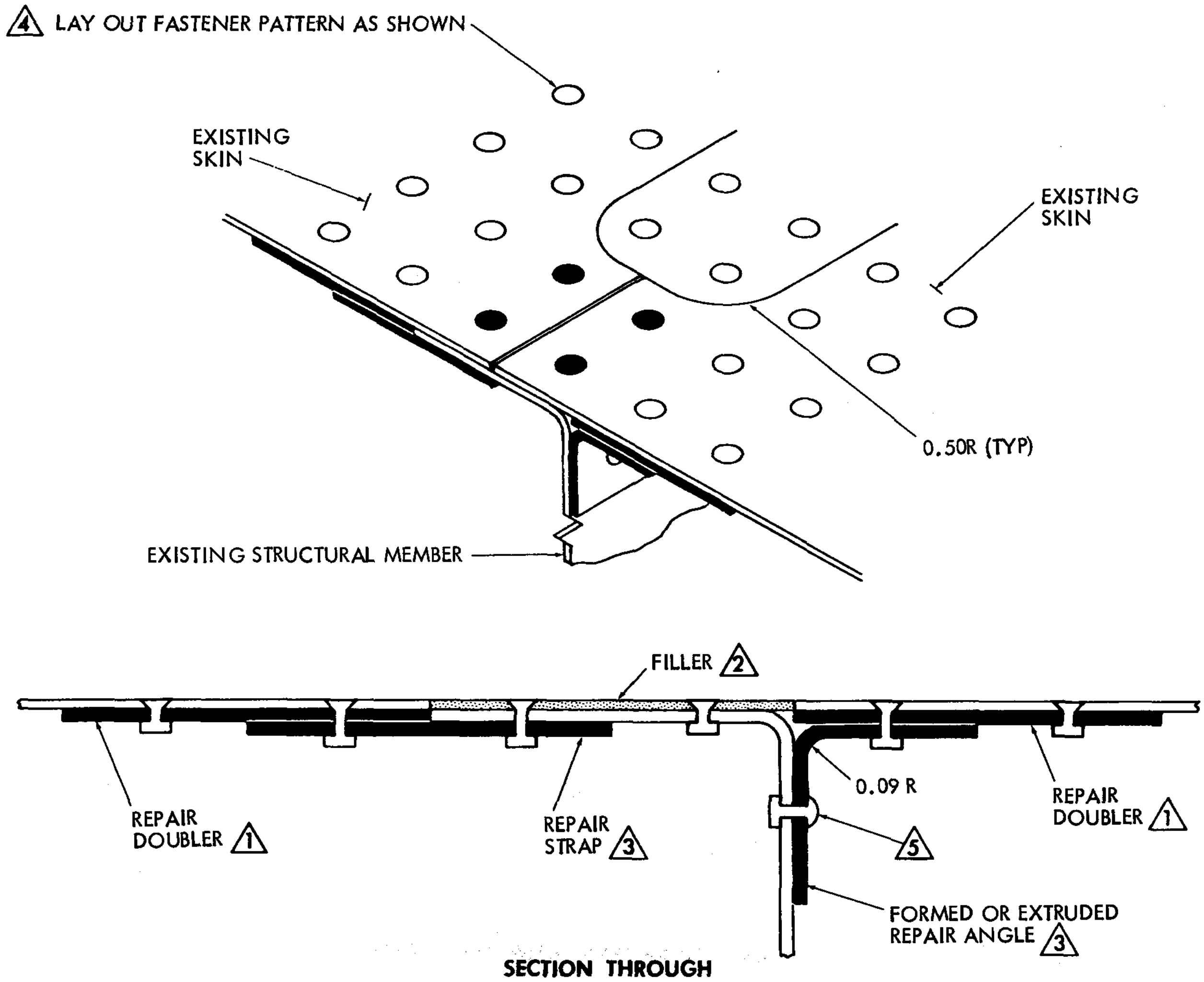


Figure 59-55. Typical Flush Skin Repair for Unobstructed Areas (Sheet 2 of 2)



SECTION THROUGH REPAIRED AREA

NOTE

1. MAKE DOUBLER OF SAME MATERIAL AS SKIN, BUT ONE THICKNESS HEAVIER AND BUTT AGAINST STRUCTURAL MEMBER. THE REMAINING EDGES OF THE DOUBLER MUST NOT BE LESS THAN 2.0 INCHES FROM ANY PART OF THE CUTOUT AREA.
2. MAKE FILLER OF SAME MATERIAL AND THICKNESS AS SKIN.
3. MAKE STRAP AND ANGLE OF SAME MATERIAL AND THICKNESS AS DOUBLER.
4. USE SAME MATERIAL, SIZE, TYPE, SPACING, DISTANCE BETWEEN ROWS, AND EDGE DISTANCE OF FASTENERS AS THE NEAREST ADJACENT SKIN SPLICE OF THE DAMAGED PANEL. USE A MINIMUM OF TWO ROWS OUTSIDE OF FILLER.
5. USE ONE ROW OF PROTRUDING HEAD FASTENERS OF SAME MATERIAL AS ANGLE AT 4D MINIMUM SPACING AND 2D MINIMUM EDGE DISTANCE. DETERMINE SIZE AS FOLLOWS:

ANGLE THICKNESS	FASTENER SIZE (D)
0.025	1/8
0.032	5/32
0.040	5/32
0.050 AND UP	3/16

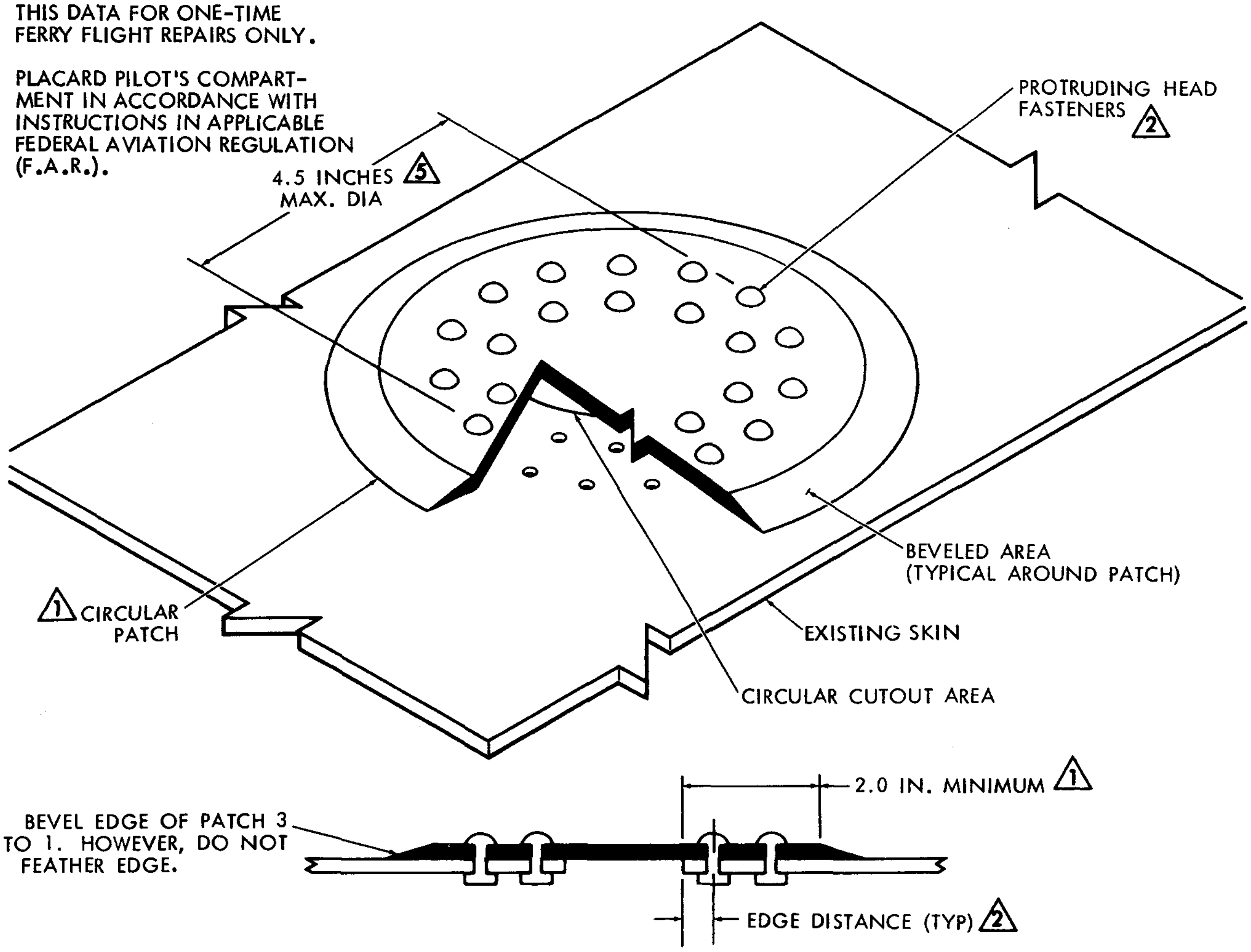
6. PRESSURE SEAL PER SECTION 51.2 AND LEAK-TEST REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4 WHEN REPAIR IS APPLIED TO FUSELAGE PRESSURE SKIN.
7. FUEL SEAL REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 28.2 WHEN REPAIR IS APPLIED TO EXTERNAL FUEL TANK SKIN. DO NOT USE THIS REPAIR ON INTEGRAL WING TANK SKIN PANELS.

Figure 59-56. Typical Flush Skin Repair at Panel Edges

WARNING

THIS DATA FOR ONE-TIME FERRY FLIGHT REPAIRS ONLY.

PLACARD PILOT'S COMPARTMENT IN ACCORDANCE WITH INSTRUCTIONS IN APPLICABLE FEDERAL AVIATION REGULATION (F.A.R.).


4.5 INCHES △5
MAX. DIA
PROTRUDING HEAD FASTENERS △2
△1 CIRCULAR PATCH
BEVELED AREA (TYPICAL AROUND PATCH)
EXISTING SKIN
CIRCULAR CUTOUT AREA
2.0 IN. MINIMUM △1
BEVEL EDGE OF PATCH 3 TO 1. HOWEVER, DO NOT FEATHER EDGE.
EDGE DISTANCE (TYP) △2

SECTION THROUGH REPAIRED AREA

NOTE

△1 MAKE PATCH OF SAME MATERIAL AND THICKNESS AS SKIN. THE TOP BEVELED EDGE OF THE PATCH MUST NOT BE LESS THAN 2.0 INCHES FROM ANY PART OF THE CUTOUT AREA.

△2 PROTRUDING HEAD FASTENERS USED TO ATTACH PATCH TO SKIN ARE TO BE THE SAME IN MATERIAL, SPACING, DISTANCE BETWEEN ROWS, AND EDGE DISTANCE AS THE NEAREST ADJACENT SKIN SPLICE OF THE DAMAGED PANEL. USE ONE SIZE SMALLER FASTENER THAN THOSE IN THE NEAREST ADJACENT SKIN SPLICE, AND A MINIMUM OF TWO ROWS OF FASTNERS. SEE SECTION 59.1 FOR FILLING OF MACHINE COUNTER SUNK HOLES.

3. BEFORE APPLYING THIS REPAIR OVER CRACKED SKIN, DETERMINE THE CRACK EXTREMITIES BY PENETRANT INSPECTION (SEE SECTION 51.3), AND STOP-DRILL CRACK ESTREMITIES WITH A NO. 30 (0.1285) DRILL.

4. DEFUEL THE EXTERNAL FUEL TANK BEFORE APPLYING THIS REPAIR TO THE TANK SKIN. DO NOT USE THIS REPAIR ON INTEGRAL WING TANK PANELS.

△5 USE RECTANGULAR PATCH WHEN THIS DIMENSION EXCEEDS 4.5 INCHES.

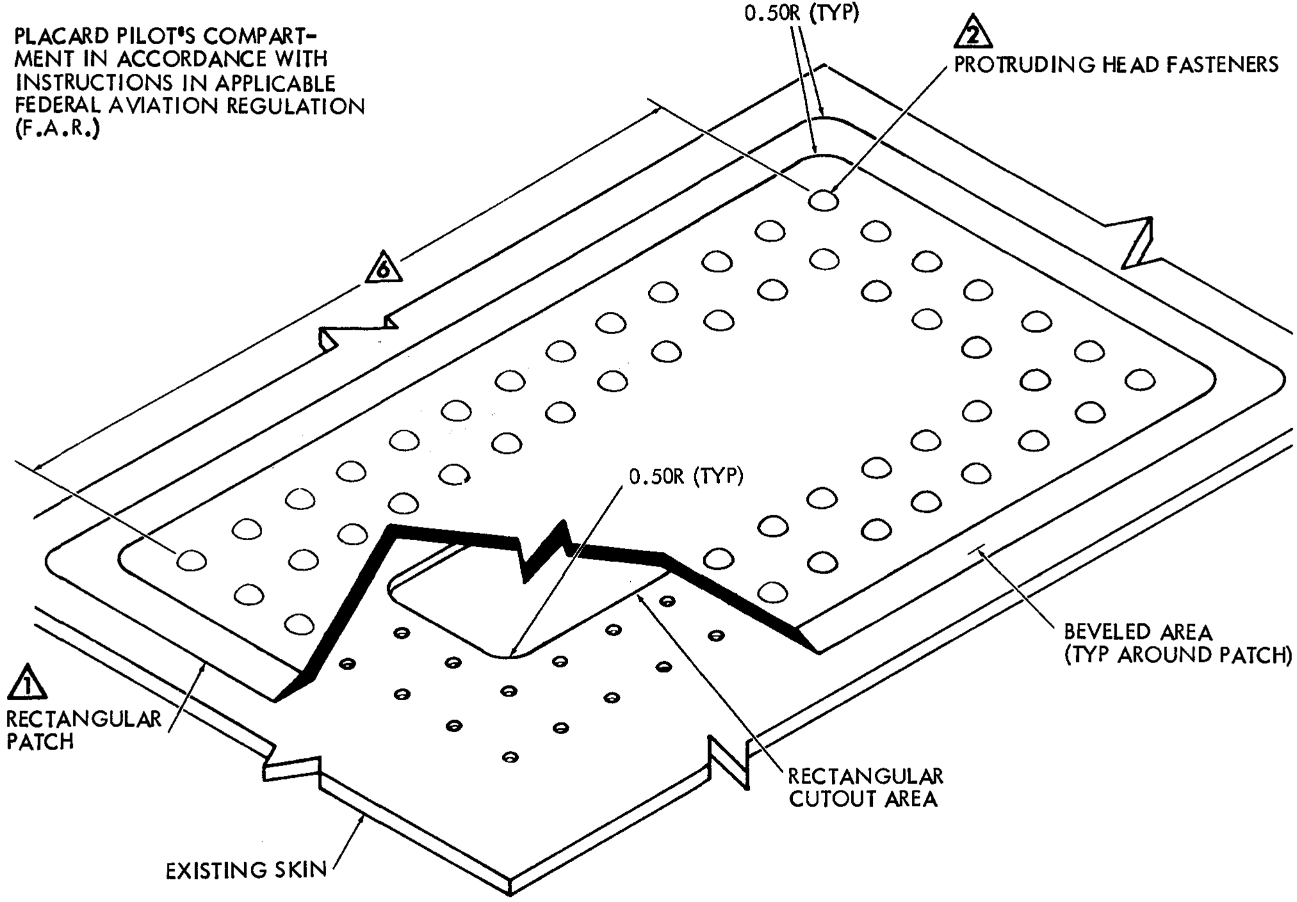
△6 USE CIRCULAR PATCH WHEN THIS DIMENSION IS 4.5 INCHES OR LESS.

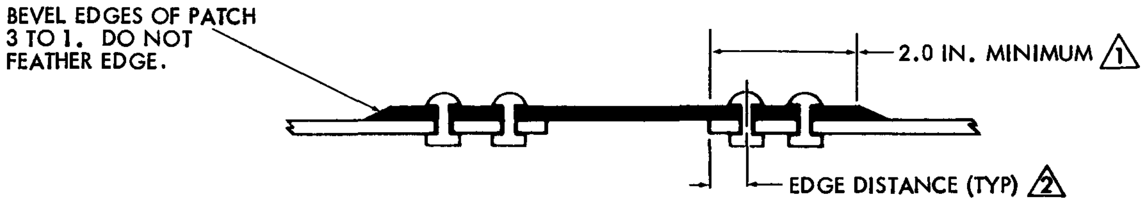
Figure 59-57. Typical One-Time Ferry Flight Repair (Sheet 1 of 2)

WARNING

THIS DATA FOR ONE-TIME FERRY FLIGHT REPAIRS ONLY.

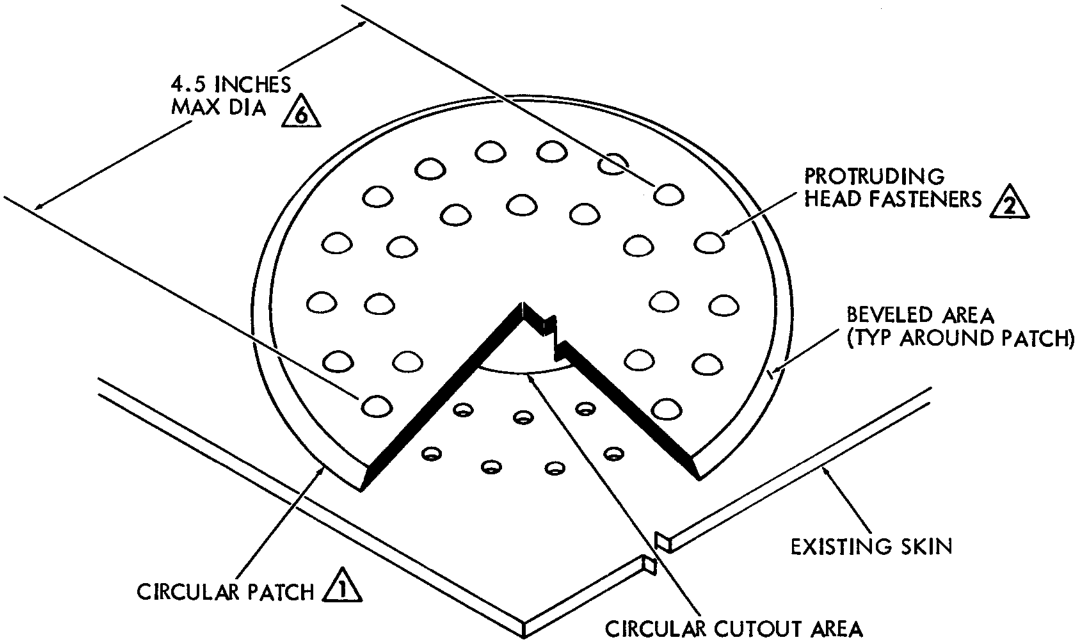
PLACARD PILOT'S COMPARTMENT IN ACCORDANCE WITH INSTRUCTIONS IN APPLICABLE FEDERAL AVIATION REGULATION (F.A.R.)

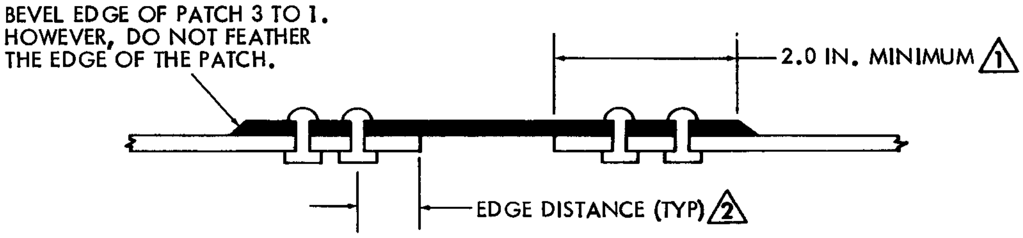




SECTION THROUGH REPAIRED AREA

Figure 59-57. Typical One-Time Ferry Flight Repair (Sheet 2 of 2)





SECTION THROUGH REPAIRED AREA

NOTE

1. MAKE PATCH OF SAME MATERIAL AS SKIN, BUT ONE THICKNESS HEAVIER. THE TOP BEVELED EDGE OF THE PATCH MUST NOT BE LESS THAN 2.0 INCHES FROM ANY PART OF THE CUTOUT AREA.
2. PROTRUDING-HEAD FASTENERS USED TO ATTACH PATCH TO SKIN ARE TO BE THE SAME IN MATERIAL, SIZE, SPACING, DISTANCE BETWEEN ROWS, AND EDGE DISTANCE AS THE NEAREST ADJACENT SKIN SPLICE OF THE DAMAGED PANEL. USE AT LEAST TWO ROWS OF FASTENERS AROUND CUTOUT AREA. SEE SECTION 59.1 FOR FILLING OF MACHINE COUNTER SUNK HOLES.
3. BEFORE APPLYING THIS REPAIR OVER CRACKED SKIN, DETERMINE THE CRACK EXTREMITIES BY PENETRANT INSPECTION (SEE SECTION 51.3), AND STOP-DRILL CRACK EXTREMITIES WITH A NO. 30 (0.1285) DRILL.
4. PRESSURE-SEAL PER SECTION 51.2 AND LEAK-TEST THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 51.4 WHEN THE REPAIR IS APPLIED TO FUSELAGE PRESSURE SKIN.
5. FUEL-SEAL THE REPAIR ACCORDING TO INSTRUCTIONS IN SECTION 28.2 WHEN THE REPAIR IS APPLIED TO EXTERNAL FUEL TANK SKIN. DO NOT USE THIS REPAIR ON INTEGRAL WING TANK PANELS.
6. USE RECTANGULAR PATCH WHEN THIS DIMENSION EXCEEDS 4.5 INCHES.
7. USE CIRCULAR PATCH WHEN THIS DIMENSION IS 4.5 INCHES OR LESS.

Figure 59-58. Typical Temporary Skin Repair (Sheet 1 of 2)

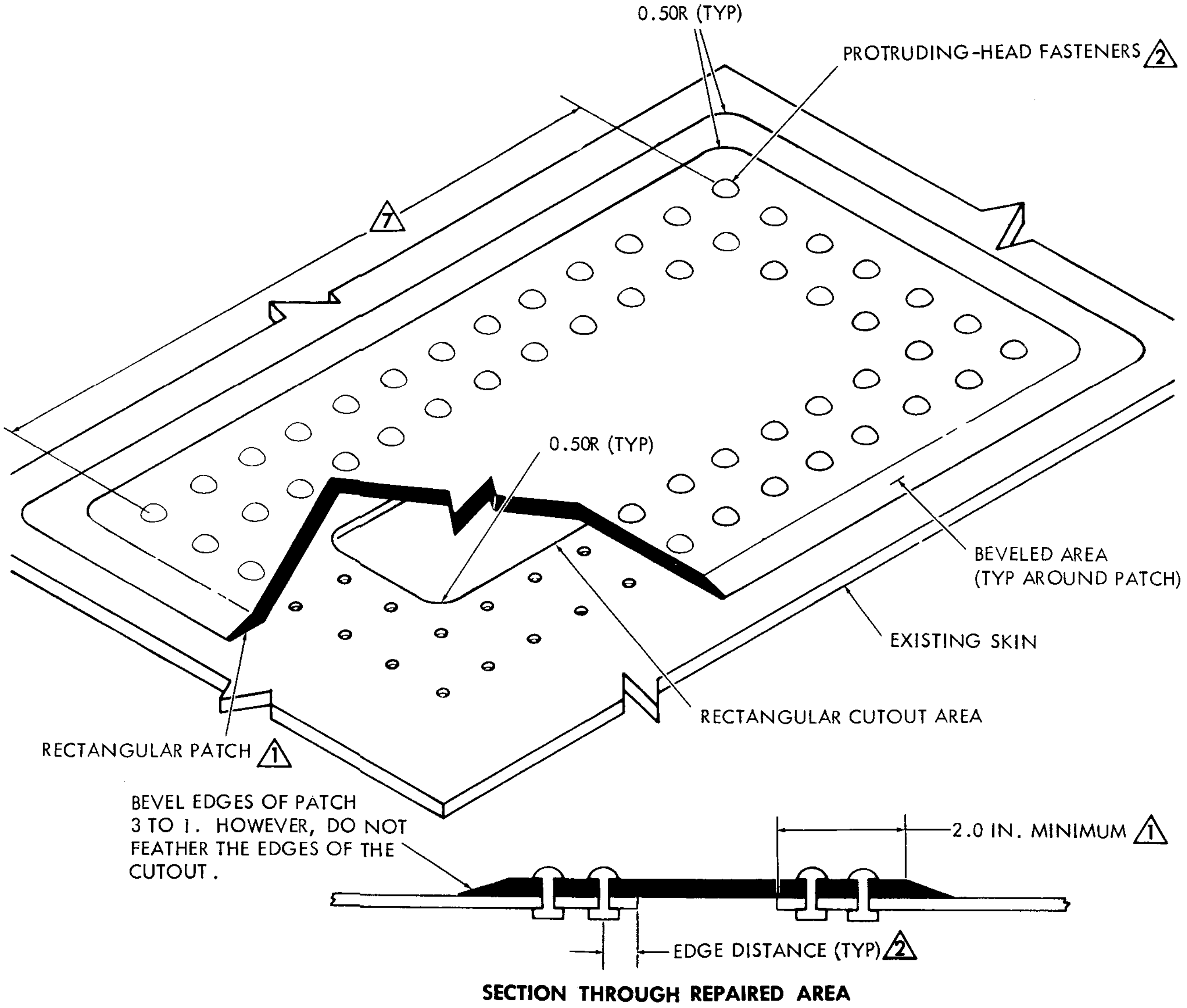
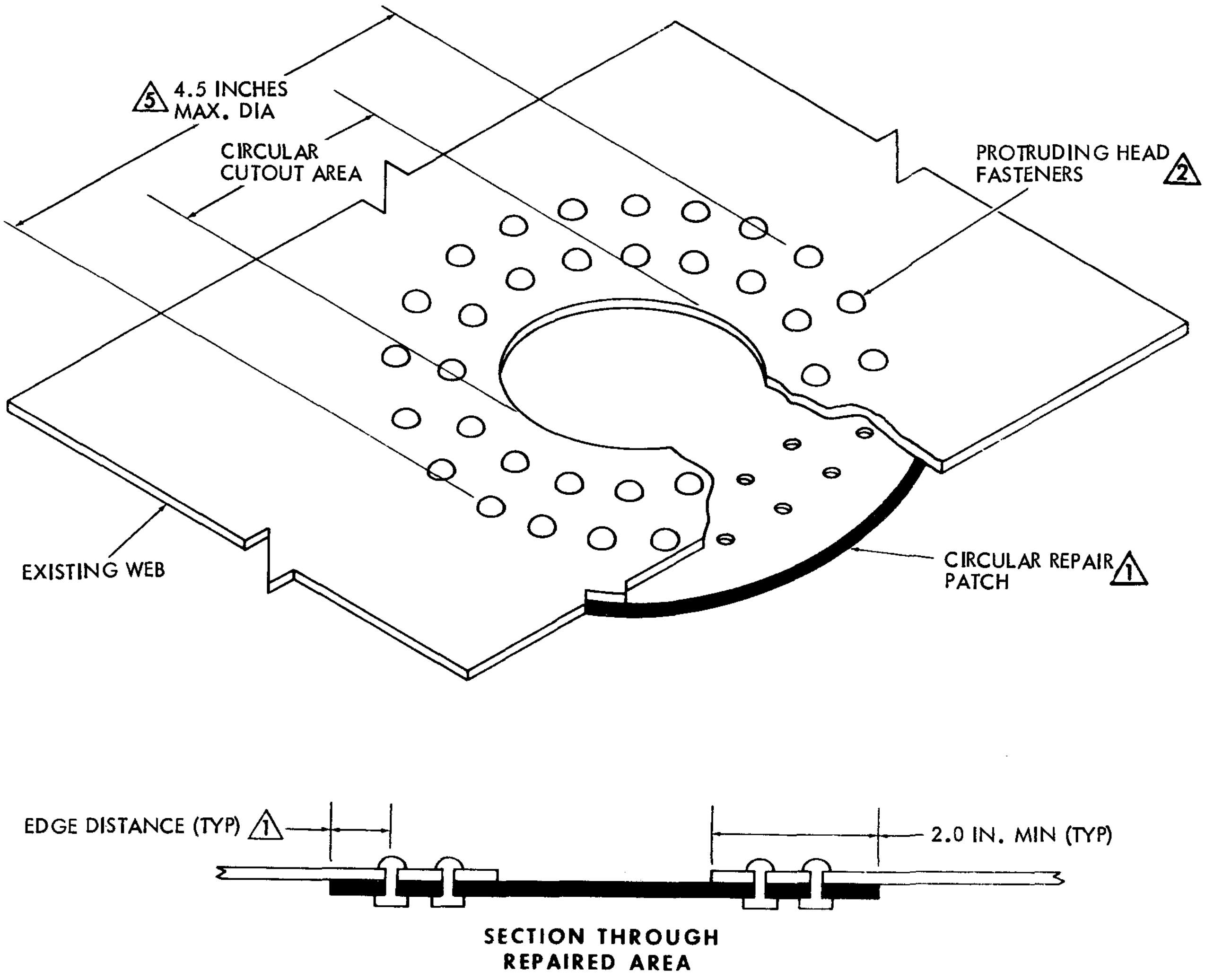


Figure 59-58. Typical Temporary Skin Repair (Sheet 2 of 2)



NOTE

1. MAKE PATCH OF SAME MATERIAL AS EXISTING WEB, BUT ONE THICKNESS HEAVIER. THE EDGE OF THE PATCH MUST NOT BE LESS THAN 2.0 INCHES FROM ANY PART OF THE CUTOUT AREA.
2. USE AT LEAST TWO ROWS OF PROTRUDING-HEAD FASTENERS AROUND CUTOUT AREA. THESE FASTENERS ARE TO BE OF SAME MATERIAL, SIZE, SPACING, DISTANCE BETWEEN ROWS AND EDGE DISTANCE AS NEAREST ROW OF WEB ATTACHMENT FASTENERS.
3. PRESSURE SEAL PER SECTION 51.2 AND LEAK-TEST REPAIRS ACCORDING TO INSTRUCTIONS IN SECTION 51.4 WHEN REPAIR IS APPLIED TO FUSELAGE PRESSURIZED SECTION.
4. FUEL-SEAL REPAIRS IN FUEL TANK AREAS IN ACCORDANCE WITH INSTRUCTIONS IN SECTION 28.2.
5. USE RECTANGULAR PATCH WHEN THIS DIMENSION EXCEEDS 4.5 INCHES.
6. USE CIRCULAR PATCH WHEN THIS DIMENSION IS 4.5 INCHES OR LESS.

Figure 59-59. Typical Web Repair (Sheet 1 of 2)

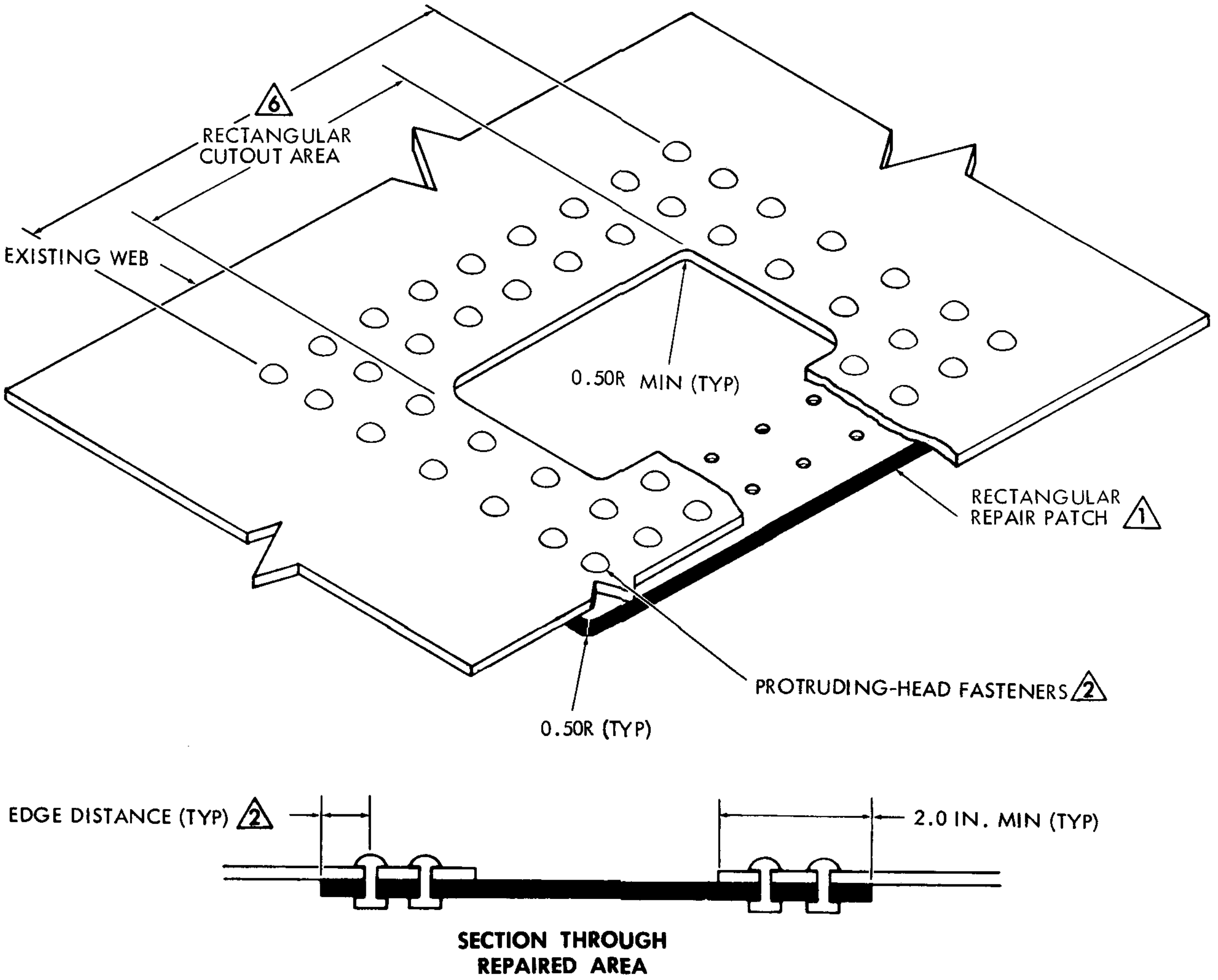
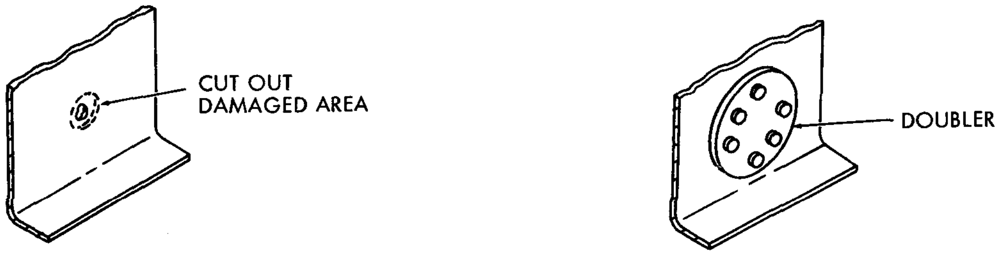

6 RECTANGULAR CUTOUT AREA
EXISTING WEB
0.50R MIN (TYP)
RECTANGULAR REPAIR PATCH 1
PROTRUDING-HEAD FASTENERS 2
0.50R (TYP)
EDGE DISTANCE (TYP) 2
2.0 IN. MIN (TYP)
SECTION THROUGH REPAIRED AREA

Figure 59-59. Typical Web Repair (Sheet 2 of 2)



APPLICATION

FOR REPAIR OF DAMAGED WEB IN SECONDARY STRUCTURE.

LIMITATIONS

1. DO NOT USE ON PRESSURIZED SKIN.

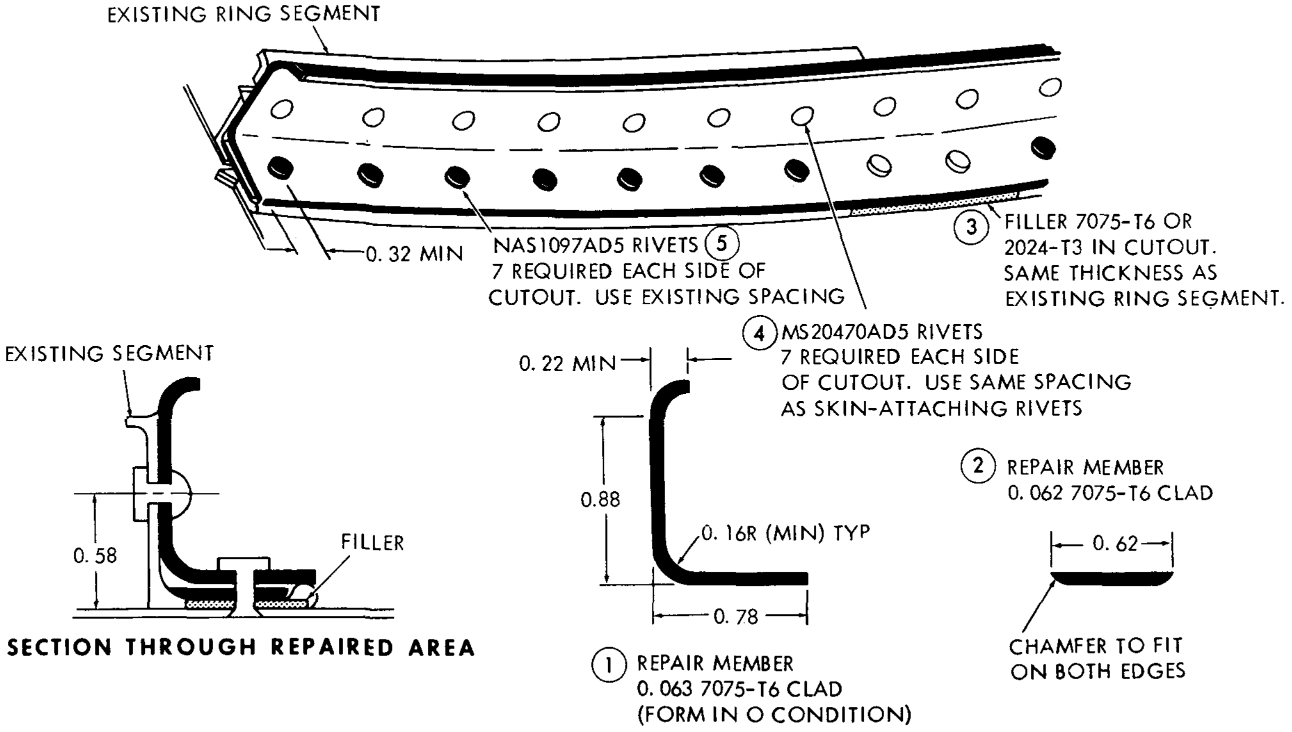
PARTS REQUIRED

1. ONE CIRCULAR DOUBLER OF SAME MATERIAL AND THICKNESS AS WEB - WIDTH SUFFICIENT TO MAINTAIN MINIMUM ED OF 2.

PROCEDURE

1. CUT ROUND HOLE TO REMOVE DAMAGED AREA AS SHOWN.
2. ATTACH DOUBLER TO EITHER SIDE OF WEB, USING SAME RIVET TYPE AND SPACING AS USED TO ATTACH PART BEING REPAIRED.
3. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-60. Damaged Web-Secondary Structure-Doubler Repair

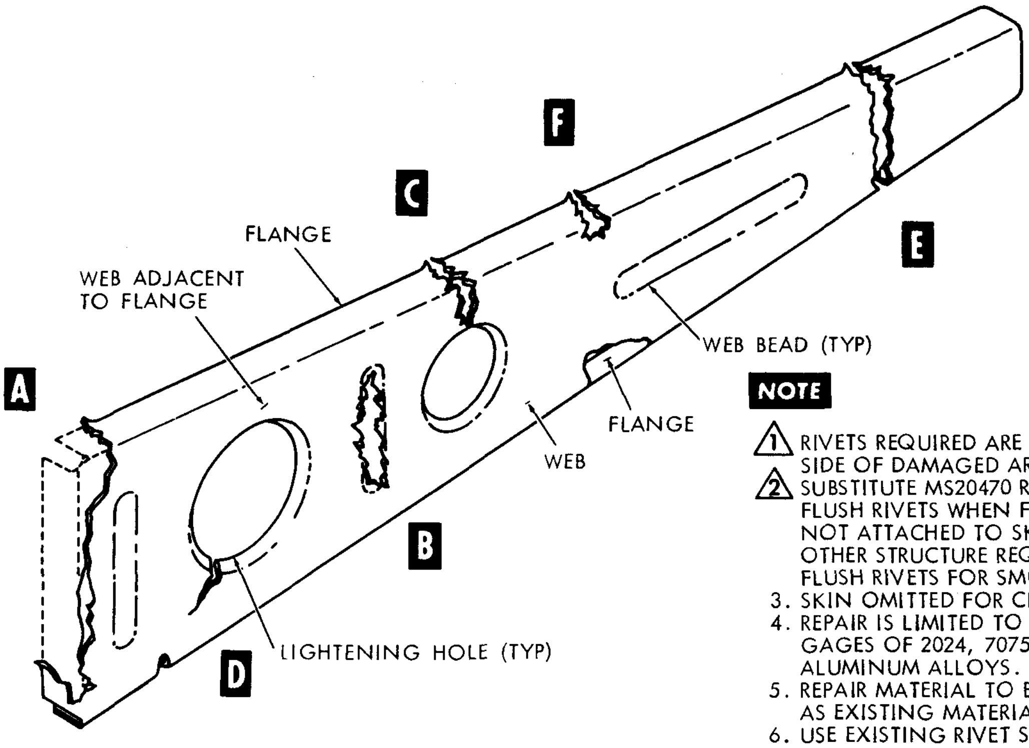


APPLICATION
FUSELAGE L RING SEGMENT REPAIR

PARTS REQUIRED
① ANGLE, LENGTH TO PICK UP SEVEN RIVETS EACH SIDE OF CUT OUT
② STRAP, LENGTH TO PICK UP SEVEN RIVETS EACH SIDE OF CUT OUT
③ FILLER, THICKNESS EQUAL TO SEGMENT LENGTH AND WIDTH AS REQUIRED
④ MS 20470 AD5 RIVETS
⑤ NAS 1097 AD5 OR NAS 1097 AD4 RIVETS, USE EXISTING SPACING AND RIVET SIZE

PROCEDURES:
1. REMOVE RIVETS & DAMAGED AREA
2. INSTALL ANGLE, STRAP, AND FILLERS
3. FINISH SAME AS ADJACENT STRUCTURE
4. PRESSURE-SEAL PER SECTION 51.2 WHEN REPAIR IS APPLIED TO FUSELAGE PRESSURIZED SECTION.
5. LEAK-CHECK PER INSTRUCTIONS IN SECTION 51.4 WHEN REPAIR IS APPLIED TO FUSELAGE PRESSURIZED SECTION.

Figure 59-61. Typical Fuselage Ring "L" Segment Repair



NOTE

△1 RIVETS REQUIRED ARE ON EACH SIDE OF DAMAGED AREA.
△2 SUBSTITUTE MS20470 RIVETS FOR FLUSH RIVETS WHEN FLANGE IS NOT ATTACHED TO SKIN OR OTHER STRUCTURE REQUIRING FLUSH RIVETS FOR SMOOTHNESS.
3. SKIN OMITTED FOR CLARITY.
4. REPAIR IS LIMITED TO EXISTING GAGES OF 2024, 7075, AND 7079 ALUMINUM ALLOYS.
5. REPAIR MATERIAL TO BE SAME AS EXISTING MATERIAL.
6. USE EXISTING RIVET SPACING IN FLANGE. USE SAME RIVET SPACING IN WEB ADJACENT TO FLANGE. USE 4D TO 5D (RIVET DIAMETERS) CENTER-TO-CENTER RIVET SPACING WHERE THE FLANGE IS NOT ATTACHED TO SKIN OR OTHER STRUCTURE.
7. FOR RIVET EDGE MARGIN, USE 2 TIMES RIVET DIAMETER (2D) OF RIVET USED.

EXISTING GAGE	REPAIR MATERIAL GAGE	RIVET REQUIREMENTS △1				
		FLANGE		WEB ADJACENT TO FLANGE		WEB
		RIVETS △2	NUMBER	RIVETS	NUMBER	RIVETS
0.020	0.025	NAS1097AD4	4	MS20470AD4	4	MS20470AD4
0.025	0.032	NAS1097AD4	3	MS20470AD4	3	MS20470AD4
0.032	0.040	NAS1097AD4	4	MS20470AD4	4	MS20470AD4
0.040	0.050	NAS1097AD4	4	MS20470AD4	4	MS20470AD4
		NAS1097AD5	3	MS20470AD5	3	MS20470AD5
0.050	0.063	NAS1097AD5	5	MS20470AD5	4	MS20470AD5

Figure 59-62. Typical Rib and Former Repairs (Sheet 1 of 3)

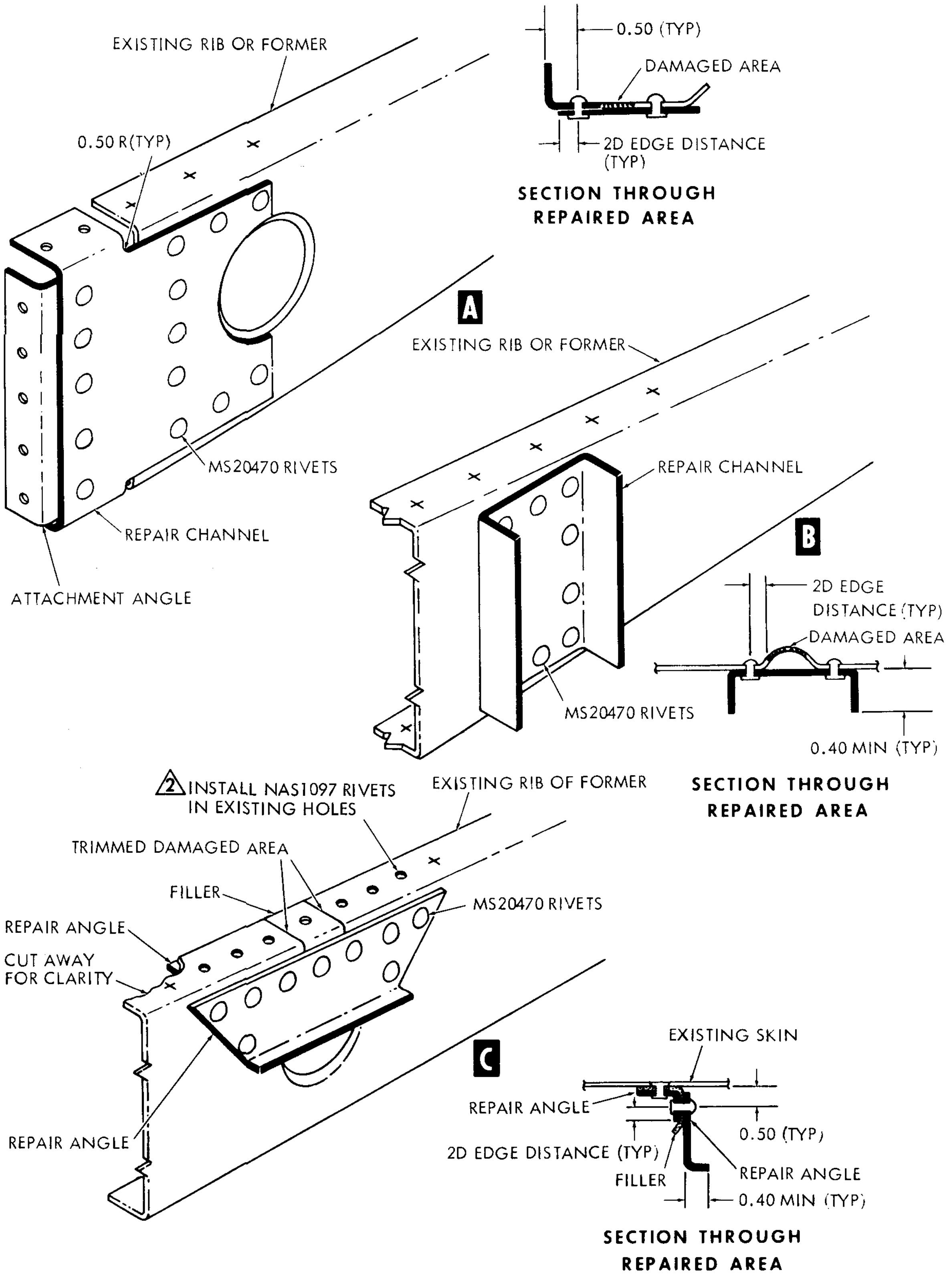


Figure 59-62. Typical Rib and Former Repairs (Sheet 2 of 3)

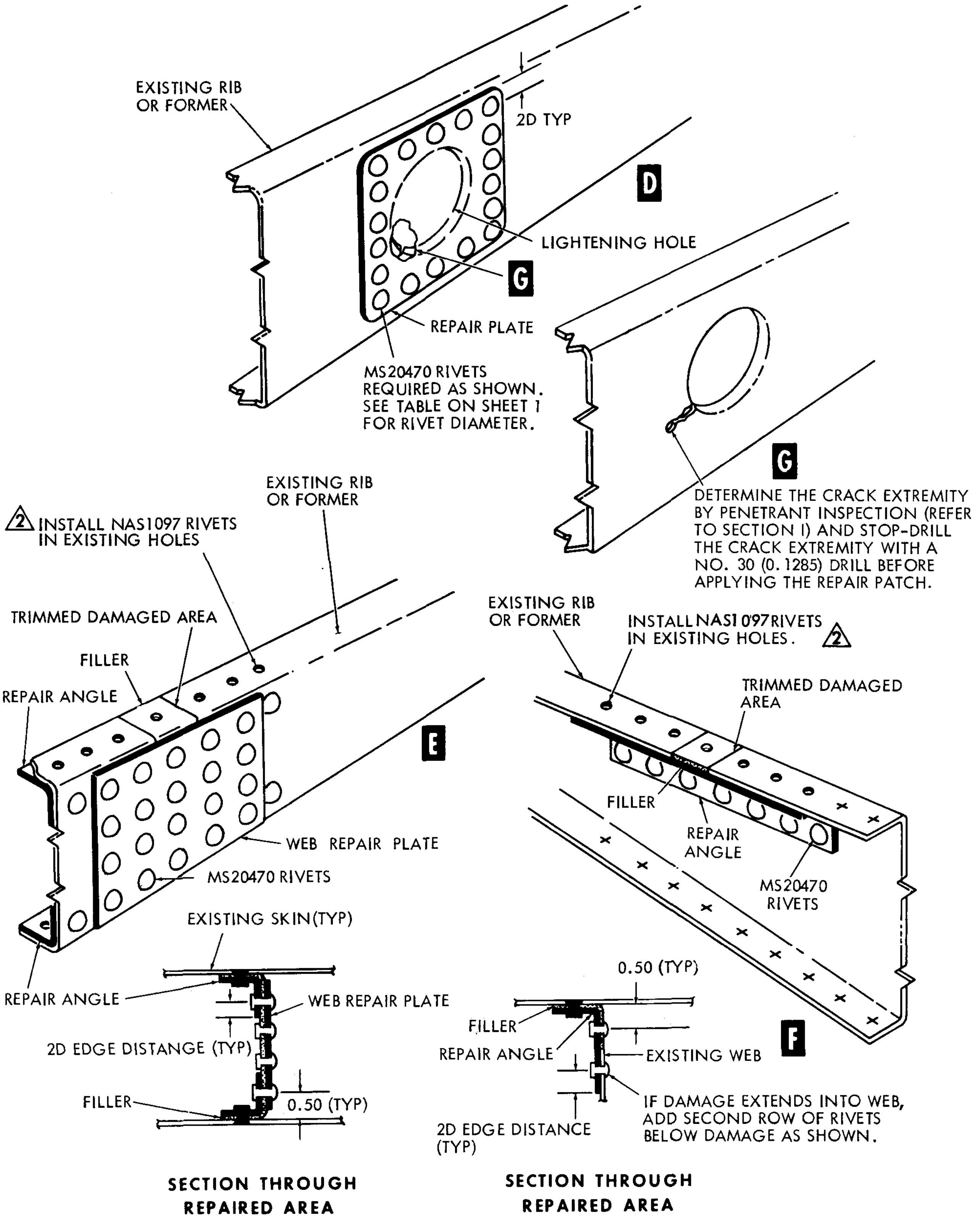
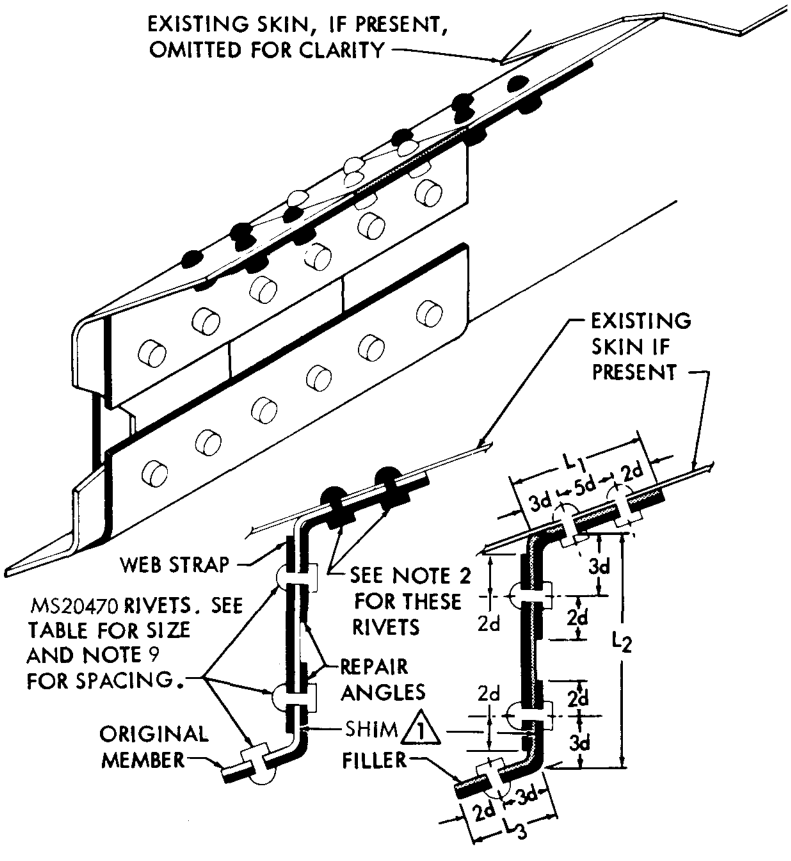


Figure 59-62. Typical Rib and Former Repairs (Sheet 3 of 3)


EXISTING SKIN, IF PRESENT, OMITTED FOR CLARITY
EXISTING SKIN IF PRESENT
WEB STRAP
SEE NOTE 2 FOR THESE RIVETS
MS20470 RIVETS. SEE TABLE FOR SIZE AND NOTE 9 FOR SPACING.
REPAIR ANGLES
ORIGINAL MEMBER
SHIM △1
FILLER
L1
3d
5d
2d
3d
2d
2d
L2
2d
2d
3d
2d
3d
L3

SECTION THROUGH REPAIRED AREA

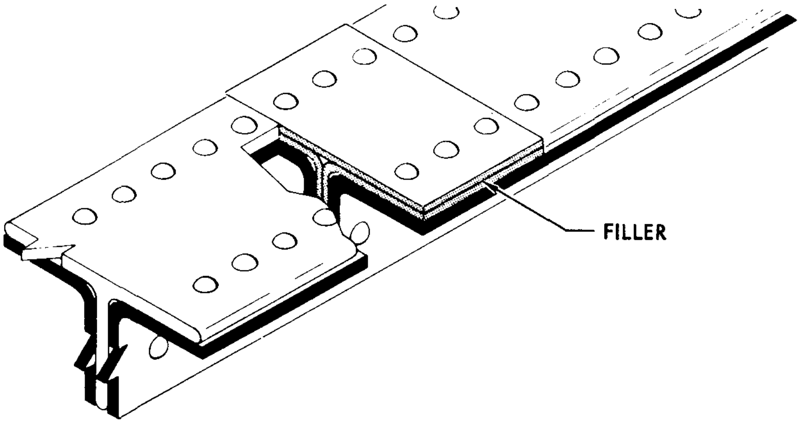
NOTE

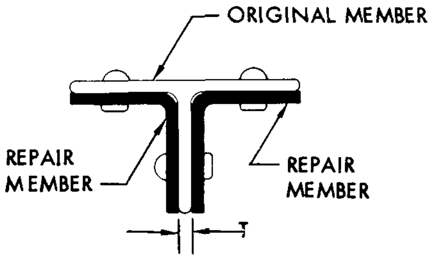
△1 WHEN ZEE IS AN EXTRUSION, USE MINIMUM THICKNESS SHIM TO PROVIDE CLEARANCE BETWEEN THE REPAIR MEMBER RADIUS AND EXTRUSION CORNER. WHEN REPAIR MEMBER IS ONE PIECE, USE STRIP SHIM AT WEB SINGLE ROW ATTACHMENT.
2. REPAIR MATERIAL TO BE SAME AS EXISTING MATERIAL.
3. USE THE SAME TYPE OF RIVETS IN SKIN FLANGE AS INSTALLED ORIGINALLY. SEE NOTE 8 FOR SPACING. IF THERE IS NO SKIN ATTACHED TO THE MEMBER, USE MS20470 RIVETS. IF FLUSH RIVETS ARE USED, USE SAME SIZE AS ORIGINALLY INSTALLED AND PREPARE HOLES IN REPAIR MEMBER AND FILLER ACCORDING TO INSTRUCTIONS IN SECTION 59.1.
4. FILLER MEMBER TO BE SAME GAGE AND MATERIAL AS ORIGINAL ZEE.
5. USE SAME NUMBER OF ROWS OF ATTACHMENTS PER FLANGE AS IN ORIGINAL ZEE.
6. FLANGE ANGLE WITH WEB MAY VARY.
7. WHEN L_2 IS LESS THAN 10 TIMES THE DIAMETER OF THE REQUIRED RIVETS, FORM A SINGLE ZEE INSTEAD OF THE TWO ANGLES AND LEAVE OFF THE WEB STRAP. USE ONLY ONE ROW OF RIVETS THROUGH THE WEB.
8. USE EXISTING SKIN FLANGE RIVET SPACING IF FLANGE IS DRILLED. IF MEMBER WAS ORIGINALLY UNDRILLED, USE 5 TIMES THE RIVET DIAMETER FOR CENTER-TO-CENTER SPACING.
9. BREAK SHARP EDGES.

ZEE GAGE	REPAIR MEMBER GAGE	SIZE OF RIVETS REQUIRED	★NUMBER OF RIVETS REQUIRED WHEN $L_1 + L_2 + L_3$ IS										
			2.00	2.50	3.00	3.50	4.00	4.50	5.00	5.50	6.00	6.50	7.00
0.020	0.025	AD4	8	10	12	14	16	18	20	22	24	26	29
0.025	0.032	AD4	10	13	15	18	20	23	25	28	30	33	36
0.032	0.040	AD4	12	15	18	22	25	28	30	34	37	40	43
0.040	0.050	AD5		13	15	18	20	23	25	28	30	33	35
0.050	0.063	AD5		15	18	21	24	27	30	33	36	39	42
0.063	0.071	AD5		19	23	27	30	34	38	42	46	50	53
0.071	0.080	D6			15	17	20	22	25	27	30	32	34
0.080	0.090	D6			17	20	23	25	27	31	33	36	39
0.090	0.100	D6			18	22	25	29	32	34	37	41	43
0.100	0.125	D8					16	17	20	22	23	25	27

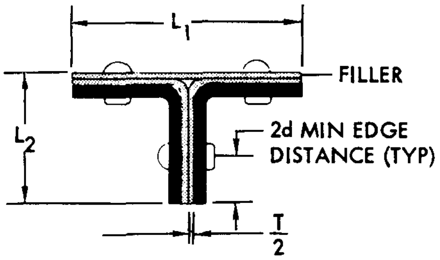
★ TABLE GIVES NUMBER OF RIVETS REQUIRED ON EACH SIDE OF THE DAMAGE. DIVIDE EQUALLY AMONG ALL ROWS. ADD TO NUMBER IN TABLE AS NECESSARY TO MAKE IT DIVISIBLE BY NUMBER OF ROWS USED. FOR IN-BETWEEN VALUES OF $L_1 + L_2 + L_3$ USE NEXT HIGHER COLUMN.

Figure 59-63. Typical Formed and Extruded "Z" Repair





REPAIR SECTION ADJACENT TO DAMAGED AREA



REPAIR SECTION THROUGH DAMAGED AREA

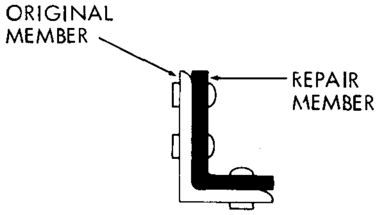
NOTE

1. TOTAL THICKNESS OF FILLER MUST EQUAL AT LEAST THE THICKNESS OF THE THINNEST FLANGE OF THE ORIGINAL TEE. SHIMS MAY BE USED AS NECESSARY FOR THE OTHER FLANGES.
2. REPAIR MEMBERS AND FILLERS TO BE SAME MATERIAL AS ORIGINAL TEE.
3. USE SAME NUMBER OF ROWS OF ATTACHMENTS PER FLANGE AS IN EXISTING ATTACHMENT PATTERN.
4. USE SAME TYPE ATTACHMENTS AS IN EXISTING STRUCTURE. IF FLUSH ATTACHMENTS ARE TO BE USED PREPARE THE HOLES ACCORDING TO INSTRUCTIONS IN SECTION 59.1.
5. ANGLE MAY BE OTHER THAN 90°.
6. REPAIR DOES NOT APPLY TO ANY TEE FOR WHICH $L_1 + L_2$ IS LESS THAN 2.50 INCHES. SEE TABLE FOR OTHER LIMITATIONS.
7. THE TABLE GIVES NUMBER OF RIVETS REQUIRED ON EACH SIDE OF THE DAMAGE. DIVIDE RIVETS EQUALLY AMONG ALL ROWS. ADD TO THE NUMBER IN THE TABLE AS NECESSARY TO MAKE IT DIVISIBLE BY THE NUMBER OF ROWS USED. FOR IN-BETWEEN VALUES OF $L_1 + L_2$ USE NEXT HIGHER COLUMN.

★ TEE GAGE	REPAIR MEMBER GAGE	SIZE OF RIVET REQUIRED	NUMBER OF RIVETS REQUIRED WHEN L_1+L_2 IS:								
			2.50	3.00	3.50	4.00	4.50	5.00	5.50	6.00	6.50
0.025	0.032	AD4	15	18	21	23	26	29	32	35	38
0.032	0.040	AD4	18	21	25	28	32	36	39	43	46
0.040	0.050	AD5		17	20	23	26	29	32	35	38
0.050	0.063	AD5		21	24	28	32	35	38	42	45
0.063	0.071	AD5		26	30	35	40	44	48	53	57
0.071	0.080	D6			17	20	23	25	27	30	33
0.080	0.090	D6			20	23	25	28	31	34	37
0.090	0.100	D6			22	25	28	32	35	38	41
0.100	0.125	D6				28	32	35	39	42	46
0.125	0.160	NAS1466H				16	18	20	22	24	26
0.160	0.190	NAS1466H					23	25	28	30	33
0.190		NAS1466H					27	30	33	36	39

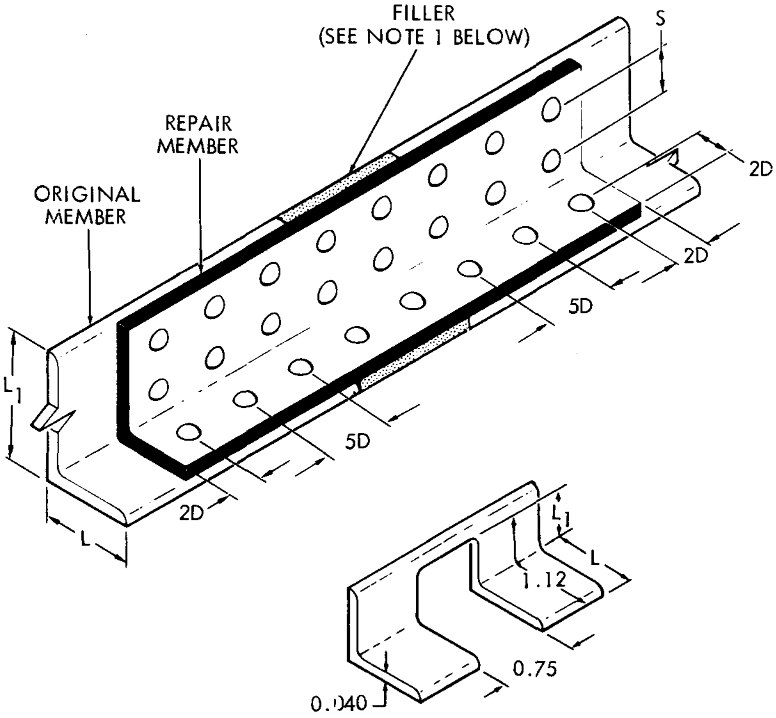
★ USE THICKNESS OF THICKEST FLANGE. IF EXACT GAGE IS NOT GIVEN, USE NEXT LARGER GAGE. JS-1-2-X0/2-5961

Figure 59-64. Typical Extruded "T" Repair



SECTION THROUGH REPAIRED AREA

REPAIR MEMBER
4130 STEEL COND N



NOTE

1. FILLER MEMBER TO BE SAME GAGE AND MATERIAL AS ORIGINAL ANGLE.
2. LAY OUT RIVET PATTERN AROUND EXISTING ORIGINAL ATTACHMENTS.
3. REPLACE ORIGINAL ATTACHMENTS WITH SAME DIAMETER AND TYPE ATTACHMENTS.
4. REPAIR MEMBER RADII SHOULD NEST OR BE PRACTICABLE MINIMUM. RIVET HEADS SHOULD NOT ENCROACH ON RADII.
5. MS20600M BLIND RIVETS MAY BE SUBSTITUTED FOR NAS508M RIVETS, PROVIDED THE NUMBER OF RIVETS REQUIRED IS INCREASED AT A RATIO OF 3 TO 2 FOR AD RIVETS, AND 2 TO 1 FOR D RIVETS, AND PROVIDED THE GAGE OF THE ORIGINAL ANGLE IS LESS THAN 0.125.

EXAMPLE

ASSUME A DAMAGED AREA, WHEN CLEANED UP, IS 0.75 INCH WIDE AND EXTENDS 1.12 INCHES ACROSS A 0.040 ANGLE. DETERMINE THE NUMBER OF NAS508M5 RIVETS REQUIRED BY INTERPOLATING BETWEEN 0.040-1 INCH, AND 0.040-1.25 INCHES, WHICH GIVES 6-1/2 RIVETS REQUIRED. USE 7 RIVETS. ALWAYS INTERPOLATE TO HIGHEST VALUE. NUMBER OF MS20600M BLIND RIVETS IS 3/2 X 7 = 10.5; USE 11 RIVETS EACH SIDE OF DAMAGED AREA.

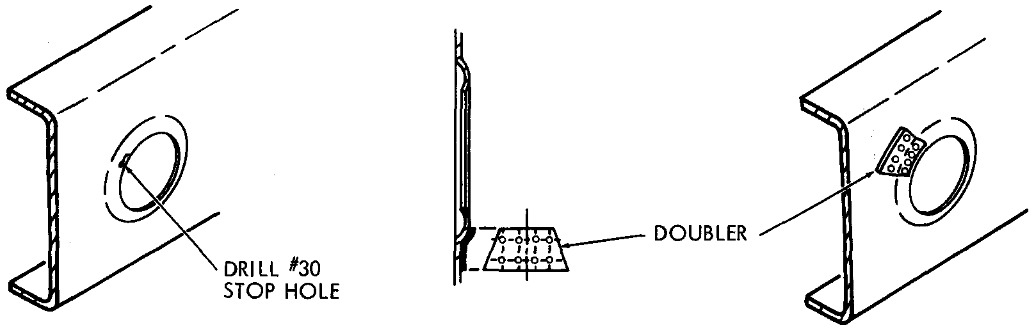
6. REFER TO SECTION 51.7 FOR CORROSION PREVENTION PRECAUTIONS FOR DISSIMILAR METALS.

RIVET REQUIREMENTS

NUMBER OF RIVETS REQUIRED ON EACH SIDE OF DAMAGED AREA. INTERPOLATE FOR INTERMEDIATE VALUES

ANGLE GAGE	REPAIR MEMBER GAGE	LONGITUDINAL SPACING (5D)	MINIMUM TRANSVERSE SPACING (S)	MINIMUM EDGE DISTANCE	TYPE OF RIVET REQUIRED	NUMBER OF RIVETS REQUIRED WHEN L + L_1 IS									
						0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.50	3.00	4.00
0.025	0.032	0.63	0.55	0.25	NAS508M4	3	4	6	7	9	11	12	15	18	24
0.032	0.040	0.63	0.55	0.25	NAS508M4	4	5	7	9	11	12	14	18	21	28
0.040	0.050	0.78	0.67	0.31	NAS508M5	3	4	6	7	9	10	12	14	17	23
0.050	0.063	0.78	0.67	0.31	NAS508M5	4	6	7	9	11	12	14	18	21	29
0.063	0.071	0.94	0.81	0.38	NAS508M6	3	4	5	6	7	8	9	11	14	18
0.071	0.080	0.94	0.81	0.38	NAS508M6	3	4	5	7	8	9	10	13	15	20
0.080	0.090	0.94	0.81	0.38	NAS508M6	3	5	6	7	9	10	12	14	17	23
0.090	0.100	0.94	0.81	0.38	NAS508M6	4	5	7	8	10	11	13	16	19	26
0.100	0.125	1.25	1.07	0.50	NAS508M8	2	3	4	5	6	7	8	10	12	16
0.125	0.160	1.25	1.07	0.50	NAS508M8	3	4	5	6	8	9	10	12	15	20
0.160	0.190	1.25	1.07	0.50	NAS508M8	3	5	6	8	9	11	12	15	18	25
0.190	0.250	1.25	1.07	0.50	NAS508M8	4	6	8	9	11	13	15	18	22	29

Figure 59-65. Typical Extruded Angle Repair



APPLICATION

FOR REPAIR OF CRACK IN FLANGE OF BEADED LIGHTENING HOLE IN SECONDARY STRUCTURE (EQUIPMENT SUPPORT BRACKETS, ETC.).

PARTS REQUIRED

1. DOUBLER OF SAME MATERIAL, THICKNESS, AND CONTOUR AS HOLE FLANGE SHAPED TO TAKE RIVETS IN HOLE FLANGE AND IN WEB AS SHOWN.

   NOTE: DOUBLER MAY BE SIZED TO REPAIR ADJACENT CRACKS IF NECESSARY.

2. UNIVERSAL HEAD RIVETS (AN 470) OF SAME SIZE AND MATERIAL AS USED TO ATTACH PART BEING REPAIRED.

PROCEDURE

1. ROUT OUT CRACK IF CRACK IS LESS THAN 2/3 WIDTH BEADED FLANGE.
2. IF CRACK IS LONGER THAN 2/3 WIDTH OF FLANGE, DRILL #30 STOP HOLE AT END OF CRACK.
3. ATTACH DOUBLER OVER CRACK USING AT LEAST 8 RIVETS, 2 IN HOLE FLANGE AND 2 IN WEB AT EACH SIDE OF CRACK - SPACE AT 4 TIMES RIVET DIAMETER (APPROX) - ED AT 2 TIMES RIVET DIAMETER.
4. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-66. Lightening Hole Flange Crack-Secondary Structure

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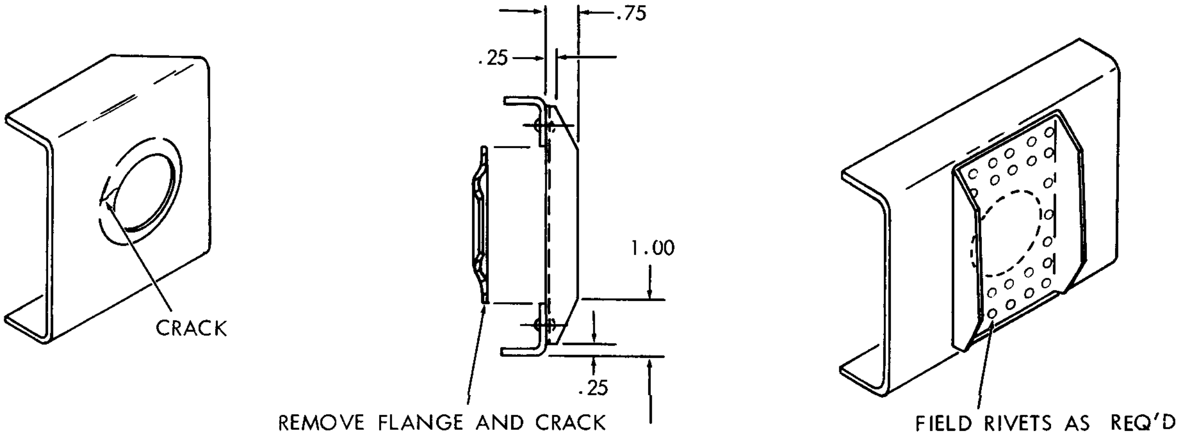


APPLICATION

FOR REPAIR OF CRACK IN FLANGE OF BEADED LIGHTENING HOLE IN PRIMARY STRUCTURE, SUCH AS EMPENNAGE RIBS.

PARTS REQUIRED

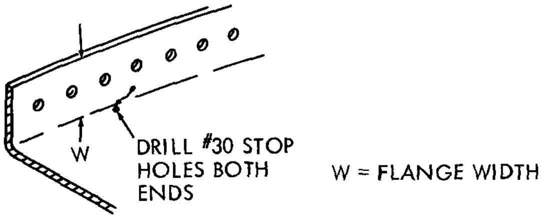
1. DOUBLER, ONE GAGE THICKER THAN PART BEING REPAIRED, SAME MATERIAL, LARGE ENOUGH TO GET ONE ROW OF RIVETS AROUND HOLE.
2. RIVETS OF SAME TYPE AND SIZE AS USED TO ATTACH PART BEING REPAIRED. HEAD AS REQUIRED.

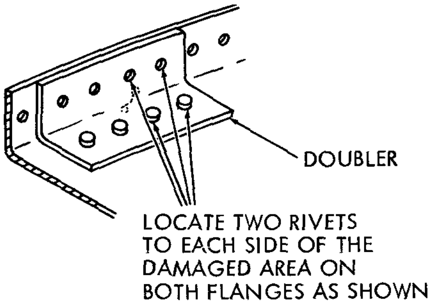
PROCEDURE

1. ROUT OUT CRACK AND FLANGE AROUND HOLE MAKING SURE ALL OF CRACK IS REMOVED.
2. INSTALL DOUBLER, USING ONE ROW OF RIVETS AROUND OUTER EDGE OF DOUBLER, SPACE SAME AS THOSE USED TO ATTACH PART BEING REPAIRED. ED = 2
3. INSTALL FIELD RIVETS AS REQUIRED TO PREVENT VIBRATION.
4. FINISH THE SAME AS PARTS BEING REPAIRED.

   NOTE: IF NECESSARY FOR ACCESS OR ROUTING OF LINES, WIRES, ETC., HOLE SAME SIZE AS ORIGINAL LIGHTENING HOLE MAY BE CUT IN DOUBLER.

Figure 59-67. Cracked Flange-Beaded Lightening Hole-Doubler Repair




DOUBLER
LOCATE TWO RIVETS TO EACH SIDE OF THE DAMAGED AREA ON BOTH FLANGES AS SHOWN

APPLICATION

FOR REPAIR OF CRACK IN BEND OF FORMED FLANGE.

LIMITATIONS

1. DO NOT USE IF LENGTH OF CRACK EXCEEDS 1 W.
2. DO NOT USE IF END RIVETS IN BOTH LEGS OF DOUBLER CANNOT BE LOCATED OUTSIDE DAMAGED AREA.
3. DO NOT USE IF ED AND ORIGINAL RIVET PATTERN OF FLANGE CANNOT BE MAINTAINED IN BOTH LEGS OF DOUBLER.
4. DO NOT USE IF DAMAGE IS CLOSER THAN 2 TIMES RIVET DIAMETER TO A RIVET HOLE. DO NOT USE FOR PRESSURIZED STRUCTURE.

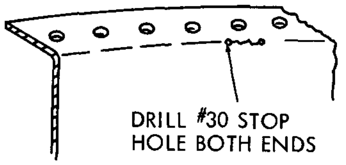
PARTS REQUIRED

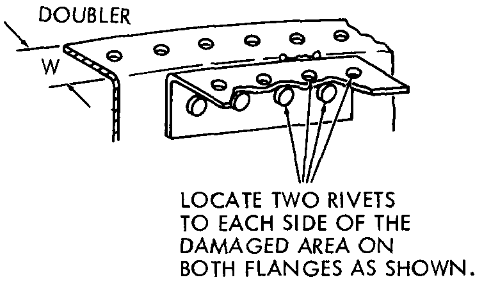
1. ONE FORMED ANGLE DOUBLER OF SAME MATERIAL, THICKNESS, CONTOUR, AND STANDARD BEND RADIUS AS DAMAGED FLANGE - LONG ENOUGH TO TAKE AT LEAST 4 RIVETS IN EACH LEG.
2. RIVETS AS REQUIRED OF SAME TYPE, KIND AND SIZE AS ORIGINAL.

PROCEDURE

1. DRILL #30 STOP HOLE AT EACH END OF CRACK.
2. INSTALL DOUBLER INSIDE FLANGE AND ATTACH TO WEB, AS SHOWN.
3. LOCATE TWO RIVETS TO EACH SIDE OF THE DAMAGED AREA ON BOTH FLANGES.
4. ON ASSEMBLY, ATTACH TO ADJACENT STRUCTURE BY RIVETING THROUGH FLANGE AND DOUBLER LEG.
5. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-68. Flange Bend Crack-Inside Doubler Repair





W = FLANGE WIDTH

APPLICATION

FOR REPAIR OF CRACK IN BEND OF FORMED FLANGE.

LIMITATIONS

1. DO NOT USE FOR PRESSURIZED STRUCTURE
2. DO NOT USE IF LENGTH OF CRACK EXCEEDS 1 W.
3. DO NOT USE IF END RIVETS IN BOTH LEGS OF DOUBLER CANNOT BE LOCATED OUTSIDE DAMAGED AREA.
4. DO NOT USE IF ED AND ORIGINAL RIVET PATTERN OF FLANGE CANNOT BE MAINTAINED IN BOTH LEGS OF DOUBLER.
5. DO NOT USE IF DAMAGE IS CLOSER THAN 2 TIMES RIVET DIAMETER TO A RIVET HOLE.

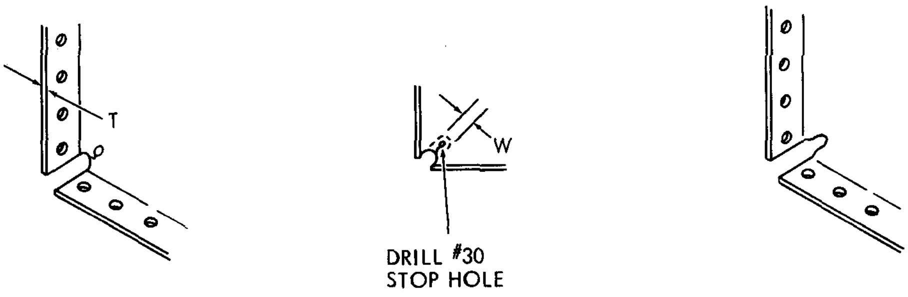
PARTS REQUIRED

1. ONE FORMED ANGLE DOUBLER OF SAME MATERIAL, THICKNESS, CONTOUR AND STANDARD BEND RADIUS AS DAMAGED FLANGE LONG ENOUGH TO TAKE AT LEAST 4 RIVETS IN EACH LEG.
2. RIVETS AS REQUIRED OF SAME TYPE, KIND AND SIZE AS ORIGINAL.

PROCEDURE

1. DRILL #30 STOP HOLE AT EACH END OF CRACK.
2. INSTALL DOUBLER BY MOUNTING ON WEB AND ATTACH AS SHOWN.
3. LOCATE TWO RIVETS TO EACH SIDE OF THE DAMAGED AREA ON BOTH FLANGES.
4. ON ASSEMBLY, ATTACH OTHER LEG OF DOUBLER DIRECTLY TO SKIN OR ADJACENT STRUCTURE.
5. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-69. Flange Bend Crack-Outside Doubler Repair



APPLICATION

FOR REPAIR OF CORNER CRACK IN DOUBLE FORMED FLANGE.

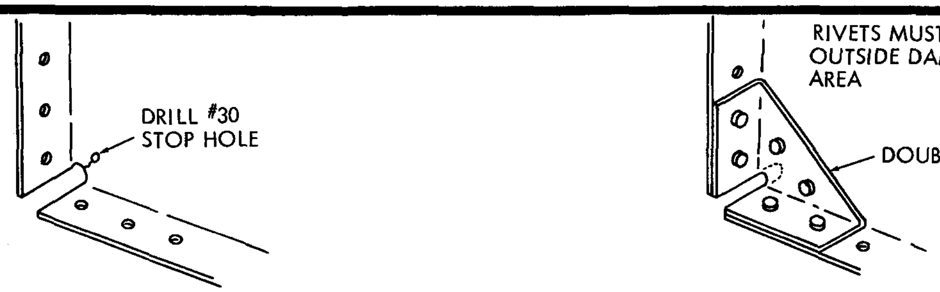
LIMITATIONS

1. DO NOT USE IF LENGTH OF CRACK EXCEEDS 8T.
2. DO NOT USE FOR PRIMARY STRUCTURE.

PROCEDURE

1. DRILL #30 STOP HOLE AT END OF CRACK.
2. CUT OUT WEB MATERIAL FROM STOP HOLE OUTWARD TO BEND RELIEF RADIUS USING A MINIMUM RADIUS of .12, WITH WIDTH OF CUTOUT (W) NOT LESS THAN .25 INCH.
3. DE-BURR EDGES OF CUTOUT.

Figure 59-70. Corner Crack-Formed Flange-Cut-Out Repair



APPLICATION

FOR REPAIR OF CORNER CRACK IN DOUBLE FORMED FLANGE.

LIMITATIONS

1. DO NOT USE IF LENGTH OF CRACK EXCEEDS 25T.
2. DO NOT USE IF ORIGINAL RIVET PATTERN CANNOT BE MAINTAINED.

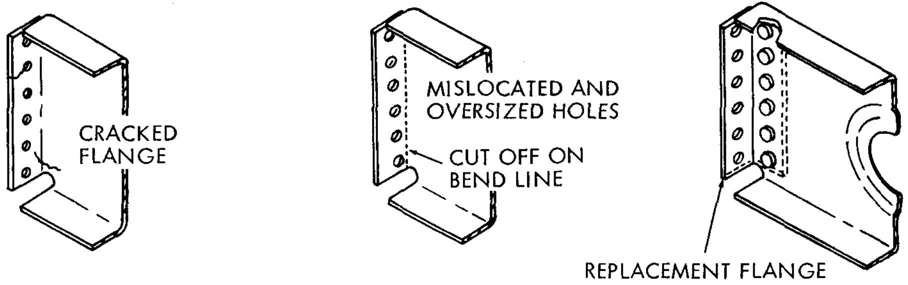
PARTS REQUIRED

1. ONE FORMED ANGLE DOUBLER OF SAME MATERIAL, THICKNESS, CONTOUR, AND STANDARD BEND RADIUS AS DAMAGED FLANGE LONG ENOUGH TO TAKE AT LEAST 6 RIVETS AS SHOWN.
2. RIVETS AS REQUIRED OF SAME TYPE, KIND AND SIZE AS ORIGINAL.

PROCEDURE

1. DRILL #30 STOP HOLE AT END OF CRACK.
2. INSTALL DOUBLER INSIDE FLANGE CORNER AND ATTACH TO WEB WITH 2 RIVETS BEYOND END OF CRACK AS SHOWN. ON ASSEMBLY, ATTACH FLANGE AND DOUBLER TO SKIN OR ADJACENT STRUCTURE.
3. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-71. Corner Crack-Formed Flange-Doubler Repair



APPLICATION

FOR REPAIR OF CRACKS AND MISLOCATED, ELONGATED OR OVERSIZE HOLES IN FORMED FLANGE OF RIB, BULKHEAD OR SIMILAR MEMBER.

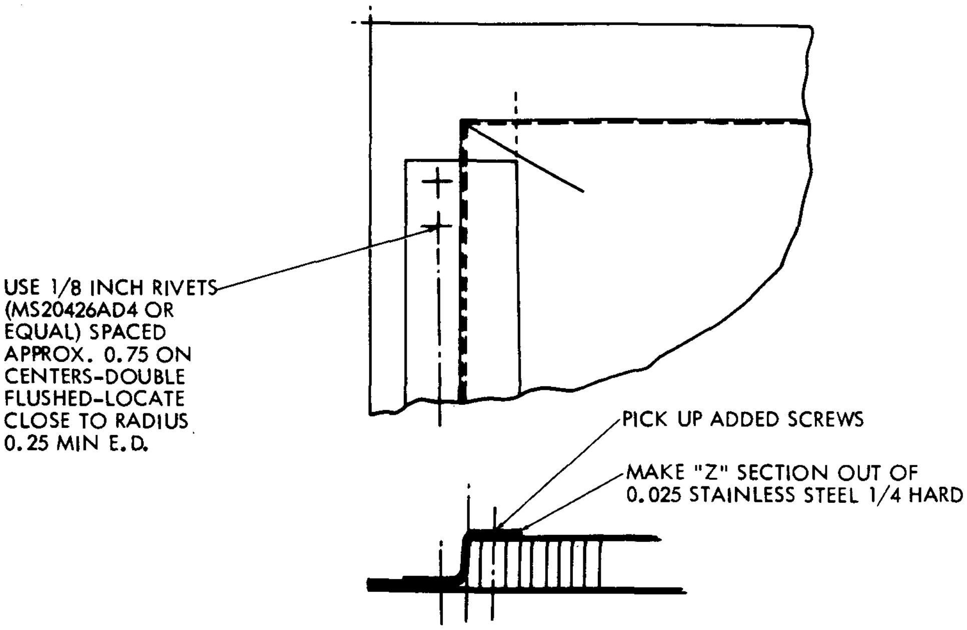
PARTS REQUIRED

1. FORMED REPLACEMENT FLANGE OF SAME MATERIAL, THICKNESS, AND STANDARD BEND RADIUS AS DAMAGED FLANGE-SIZE, BEND RADIUS AND CONTOUR TO FIT AS SHOWN AND MAINTAIN ORIGINAL RIVET PATTERN FOR ATTACHMENT TO ADJACENT STRUCTURE.
2. RIVETS AS REQUIRED.

PROCEDURE

1. CUT OFF DAMAGED FLANGE ON BEND LINE AS SHOWN AND DE-BURR CUT EDGES.
2. INSTALL REPLACEMENT FLANGE AS SHOWN AND ATTACH TO WEB WITH SAME RIVET TYPE, SPACING AND ED AS PRESCRIBED FOR ATTACHMENT OF FLANGE TO ADJACENT STRUCTURE.
3. ATTACH REPLACEMENT FLANGE TO ADJACENT STRUCTURE UPON ASSEMBLY, MAINTAINING ORIGINAL RIVET PATTERN.
4. FINISH THE SAME AS PARTS BEING REPAIRED.

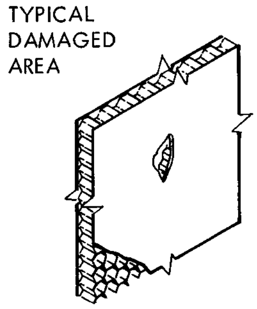
Figure 59-72. Damaged Flange-Replacement Repair

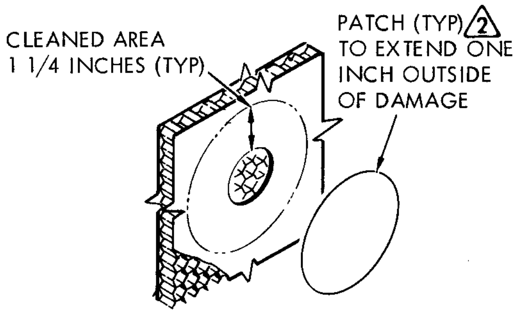

USE 1/8 INCH RIVETS (MS20426AD4 OR EQUAL) SPACED APPROX. 0.75 ON CENTERS-DOUBLE FLUSHED-LOCATE CLOSE TO RADIUS 0.25 MIN E.D.
PICK UP ADDED SCREWS
MAKE "Z" SECTION OUT OF 0.025 STAINLESS STEEL 1/4 HARD

NOTE

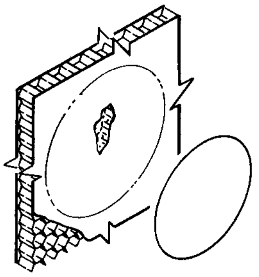
1. STOP DRILL CRACK
2. "Z" SECTION SHOULD EXTEND AT LEAST 0.75 INCH PAST END OF CRACK
3. ADJUST DOOR AS REQUIRED
4. IN AREA OF CRACK ON EITHER DOOR (NEAR UPLOCK HOOK), "Z" SECTION SHOULD RUN UNDER -31 BEAM AND PICKUP EXISTING FASTENERS.

Figure 59-73. Honeycomb Repair, JL 371 Door

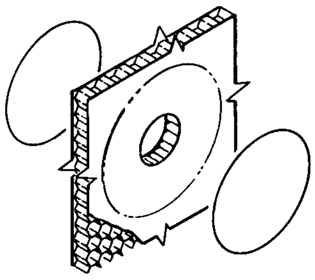




A



B



C

REPAIR A (DAMAGE TO ONE FACE ONLY)

1. CLEAN DAMAGED FACE 1 1/4 INCHES OUTSIDE OF DAMAGE.
   a. REMOVE PAINT AND OTHER ORGANIC FILMS, AND ABRADE SURFACE OF FACING LIGHTLY; USE NO. 200 GRIT SANDPAPER.
   b. LIGHTLY SAND MATING SURFACES OF PATCH.△1
   c. CLEAN PATCH AND DAMAGED AREA WITH CLEAN KETONE OR ACETONE AND CLEAN WHITE CLOTHS; WIPE DRY WITH A CLEAN WHITE CLOTH.
2. CLEANED UP DAMAGED FACE NOT TO EXCEED 2 INCHES IN DIAMETER.
3. APPLY SPECIFICATION MMM-A-132, TYPE I, CLASS 3 EPOXY ADHESIVE △3 TO DAMAGED FACE AND TO EXPOSED HONEYCOMB SO AS TO FILL HONEYCOMB CELLS FLUSH WITH SURFACE OF DAMAGED FACE, AND SMOOTH IT. APPLY THIN COAT OF EPOXY ADHESIVE TO ABRADED SURFACE OF PATCH.
4. APPLY CIRCULAR PATCH TO DAMAGED FACING (PATCH TO EXTEND 1 INCH OUTSIDE OF DAMAGE), AND SMOOTH OUT WITH HANDS OR HARD-SURFACE ROLLER. REMOVE EXCESS ADHESIVE SQUEEZED OUT, AND FAIR ADHESIVE TO FACING AROUND EDGE OF PATCH.
5. APPLY MASKING TAPE OVER PATCH AND FACING TO HOLD PATCH IN PLACE UNTIL ADHESIVE CURES.△4
6. REMOVE TAPE AFTER ADHESIVE CURES, AND PAINT REPAIRED AREA.

REPAIR B (DAMAGE TO ONE FACING AND CORE)

1. CUT OUT THE DAMAGED CORE AND FACING AND ROUTE THE HONEYCOMB CORE BACK 1/2 INCH FROM THE CLEANED UP EDGE OF THE DAMAGED FACE.
2. CLEANED UP DAMAGED AREA NOT TO EXCEED 3 INCHES IN DIAMETER.
3. CLEAN FACING AND PATCH AS IN REPAIR A.
4. FILL CORE VOID WITH EPOXY ADHESIVE.
5. APPLY PATCH, CURE AND PAINT AS IN REPAIR A.

REPAIR C (DAMAGE TO BOTH FACINGS AND CORE)

1. CUT OUT BOTH DAMAGED FACES AND CORE AND ROUTE THE HONEYCOMB CORE BACK 1 INCH FROM THE CLEANED UP EDGE OF THE DAMAGED FACES.
2. CLEANED UP DAMAGED AREA NOT TO EXCEED 3 INCHES IN DIAMETER.
3. APPLY BACKUP PATCH ON ONE FACE AS INSTRUCTED IN REPAIR A.
4. AFTER BACKUP PATCH HAS CURED, COMPLETE REPAIR AS IN REPAIR B.

NOTE

△1 MAKE PATCH OF SAME MATERIAL AND THICKNESS AS EXISTING FACING.

△2 ON PANELS SHEATHED WITH FIBERGLAS, COVER REPAIR PATCH WITH FIBERGLAS PATCH.

△3 SPECIFICATION MMM-A-132, TYPE 1, CLASS 3 ADHESIVE MUST BE PURCHASED IN PASTE FORM SUITABLE FOR APPLICATION TO VERTICAL SURFACE.

△4 INITIAL CURE WILL REQUIRE 24 HOURS AT 72°F TO 80°F, BUT FULL CURE REQUIRES 7 DAYS AT 72°F TO 80°F. DO NOT SUBJECT TO DESIGN LOADINGS PRIOR TO FULL CURE. AN ACCELERATED CURE IS DESIRABLE AND CONSISTS OF 2 HOURS AT 72°F TO 80°F, FOLLOWED BY 2 HOURS AT 150°F.

JS-1-2-X0/2-5965

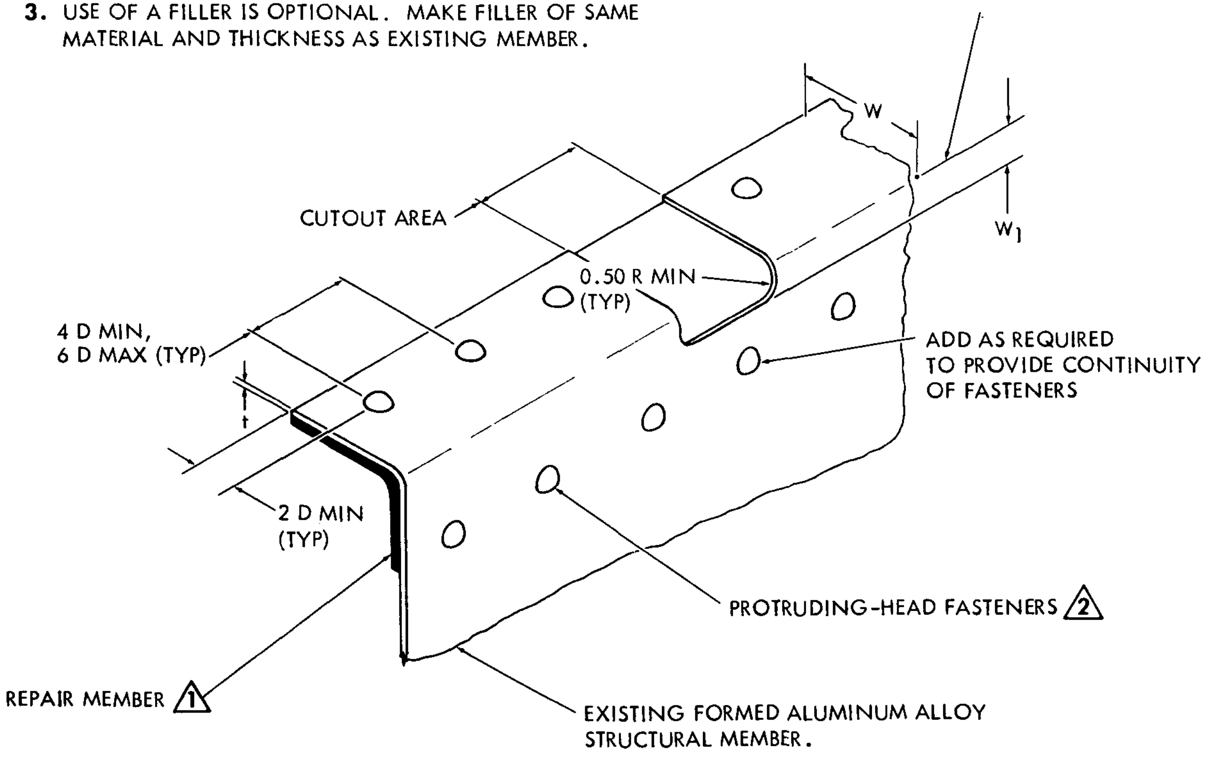
Figure 59-74. Typical Honeycomb Panel Repair

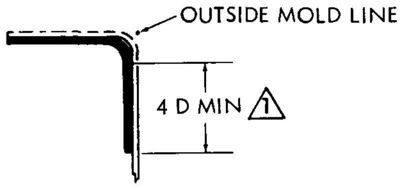
NOTE

△1 MAKE REPAIR MEMBER OF SAME MATERIAL AS EXISTING MEMBER, BUT ONE THICKNESS HEAVIER. PROVIDE AT LEAST THE MINIMUM CROSS-SECTIONAL AREA AS SHOWN.

△2 USE ALUMINUM ALLOY PROTRUDING HEAD FASTENERS AS SHOWN. WHEN TWO OR MORE ROWS OF FASTENERS ARE REQUIRED, USE AN EQUAL NUMBER OF FASTENERS IN EACH ROW. IF THE NUMBER REQUIRED IS ODD, USE THE NEXT HIGHEST EVEN NUMBER. FOR INSTANCE, IF 7 ARE REQUIRED AND TWO ROWS ARE SELECTED; USE 8 FASTENERS, 4 IN EACH ROW.

3. USE OF A FILLER IS OPTIONAL. MAKE FILLER OF SAME MATERIAL AND THICKNESS AS EXISTING MEMBER.

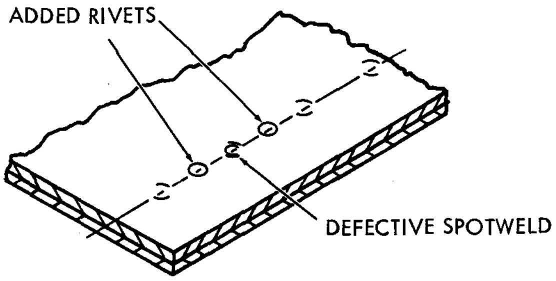




CROSS-SECTION
AT CUTOUT AREA

t	FASTENER SIZE (D)	NUMBER OF FASTENERS REQUIRED EACH SIDE OF CUTOUT AREA WHEN $W + W_1$ IS:						
		0.50	0.75	1.00	1.25	1.50	1.75	2.00
0.025	1/8	3	4	6	7	9	11	12
0.032	1/8	4	5	7	9	11	12	14
0.040	5/32	3	4	6	7	9	10	12
0.050	5/32	4	6	7	9	11	12	14
0.063	3/16	3	4	5	6	7	8	9
0.071	3/16	3	4	5	7	8	9	10
0.080	3/16	3	5	6	7	9	10	12
0.090	3/16	4	5	7	8	10	11	13

Figure 59-75. Typical Repair for Formed Aluminum Alloy Sections



APPLICATION

FOR REPAIR OF DEFECTIVE SPOTWELDS.

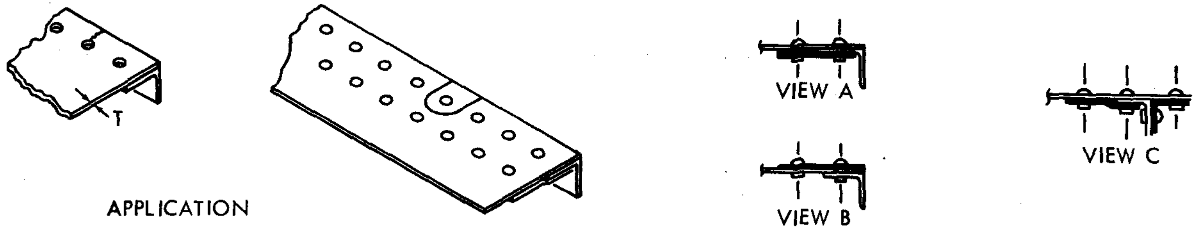
PARTS REQUIRED

"B" TYPE RIVETS, DIAMETER COMPATIBLE WITH PARTS BEING REPAIRED.

PROCEDURE

LOCATE RIVETS AS SHOWN, CENTERED EACH SIDE OF DEFECTIVE SPOTWELD, ON SPOTWELD, ON SPOTWELD CENTERLINE. DOUBLE FINISH RIVETS MAY BE USED IF NECESSARY.

Figure 59-76. Defective Spotweld-Rivet Repair



APPLICATION

FOR REPAIR OF CRACKED RIVET HOLE IN SHEET METAL.

LIMITATIONS

1. DO NOT USE IF CRACK CAN BE COMPLETELY REMOVED BY USING NEXT LARGER SIZE RIVET.
2. DO NOT USE IF CRACK IS LONGER THAN 10T.
3. DO NOT USE IF MORE THAN 2 OUT OF 5 CONSECUTIVE HOLES OR OVER 10% OF ALL HOLES IN PATTERN ARE CRACKED.
4. DO NOT USE IF ED = 2 CANNOT BE MAINTAINED.

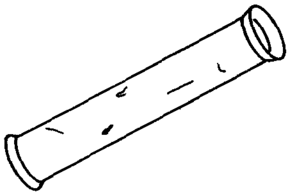
PARTS REQUIRED

1. SHEET METAL PLUG OF SAME THICKNESS AND MATERIAL AS DAMAGED SHEET, SHAPED TO FIT CUTOUT AREA AS SHOWN.
2. DOUBLER OF SAME MATERIAL AND ONE GAGE THICKER THAN DAMAGED SHEET - LONG ENOUGH TO TAKE 3 RIVETS ON EACH SIDE OF CRACKED HOLE WIDE ENOUGH TO TAKE ADDITIONAL ROW OF RIVETS - JOGGLED FROM SHEET OVER SUPPORTING STRUCTURE. FOR INTERNAL STRUCTURE, A FLAT DOUBLER MAY BE USED AS SHOWN IN VIEW B. THE JOGGLED DOUBLER MAY BE REPLACED WITH A SHIM AND FLAT DOUBLER AS SHOWN IN VIEW A. WHEN CRACKED HOLES OCCUR IN FULL SHEET ATTACHMENT, SUCH AS SKIN STRINGERS, AN ADDITIONAL DOUBLER ON THE OPPOSITE SIDE OF THE STRINGER MUST BE USED AS SHOWN IN VIEW C.
3. RIVETS OF SAME SIZE, KIND AND TYPE AS ORIGINAL - LENGTH AS REQUIRED.

PROCEDURE

1. CUT OUT CRACKED HOLE MAINTAINING ED OF SURROUNDING RIVETS TO 1-1/2 (MIN) TIMES RIVET SHANK DIAMETER. USE MAXIMUM RADIUS ON CUTOUT CONSISTENT WITH PRESCRIBED ED.
2. ATTACH DOUBLER TO SHEET USING RIVETS AS PRESCRIBED AND SPACING SAME AS ORIGINAL PATTERN. MAINTAIN ED ON DOUBLER OF TWICE RIVET DIAMETER.
3. ATTACH SHEET AND DOUBLER TO SUPPORTING MEMBER AND INSTALL.
4. FINISH THE SAME AS PARTS BEING REPAIRED.

Figure 59-77. Cracked Rivet Hole-Plug Repair



APPLICATION

FOR THE REPAIR OF SUPERFICIAL SCRATCHES, ABRASIVE MARKS, CUTS, AND NICKS IN ALUMINUM ALLOY TUBING. RESTRICTED TO DUCTING, VENT TUBES, DRAIN LINES, AND PIPING LINES WHEN PRESSURES DO NOT EXCEED 15 PSI.

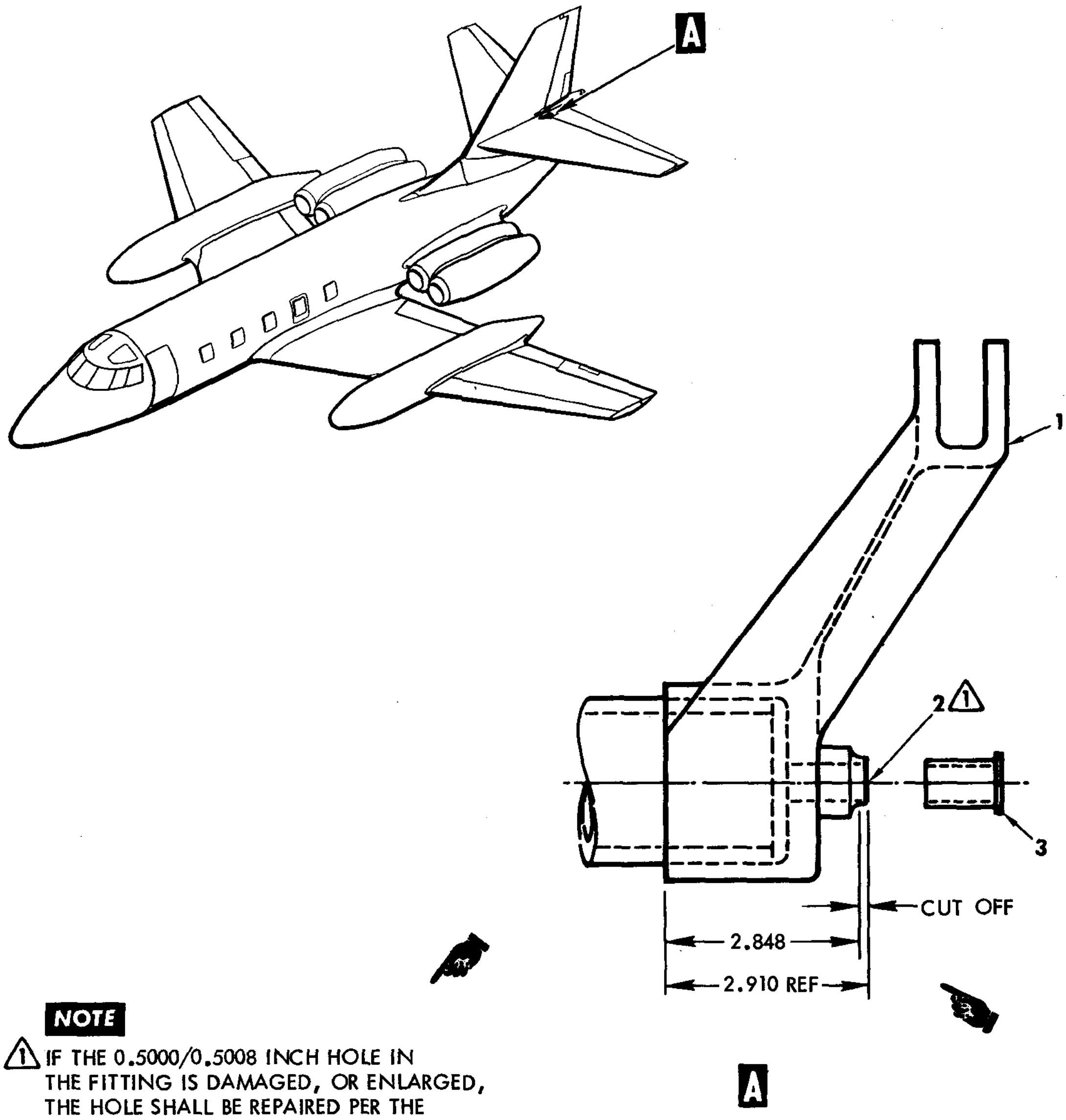
LIMITATIONS

1. DO NOT USE IF DEFECTS EXCEED 10% OF THE TUBING WALL THICKNESS.

PROCEDURE

1. BLEND IN DEFECTS WITH #400 WET OR DRY SANDPAPER OR ALUMINUM WOOL. DO NOT REMOVE MATERIAL BELOW BOTTOM OF FLAW. SURFACE FINISH OF REWORKED AREA SHALL NOT EXCEED RMS 100.
2. FINISH SAME AS ORIGINAL PART.

Figure 59-78. Surface Damage-Tubing



NOTE

△1 IF THE 0.5000/0.5008 INCH HOLE IN THE FITTING IS DAMAGED, OR ENLARGED, THE HOLE SHALL BE REPAIRED PER THE FOLLOWING PROCEDURE.

STEP 1

CUT OFF FITTING (1) AND ENLARGE THE HOLE (2) TO 0.625/0.6259 INCH AS SHOWN IN DETAIL A.

STEP 2

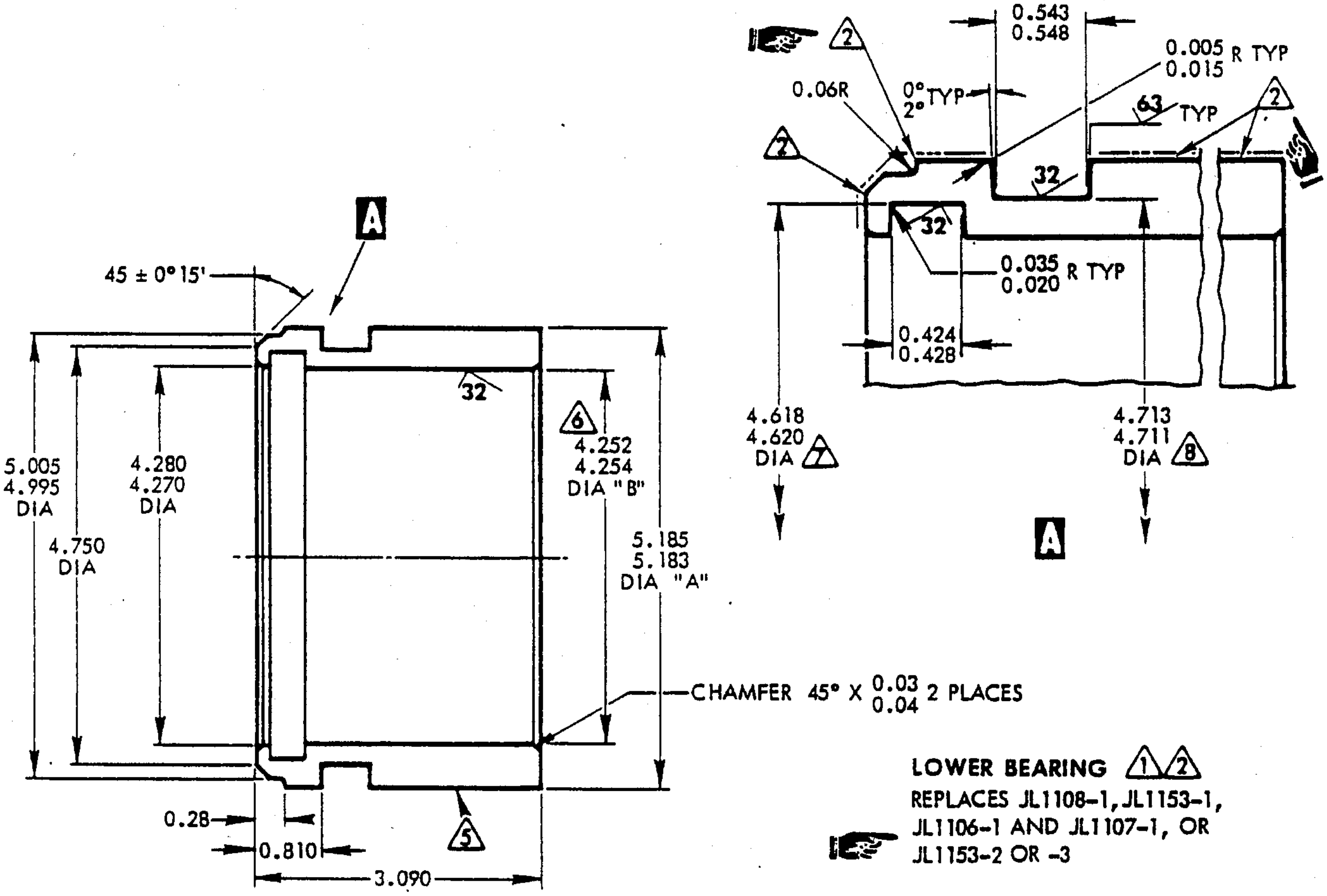
FINISH THE HOLE PER HOMI SECTION 51.8

STEP 3

PRESS AN LS3859-8-34 BUSHING (3) INTO THE ENLARGED HOLE.

JS-59-0-079

Figure 59-79. Elevator Horn Fitting (JE190-1) Repair



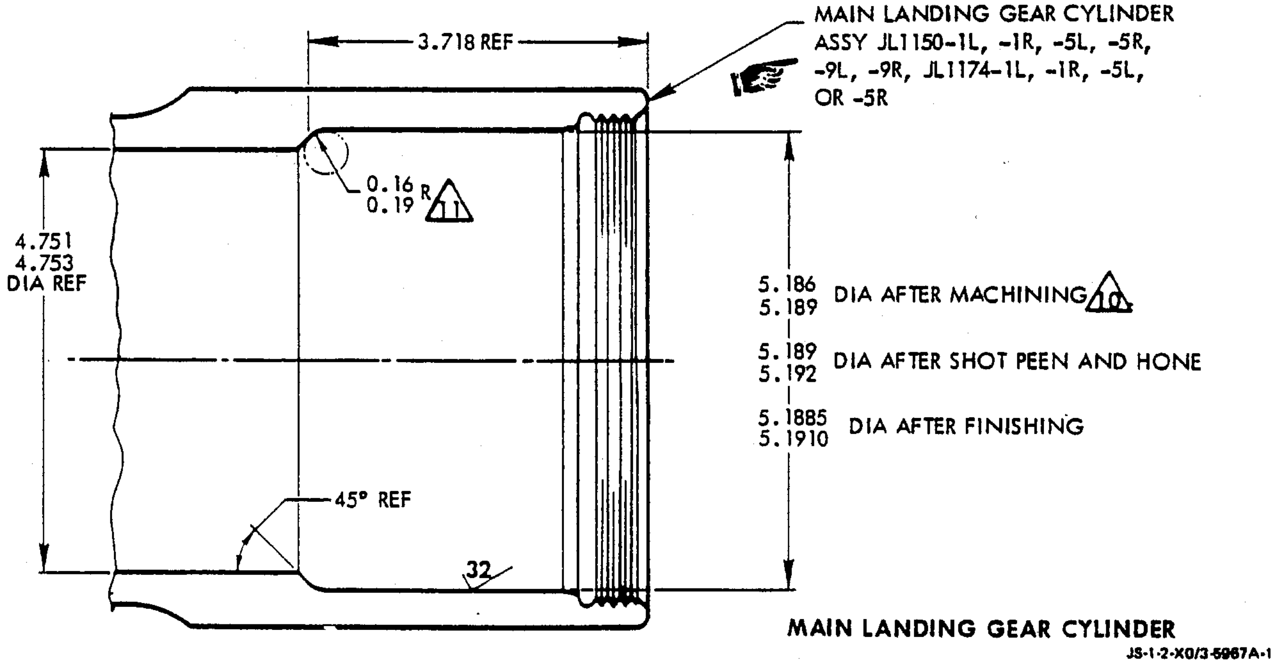

MAIN LANDING GEAR CYLINDER ASSY JL1150-1L, -1R, -5L, -5R, -9L, -9R, JL1174-1L, -1R, -5L, OR -5R
3.718 REF
0.16
0.19 R 11
4.751
4.753
DIA REF
5.186
5.189 DIA AFTER MACHINING 10
5.189
5.192 DIA AFTER SHOT PEEN AND HONE
5.1885
5.1910 DIA AFTER FINISHING
45° REF
32
MAIN LANDING GEAR CYLINDER
JS-1-2-X0/3-5967A-1

Figure 59-80. Main Landing Gear Cylinder Rework (Sheet 1 of 2)

NOTE

△1 MAKE FROM AL BRONZE WROUGHT BAR PER QQ-C-465B, ALLOY C63000. NO HEAT TREAT REQUIRED.

△2 CADMIUM PLATE SURFACE PER QQ-P-416, CLASS 3.

3. MACHINE SURFACE FINISH √150 EXCEPT AS SHOWN.

4. MACHINE SURFACES PARALLEL AND PERPENDICULAR TO CENTERLINE WITHIN 0.020 IN 20 INCHES.

△5 RUBBER STAMP PART NUMBER HERE.

△6 CONCENTRIC WITH DIAMETER "A" WITHIN 0.003.

△7 CONCENTRIC WITH DIAMETER "B" WITHIN 0.001.

△8 CONCENTRIC WITH DIAMETER "A" WITHIN 0.001.

9. PENETRANT INSPECT PER PARAGRAPH 58.1.3. ☞

△10 NEW MACHINED SURFACE AS SHOWN MUST BE CONCENTRIC WITH REFERENCED DIAMETER WITHIN 0.003.

△11 BLEND RADIUS INTO EXISTING 45 DEGREE MACHINED SURFACE.

12. SHOT PEEN MACHINED AREAS PER MIL-S-13165. INTENSITY 0.010 TO 0.016 (ALMEN A_2). MASK OTHER AREAS TO PREVENT DAMAGE TO CHEMICAL FINISH.

13. ANODIZE MACHINED AREAS PER MIL-A-8625, CLASS II OR IIA 0.0003 TO 0.0005 THICK. OPTIONAL: A LESS EFFECTIVE CORROSION CONTROL METHOD MAY BE OBTAINED BY TREATING THE SURFACE AS INDICATED IN THE HOMI PARAGRAPH 51.7.3.9 (ALODINE 1200S).

14. BREAK SHARP EDGES 0.01/0.03.

JS-1-2-X0/7-6967A-2

Figure 59-80. Main Landing Gear Cylinder Rework (Sheet 2 of 2)

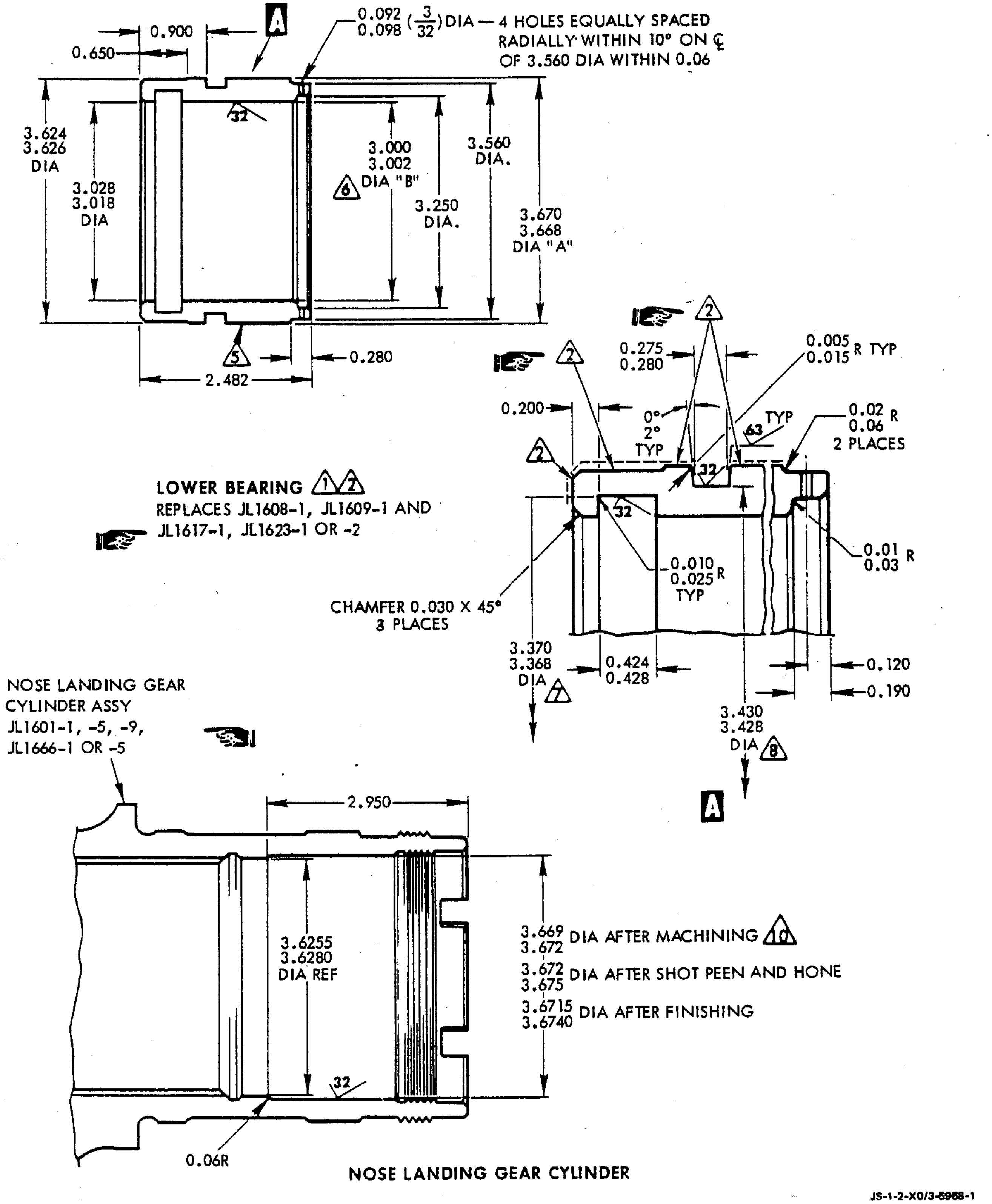

0.900
0.650
A
0.092 / 0.098 (3/32) DIA — 4 HOLES EQUALLY SPACED RADIALLY WITHIN 10° ON ℄ OF 3.560 DIA WITHIN 0.06
3.624 / 3.626 DIA
32
3.028 / 3.018 DIA
3.000 / 3.002 DIA "B"
6
3.250 DIA.
3.560 DIA.
3.670 / 3.668 DIA "A"
5
0.280
2.482
2
0.275 / 0.280
0.005 / 0.015 R TYP
0.200
0° / 2° TYP
63 TYP
0.02 / 0.06 R 2 PLACES
LOWER BEARING 1 2
REPLACES JL1608-1, JL1609-1 AND JL1617-1, JL1623-1 OR -2
0.010 / 0.025 R TYP
0.01 / 0.03 R
CHAMFER 0.030 X 45° 3 PLACES
3.370 / 3.368 DIA 7
0.424 / 0.428
0.120
0.190
3.430 / 3.428 DIA 8
NOSE LANDING GEAR CYLINDER ASSY JL1601-1, -5, -9, JL1666-1 OR -5
2.950
3.6255 / 3.6280 DIA REF
3.669 / 3.672 DIA AFTER MACHINING 10
3.672 / 3.675 DIA AFTER SHOT PEEN AND HONE
3.6715 / 3.6740 DIA AFTER FINISHING
0.06R
NOSE LANDING GEAR CYLINDER
JS-1-2-X0/3-5968-1

Figure 59-81. Nose Landing Gear Cylinder Rework (Sheet 1 of 2)

NOTE

△1 MAKE FROM AL BRONZE WROUGHT BAR PER QQ-C-465B, ALLOY C63000. NO HEAT TREAT REQUIRED.

△2 CADMIUM PLATE SURFACE PER QQ-P-416, CLASS 3.

3. MACHINE SURFACE FINISH √150 EXCEPT AS SHOWN.

4. MACHINE SURFACES PARALLEL AND PERPENDICULAR TO CENTERLINE WITHIN 0.020 IN 20 INCHES.

△5 RUBBER STAMP PART NUMBER HERE.

△6 CONCENTRIC WITH DIAMETER "A" WITHIN 0.003.

△7 CONCENTRIC WITH DIAMETER "B" WITHIN 0.001.

△8 CONCENTRIC WITH DIAMETER "A" WITHIN 0.001.

9. PENETRANT INSPECT PER PARAGRAPH 58.1.3. ☛

△10 NEW MACHINED SURFACE AS SHOWN MUST BE CONCENTRIC WITH REFERENCED DIAMETER WITHIN 0.003.

11. SHOT PEEN MACHINED AREAS PER MIL-S-13165. INTENSITY 0.010 TO 0.016 (ALMEN A_2). MASK OTHER AREAS TO PREVENT DAMAGE TO CHEMICAL FINISH.

12. ANODIZE MACHINED AREAS PER MIL-A-8625, CLASS II OR IIA 0.0003 TO 0.0005 THICK. OPTIONAL: A LESS EFFECTIVE CORROSION CONTROL METHOD MAY BE OBTAINED BY TREATING THE SURFACE AS INDICATED IN THE HOMI PARAGRAPH 51.7.3.9 (ALODINE 1200S).

13. BREAK SHARP EDGES 0.01/0.03.

14. DIMENSIONS FOR REMOVAL OF CYLINDER MATERIAL SHOWN IS THE MAXIMUM ALLOWED. LESS MAY BE REMOVED WHEN DIMENSIONS OF LOWER BEARING ARE INCREASED THE SAME AMOUNT. MAXIMUM REMOVAL FROM NOSE CYLINDER WILL REMOVE A FEW THOUSANDTHS FROM TOP OF THREAD. THIS IS OF NO CONSEQUENCE.

JS-1-2-X0/7-5968-2

Figure 59-81. Nose Landing Gear Cylinder Rework (Sheet 2 of 2)

SECTION 59.9

10,000 HOUR INSPECTION

Contents Page

GENERAL . . . 59-151
  Terminology . . . 59-152

INSPECTION AREAS . . . 59-152

INSPECTION TECHNIQUES — DETAILED VISUAL INSPECTION . . . 59-152

PREPARING AIRCRAFT PART OR AREA FOR DETAILED VISUAL INSPECTION . . . 59-152

WING INSPECTION DATA . . . 59-152
  Inspection of Wing Leading Edge . . . 59-152
  Inspection of Wing Root Top and Bottom Surfaces . . . 59-152
  Inspection of Leading Edge Flap Actuator Attach Region . . . 59-152
  Inspection of Aileron Bell Crank Support Region . . . 59-152

EMPENNAGE INSPECTION DATA . . . 59-152
  Inspection of Vertical Stabilizer Leading Edge . . . 59-152
  Inspection of Vertical Stabilizer Forward Support Region . . . 59-153
  Inspection of Horizontal Stabilizer Leading Edge . . . 59-153
  Inspection of Horizontal Stabilizer Box Section . . . 59-153
  Inspection of Elevator Outboard Counterweight Region . . . 59-153

FUSELAGE INSPECTION DATA . . . 59-153
  Inspection of Skin Splices at Fuselage Station 232 to 277 . . . 59-153
  Inspection of Fuselage Station 410 Skin Splice . . . 59-153
  Inspection of Fuselage Station 526 to 539 Skin Splice . . . 59-153

FIGURES

59-82 10,000 Hour Inspection Areas . . . 59-154
59-83 Inspection of Wing Leading Edges . . . 59-158
59-84 Inspection of Wing Root Upper and Lower Surfaces . . . 59-171
59-85 Inspection of Leading Edge Flap Actuator Region . . . 59-183
59-86 Inspection of Aileron Bellcrank Support Region . . . 59-188
59-87 Inspection of Vertical Stabilizer Leading Edge . . . 59-190
59-88 Inspection of Vertical Stabilizer Forward Support Region . . . 59-191
59-89 Inspection of Horizontal Stabilizer Leading Edge . . . 59-192
59-90 Inspection of Horizontal Stabilizer Box Section . . . 59-194
59-91 Inspection of Elevator Outboard Counterweight Region . . . 59-199
59-92 Inspection of Fuselage Station 232 to 277 Skin Splice . . . 59-200
59-93 Inspection of Fuselage Station 410 Skin Splices . . . 59-207
59-94 Inspection of Fuselage Station 526 to 539 Skin Splices . . . 59-210

59.9.1 GENERAL. The information in this section of the manual outlines the requirement for the 10,000 hour inspection and has been prepared to assist personnel in the techniques required to inspect the structure of the JetStar airplanes for cracks, breaks, corrosion, and other types of material failure.

a. The inspection areas of JetStar airplanes which require visual inspection are known areas where defects would prevent the items from performing their designed function.

b. Specific instructions peculiar to each part, area, or assembly being visually inspected is contained in this section.

c. Special instructions for parts, areas, or assemblies requiring radiographic, magnetic particle, penetrant, ultrasonic, or eddy current inspection at 10,000 hour intervals are contained in Section 58. (See figure 59-82.)

59.9.1.1 TERMINOLOGY. Such terms as "left," "right," "upper," "lower," "front," "rear," "forward," and "aft" refer to the airplane as viewed from the rear, on the centerline, looking forward. All dimensions and distances are given in inches unless otherwise noted.

59.9.2 INSPECTION AREAS. The structure in the shaded areas of figure 59-82 are referred to as "inspection areas." The index table accompanying the figure describes each area and references the paragraph which describes in detail the inspection procedure, method, and figure number of the illustration required. Inspection in all areas covered by figure 59-82 will be accomplished as indicated in paragraph 5.5.19.

59.9.3 INSPECTION TECHNIQUES — DETAILED VISUAL INSPECTION. Nondestructive inspection by visual means is by far the oldest and most economical method. Those defects which would escape detection by the naked eye may be seen when magnified. Optical devices for visual inspections are borescopes, magnifying glasses, mirrors, and flashlights. (Refer to figure 58-0-4.) The use of optical devices has two purposes:

a. To magnify defects which could not otherwise be seen by the naked eye.

b. To permit visual inspection of areas which are not accessible to the unaided eye. (Note: All visual inspections are to be performed with a 10X magnifying glass).

Detail visual inspections per paragraphs 59.1.2.3 and 51.7.2.5.

59.9.4 PREPARING AIRCRAFT PART OR AREA FOR DETAILED VISUAL INSPECTION. Remove access covers or doors to gain access to the part or area being inspected (Refer to Section 52.3). (Note: Upper wing surface planks are removed for this inspection.) Remove all foreign matter from the part or area to be inspected. Visual inspections require cleanliness of the part or area being inspected. The cleaning technique to be used will be determined by the type of foreign matter present. The removal of foreign matter must be accomplished by a method that will not effect a defect or continuity. Do not use shot blast, sand blast, grit blast, pressure blast, liquid honing, emery cloth, sandpaper, steel wire brush, or metal scraper. These will cover up defects by peening or cold working the surface, particularly on soft metals. After inspection has been performed, the original protective finish must be restored to the part or area. Refer to paragraphs 58.0.4.1 and 58.0.5 for general pre and post cleaning procedures and to Section 51.8 for general finish removal and restoration procedures.

59.9.5 WING INSPECTION DATA. Areas of the wing which require detailed visual inspection are the leading edges (including leading edge flaps) with or without de-icing boots, the top and bottom skin and structure at the wing root area and the aileron bellcrank support region — both wings.

59.9.5.1 INSPECTION OF WING LEADING EDGE. (See figure 59-83.) Visually inspect per paragraph 59.9.3 and for hole to hole cracks on the wing leading edge at the joint of the formed ribs and the leading edge skins. Internal access is available thru the access holes in the front spar from wing station 41 to wing station 98. Internal access for leading edge wing station 180.65 to wing station 271 is gained by removing leading edge shroud in attach area and disconnection of flap actuator brackets from front beam. Refer to paragraphs 59.9.4, 59.9.5.3, and Section 52.3.

59.9.5.2 INSPECTION OF WING ROOT TOP AND BOTTOM SURFACES. (See figure 59-84.) Inspection required for the upper and lower wing surface from the front beam to the rear beam, (fus sta 410 to 490) and from the inboard end of the wing panels to the end of the fasteners in the angles attached to the risers (wing sta 41.5 to approx wing sta 75). Visually inspect per paragraph 59.9.3 and for cracks around holes in the horizontal surface of the skin panels and the parts common to the vertical fasteners in the wing panels (i.e. caps, doublers, angles, splice angles). Note: Upper surface panels and upper and lower wing to fuselage fairings are removed prior to inspection. Refer to paragraph 59.9.4 and Section 52.3.)

59.9.5.3 INSPECTION OF LEADING EDGE FLAP ACTUATOR ATTACH REGION. (See figure 59-85.) Visually inspect per paragraph 59.9.3 the areas at wing stations 180.65, 197.13, 213.60 and 230.07 (LE sta 213, 232, 252, and 271) on the front beam of the wing, particularly the flap actuator fittings and their attachment to the wing ribs. Remove leading edge shroud in attach area and, with leading edge flaps full down, disconnect flap actuator brackets from front beam. The leading edge flap can then be swung down on it hinge to facilitate inspection. Clean wing front spar in the flap actuator region of all sealant, dirt, grease, oil, and other foreign matter that might interfer with a reliable visual inspection. (Refer to paragraph 59.9.4 and Section 52.3.) Note: Access to aft side of front spar is from interior of wing. (Upper surface panels are removed for this inspection.)

59.9.5.4 INSPECTION OF AILERON BELL CRANK SUPPORT REGION. (See figure 59-86.) Visually inspect per paragraph 59.9.3 and for hole to hole cracks in the aileron bellcrank support attachment to the wing ribs, the lower surface and spar caps. (Upper surface panels are removed for this inspection.) Refer to paragraph 59.9.4 and Section 52.3.

59.9.6 EMPENNAGE INSPECTION DATA. Areas of the empennage which require detailed visual inspection are the leading edges (with or without de-icing boots), the rudder and elevator counterweight regions, and the vertical stabilizer forward support region.

59.9.6.1 INSPECTION OF VERTICAL STABILIZER LEADING EDGE. (See figure 59-87.) Visually inspect per paragraph 59.9.3 the vertical leading edge at the joint of the formed ribs and the skins for hole to hole cracks and equivalent damage in the radius of the ribs. Note: The leading edge assy is removed from airplane for this inspection. (Refer to paragraph 59.9.4 and Section 52.3.)

59.9.6.2 INSPECTION OF VERTICAL STABILIZER FORWARD SUPPORT REGION. (See figure 59-88.) Visually inspect per paragraph 59.9.3 the forward lower corner of the lower vertical stabilizer structure and for hole to hole cracks in the JE43 vertical stabilizer assembly which are common to the JE16 vertical stabilizer front beam actuator fitting and JE286 scissor attach fitting. Note: The fairing is removed for this inspection. (Refer to paragraph 59.9.4 and Section 52.3.)

59.9.6.3 INSPECTION OF HORIZONTAL STABILIZER LEADING EDGE. (See figure 59-89.) Remove leading edge assy from airplane and then visually inspect per paragraph 59.9.3, the horizontal leading edge at the joint of the formed ribs and the leading edge skins and visually inspect the upper portion of the joint for hole to hole cracks and equivalent damage in the radius of the ribs.

59.9.6.4 INSPECTION OF HORIZONTAL STABILIZER BOX SECTION. (See figure 59-90.) Visually inspect per paragraph 59.9.3 the box beam to upper surface in the horizontal stabilizer box in the area from the front beam to the aft beam at box beam sta 74.88 and 102. (Note: Horizontal stabilizer leading edge is removed for this inspection.) Internal access is gained thru access holes in the front beam and access doors. Refer to paragraph 59.9.4 and Section 52.3.

59.9.6.5 INSPECTION OF ELEVATOR OUTBOARD COUNTERWEIGHT REGION. (See figure 59-91.) Visually inspect per paragraph 59.9.3 the elevator outboard counterweight region at the joint of the formed ribs and channels to the elevator skins, and the area from the outboard tip of the elevator and inboard to the first formed rib, inspect for hole to hole cracks and equivalent damage in the radius of the channels and ribs. Internal access is gained by removal of JE256-2 and JE73-1. Refer to paragraph 59.9.4 and Section 52.3.

59.9.7 FUSELAGE INSPECTION DATA. Areas of the fuselage which require detailed visual inspection are the skin splices and structure at fuselage stations 232 to 277 at waterline 73 on both sides and above the passenger door, fuselage station 410 skin splices below waterline 100, and the skin splice area on both sides from fuselage station 526 to 539.

59.9.7.1 INSPECTION OF SKIN SPLICES AT FUSELAGE STATION 232 TO 277. (See figure 59-92.) Visually inspect per paragraph 59.9.3 the skin splice at waterline 73, fuselage station 232 to 277 on both sides, the skins, straps, doublers, frames, and longeron for cracks. Visually inspect per paragraph 59.9.3 the area above the left hand passenger door for cracks. Remove the headbumper covering the upper door track, headliner and floorboards as required prior to inspection. Refer to paragraph 59.9.4 and Section 52.3. (Ref JP290, JF25 and JF61.)

59.9.7.2 INSPECTION OF FUSELAGE STATION 410 SKIN SPLICE. (See figure 59-93.) Visually inspect per paragraph 59.9.3 the skin-to-frame attachment holes (JF265), the skin-to-filler and filler-to-frame attachment holes, and skin-to-attach angle-to-frame or skin-to-splice plate-to-frame attachment holes. Remove the upper and lower wing fillets, ADF sense antenna, belly fairing access doors, walkway panel, two seat deck panels, and lower vertical trim panels as required. Refer to paragraph 59.9.4 and Section 52.3.

59.9.7.3 INSPECTION OF FUSELAGE STATION 526 TO 539 SKIN SPLICE. (See figure 59-94.) Visually inspect per paragraph 59.9.3 the area between fus sta 526 frame and fus sta 539 frame, from left BL 9 outboard to the center of the booster can tunnel outlet (approx 12 inches) and from that point approx 10 inches or to the bottom of the actuator rod. On the right the equivalent of 22 inches from right BL 9 shall be inspected. Access to the inspection area is gained by removal of the lower fuselage-to-wing fillet on each side of airplane and removal of interior trim as required. Refer to paragraph 59.9.4 and Section 52.3. (Ref JF84 and JF108.)

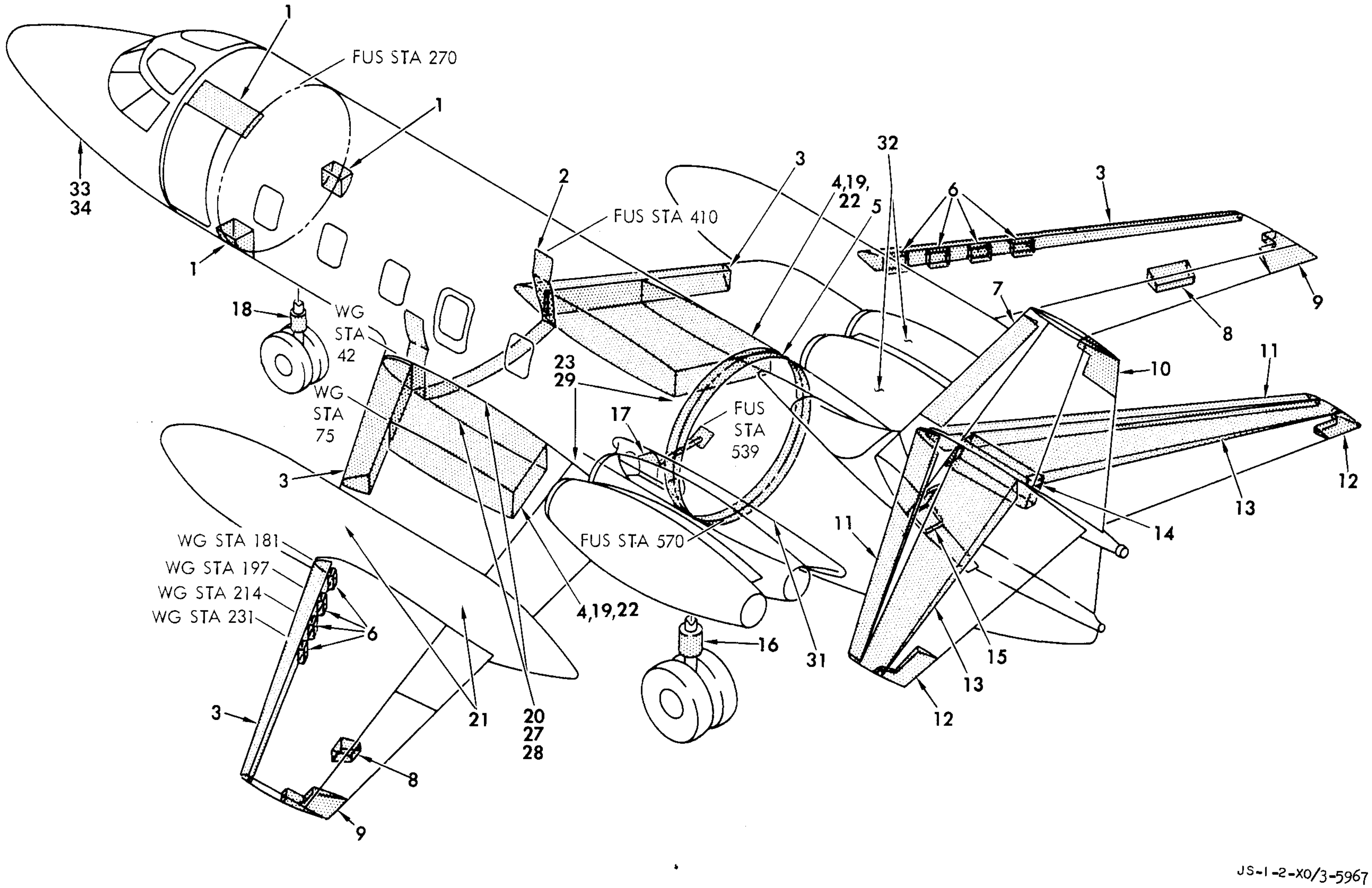


Figure 59-82. 10,000 Hour Inspection Areas (Sheet 1 of 4)

Item Number	Inspection Area	Description	See Paragraph
1	Fuselage station 232 to 277 skin splices.	Inspect the skin splice areas at waterline 73, fuselage station 270.5 on both sides, and above the passenger door on the left side.	59.9.7.1
2	Fuselage station 410 skin splices.	Inspect the skin splice area on both sides of fuselage station 410, below waterline 100.	59.9.7.2
3	Wing leading edge covered by de-icing boots.	Inspect these areas which are covered by de-icing boots, the leading edge skins and ribs which are subject to vibration.	59.9.5.1
4	Wing root area from wing station 41.5 to 75.0	Inspect both the upper and lower surface of each wing, the fittings, doublers, shims, and other components of the wing panel structure.	59.9.5.2
5	Fuselage station 570 aft pressure bulkhead.	Inspect the fuselage shell and the periphery of fuselage station 570 aft pressure bulkhead. Visually inspect bulkhead aft cap and skins also.	58.4.8
6	Leading edge flap actuator region.	Inspect the areas at wing station 180.65, 197.13, 213.60, and 230.07 on the front beam of the wing, particularly the flap actuator fittings and their attachment to the wing ribs. These fittings are inaccessible to visual inspection without wing sealant removal.	59.9.5.3
7	Vertical stabilizer leading edge covered by de-icing boots.	Inspect these areas which are covered by de-icing boots, the leading edge skins and ribs which are subject to vibration.	59.9.6.1
8	Aileron bellcrank support region.	Inspect the attachment areas of the support structure to the ribs and wing rear beam, the support fitting and its attachments to adjacent structure.	59.9.5.4
9	Aileron counterweight region.	Inspect the counterweight region on each aileron. The ribs, fittings, doublers, and trailing edge structure which are subject to cracking are in an area inaccessible to visual inspection.	58.2.10
10	Rudder upper counterweight region.	Inspect the upper counterweight region on the rudder.	58.3.6
11	Horizontal stabilizer leading edge covered by de-icing boots.	Inspect these areas which are covered by de-icing boots. The leading edge skins and ribs which are subject to vibration are inaccessible for visual inspection without removal.	59.9.6.3
12	Elevator outboard counterweight region.	Inspect the outboard counterweight region on each elevator, the ribs, doublers, fittings, and trailing edge structure which are subject to cracking.	59.9.6.5
13	Horizontal stabilizer box section.	Inspect the box section ribs, trailing edge ribs, front beam upper and lower caps, upper and lower stringers, rear beam upper and lower caps, and the trailing edge tee.	59.9.6.4

Figure 59-82. 10,000 Hour Inspection Areas (Sheet 2 of 4)

Item Number	Inspection Area	Description	See Paragraph
14	Horizontal stabilizer center section plates.	Inspect both the upper and lower horizontal stabilizer plates.	58.3.5
15	Vertical stabilizer forward support region.	Inspect the forward corner of the vertical stabilizer in the region of the support fittings, the web and doublers.	59.9.6.2
16	Main landing gear shock struts.	Inspect the piston and other internal assemblies.	58.5.3 58.5.4
17	Fuselage station 526 to 539 skin splices.	Inspect the skin splice area on both sides aft of the main wheel wells at fuselage station 539. Inspect the tunnel outlet on the right side only.	59.9.7.3
18	Nose landing gear shock strut.	Inspect the piston and other internal assemblies.	58.5.3 58.5.4
19	Wing Group	Inspect the MLG actuator and Auxiliary spar outboard support fitting.	58.2.6
20	Wing Group	Inspect the wing-to-fuselage attachment bolts, fittings, main frames, and wing root rib fitting attachment.	58.2.2 58.2.16
21	Wing Group	Inspect the external fuel tank to wing attachment bolts and fittings.	58.2.8
22	Wing Group	Inspect the MLG aft support spar and bearing support cap.	58.2.5
23	Wing Group	Inspect the MLG aft support spar-to-fuselage attachment connecting linkage and attachment bolts.	58.2.4
24	Deleted		
25	Deleted		
26	Deleted		
27	Fuselage Group	Inspect the fuselage main frames and side frames.	58.4.4
28	Fuselage Group	Inspect the fuselage main frame machined cutouts.	58.4.3
29	Fuselage Group	Inspect the longeron in the wheel well area as required per paragraph 5.4.11.g.	58.4.2
30	Deleted		

Figure 59-82. 10,000 Hour Inspection Areas (Sheet 3 of 4)

Item Number	Inspection Area	Description	See Paragraph
31	Nacelle Group	Inspect the engine nacelle attachment fittings and bolts.	58.6.2
32	Nacelle Group	Inspect the engine mount fittings and bolts.	58.6.3
33	Noted	Perform all intermediate inspections as required.	Noted
34	Noted	Inspect and lubricate aircraft per HOMI Section 12.12.	Noted

Figure 59-82. 10,000 Hour Inspection Areas (Sheet 4 of 4)

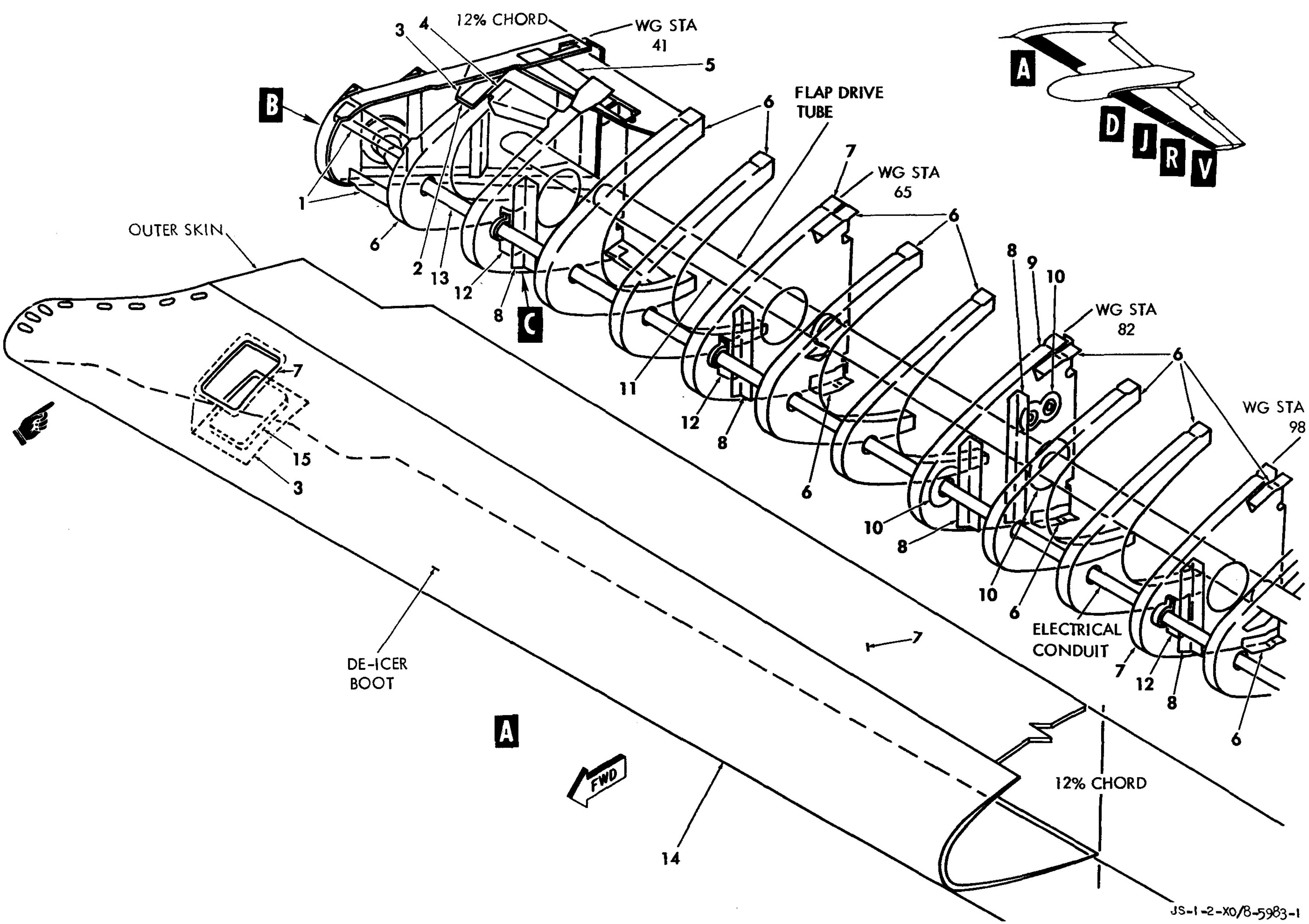


Figure 59-83. Inspection of Wing Leading Edges (Sheet 1 of 13)

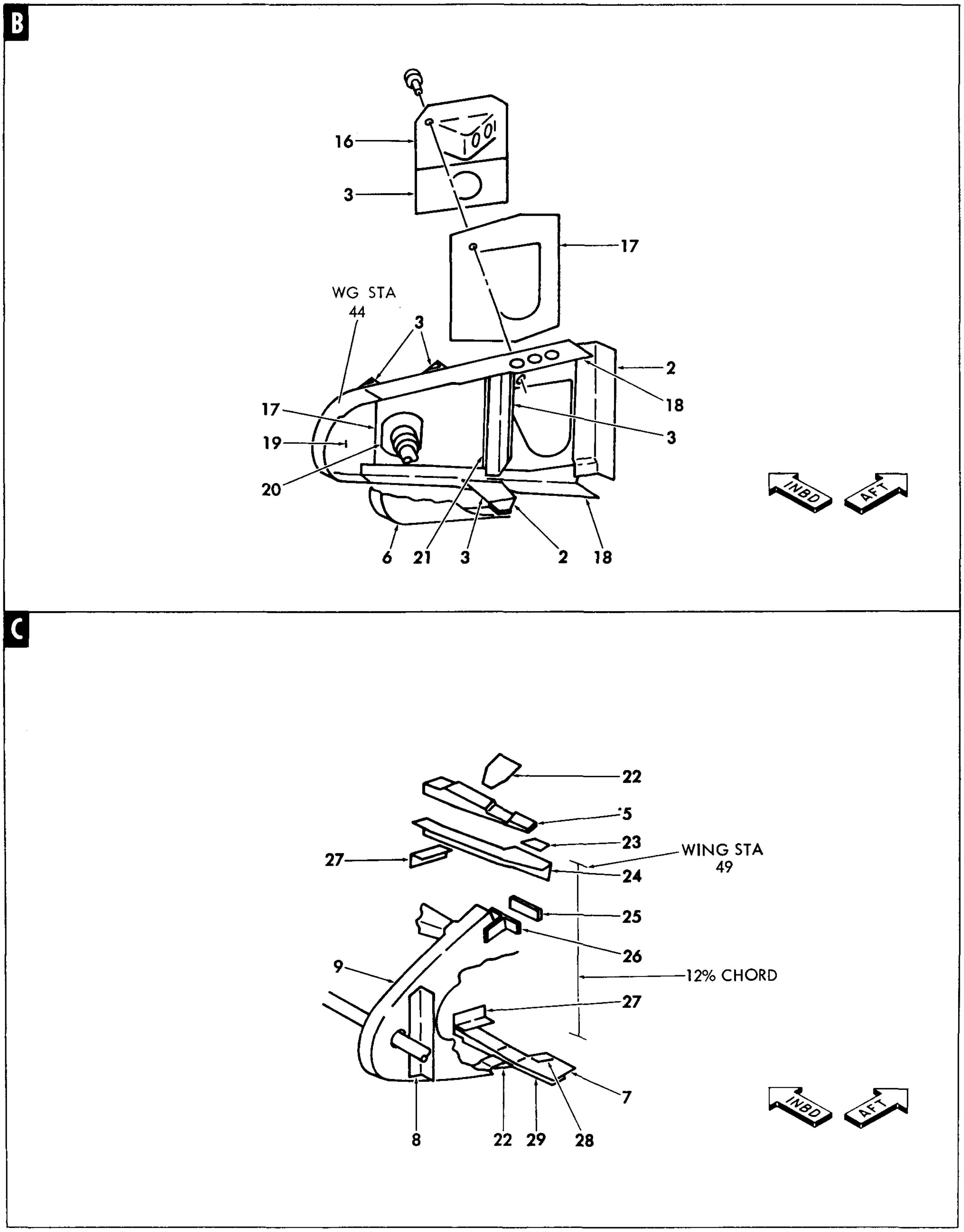
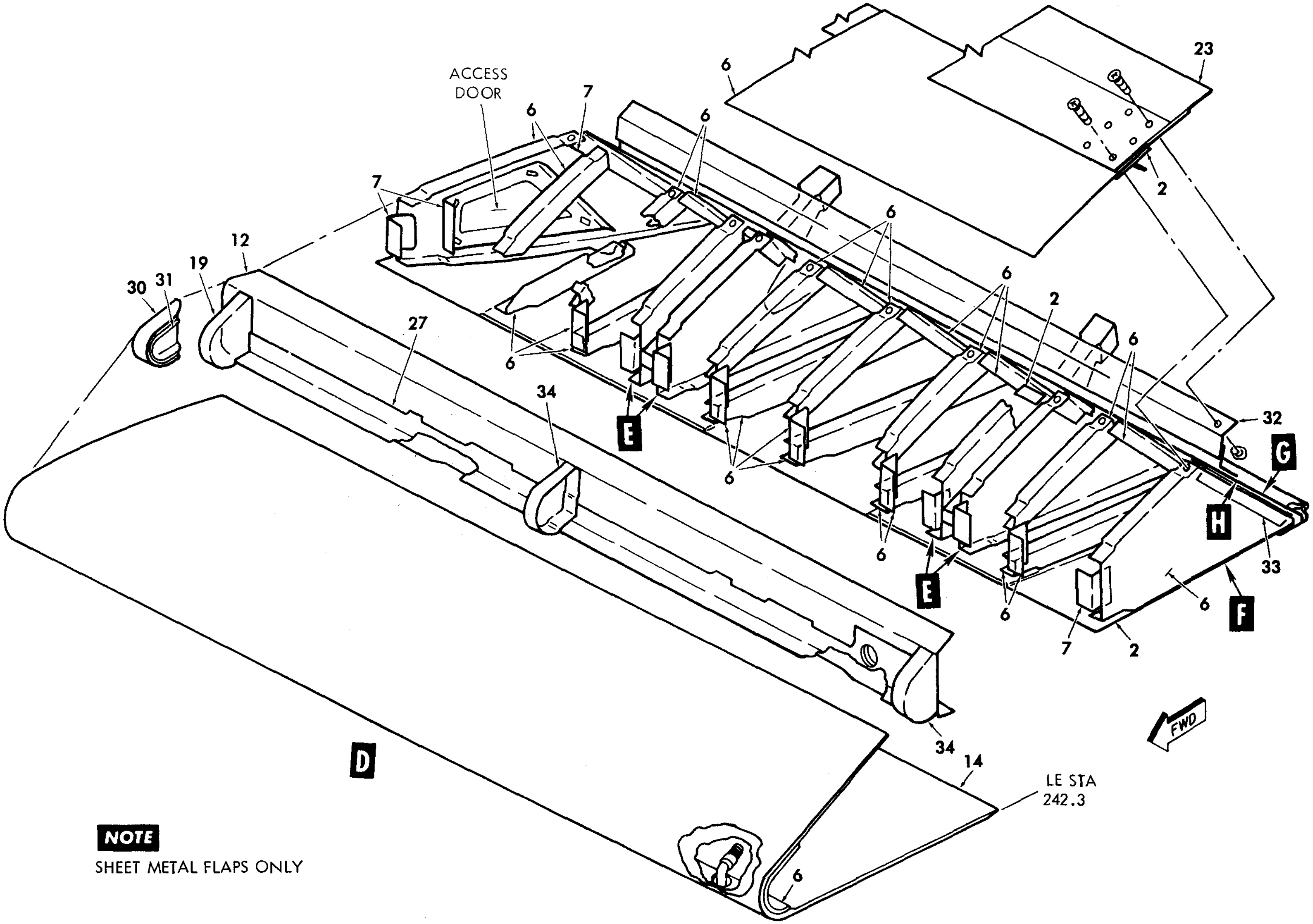


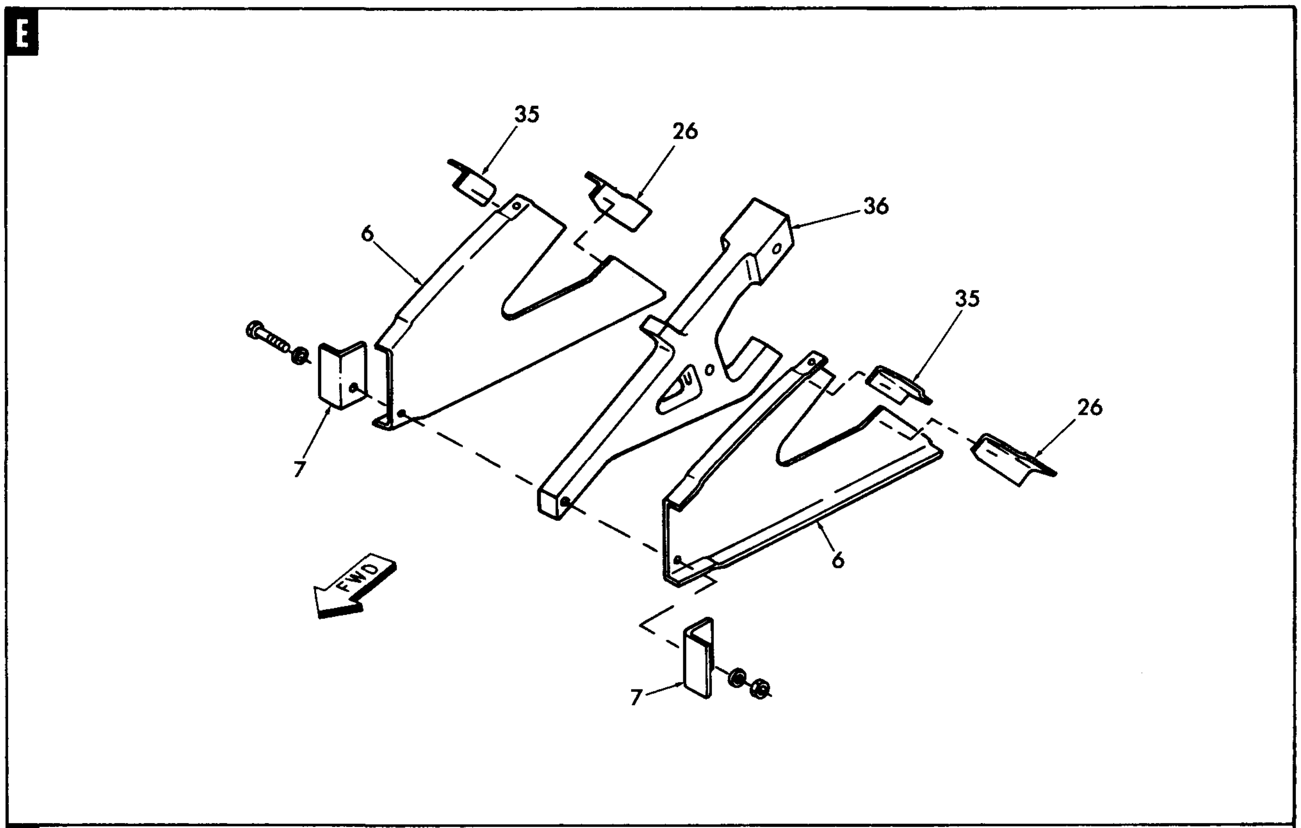
Figure 59-83. Inspection of Wing Leading Edges (Sheet 2 of 13)



NOTE

SHEET METAL FLAPS ONLY

Figure 59-83. Inspection of Wing Leading Edges (Sheet 3 of 13)



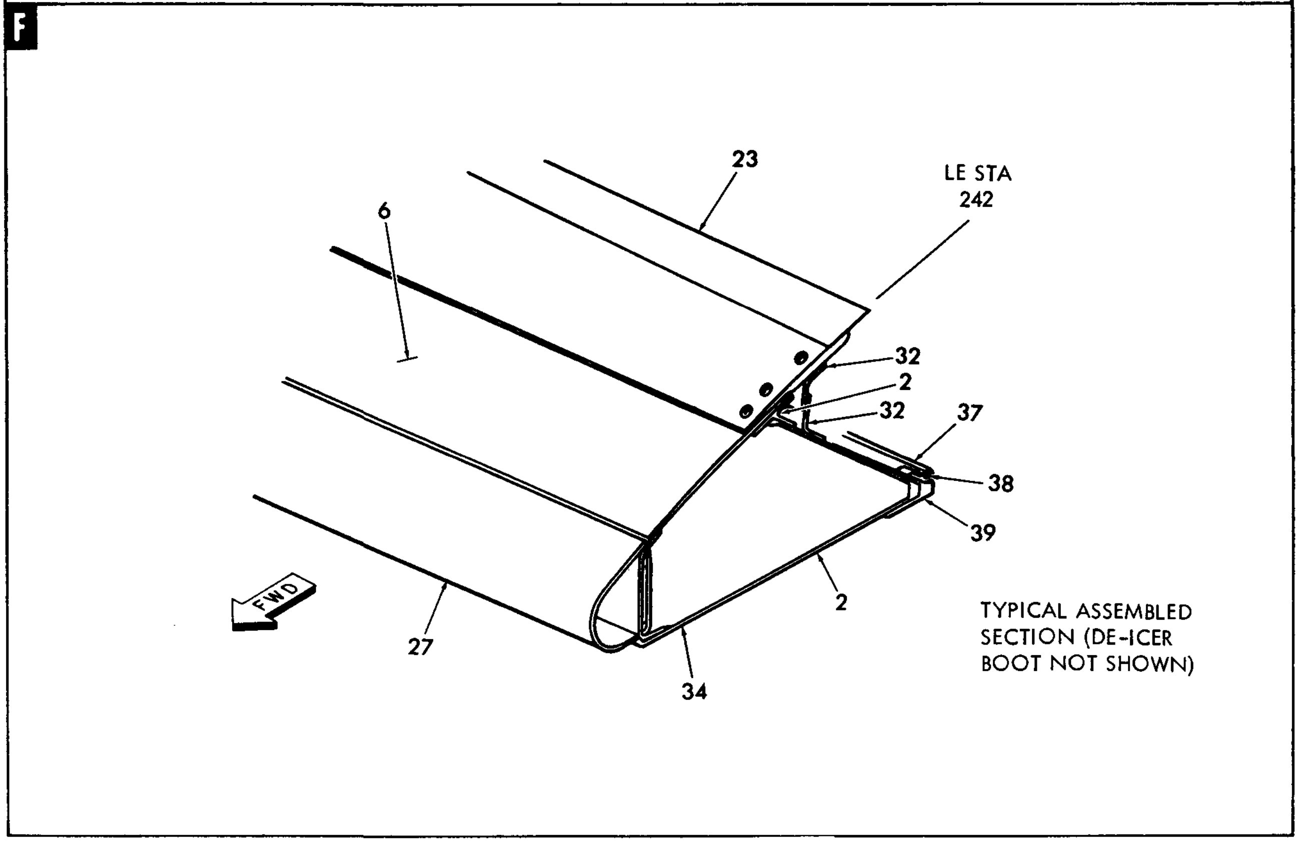


Figure 59-83. Inspection of Wing Leading Edges (Sheet 4 of 13)

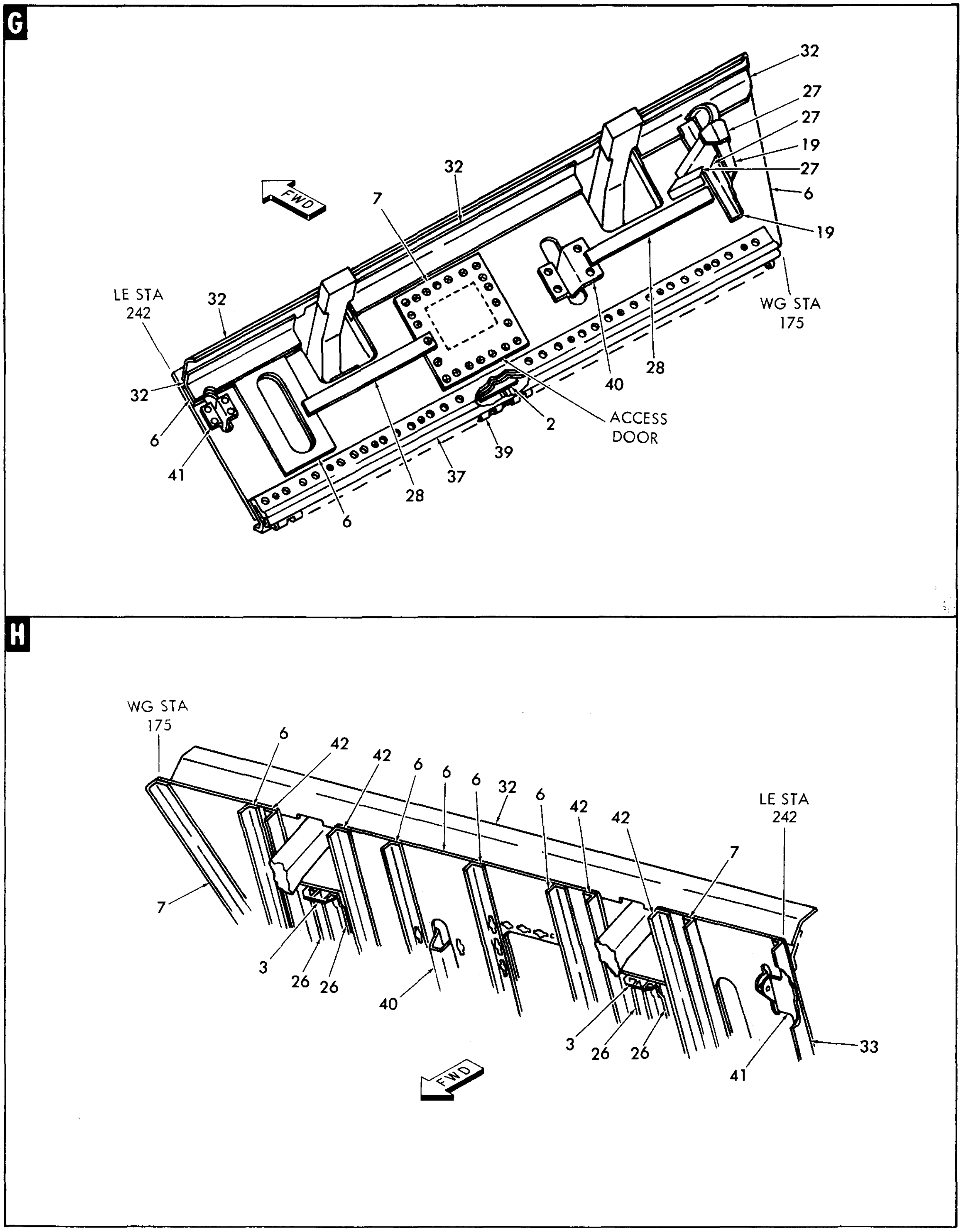
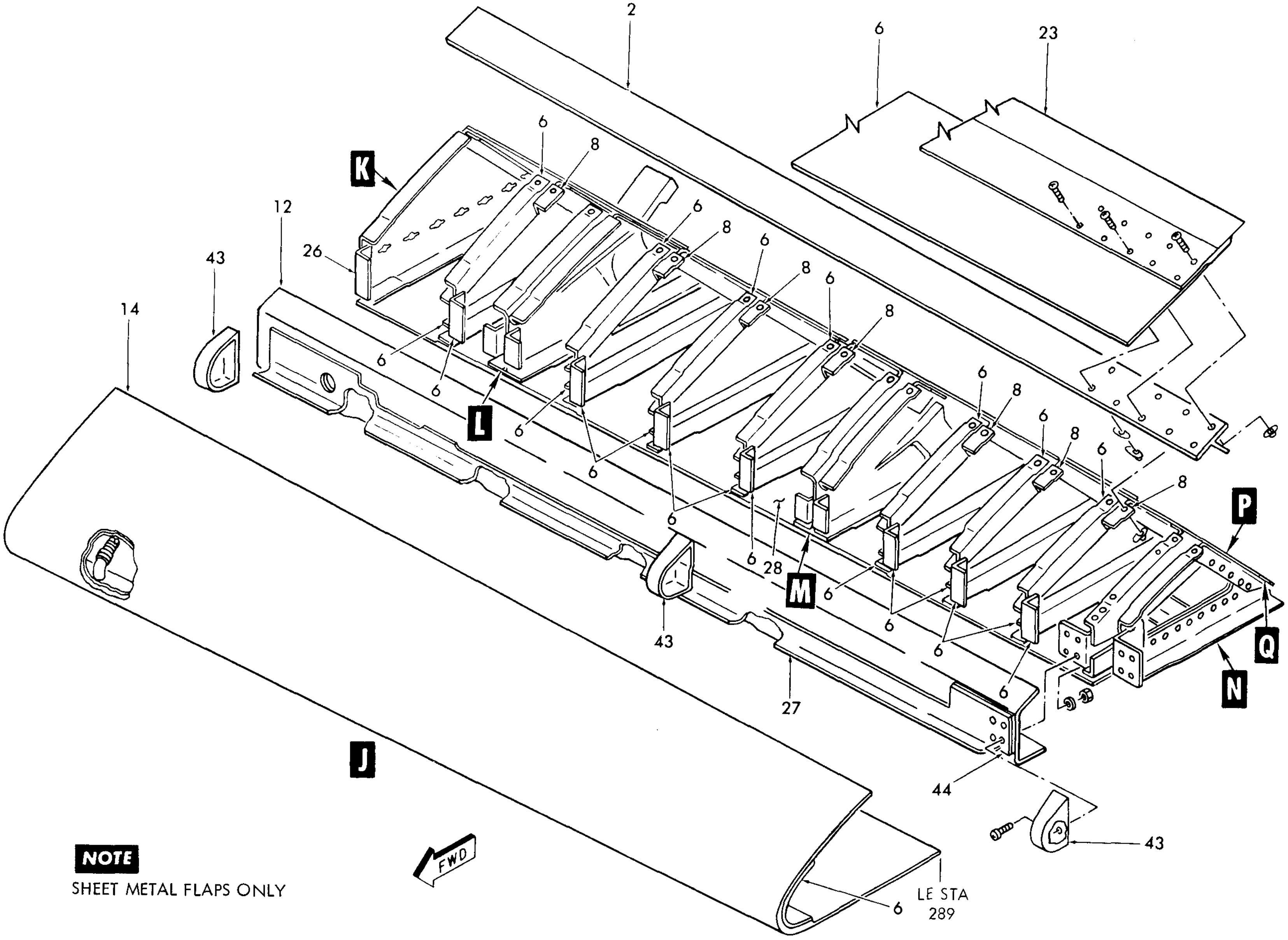


Figure 59-83. Inspection of Wing Leading Edges (Sheet 5 of 13)



NOTE
SHEET METAL FLAPS ONLY

Figure 59-83. Inspection of Wing Leading Edges (Sheet 6 of 13)

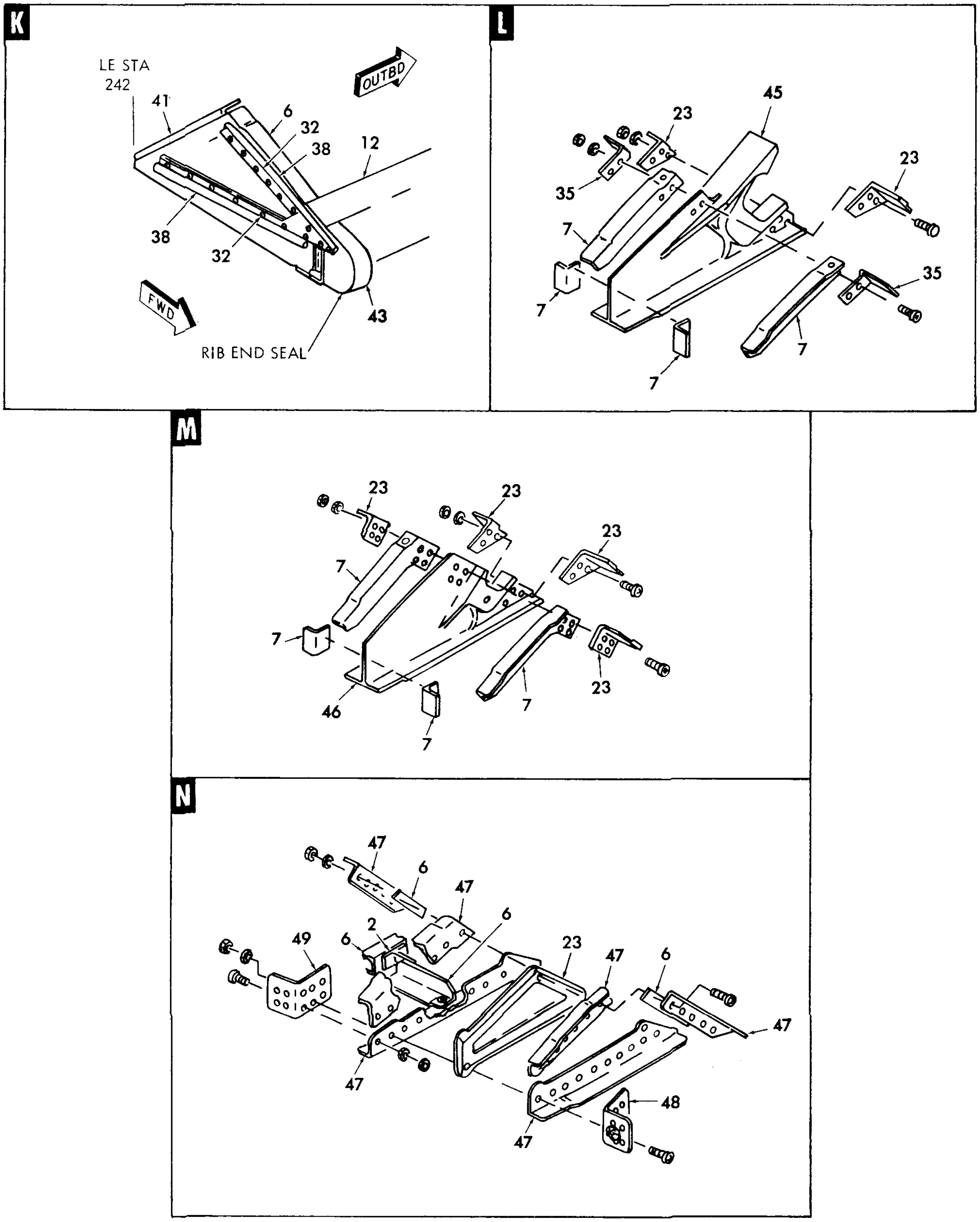


Figure 59-83. Inspection of Wing Leading Edges (Sheet 7 of 13)

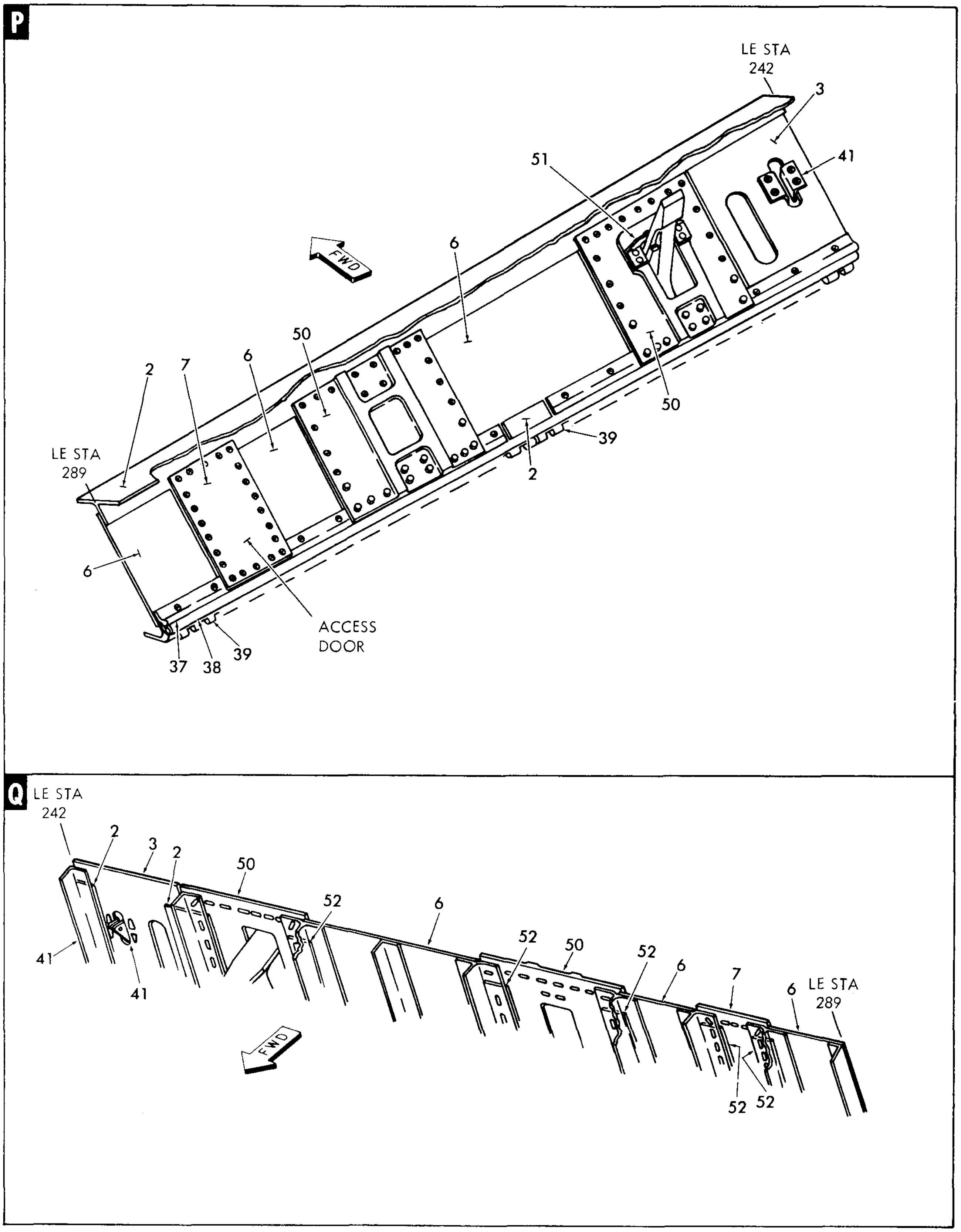
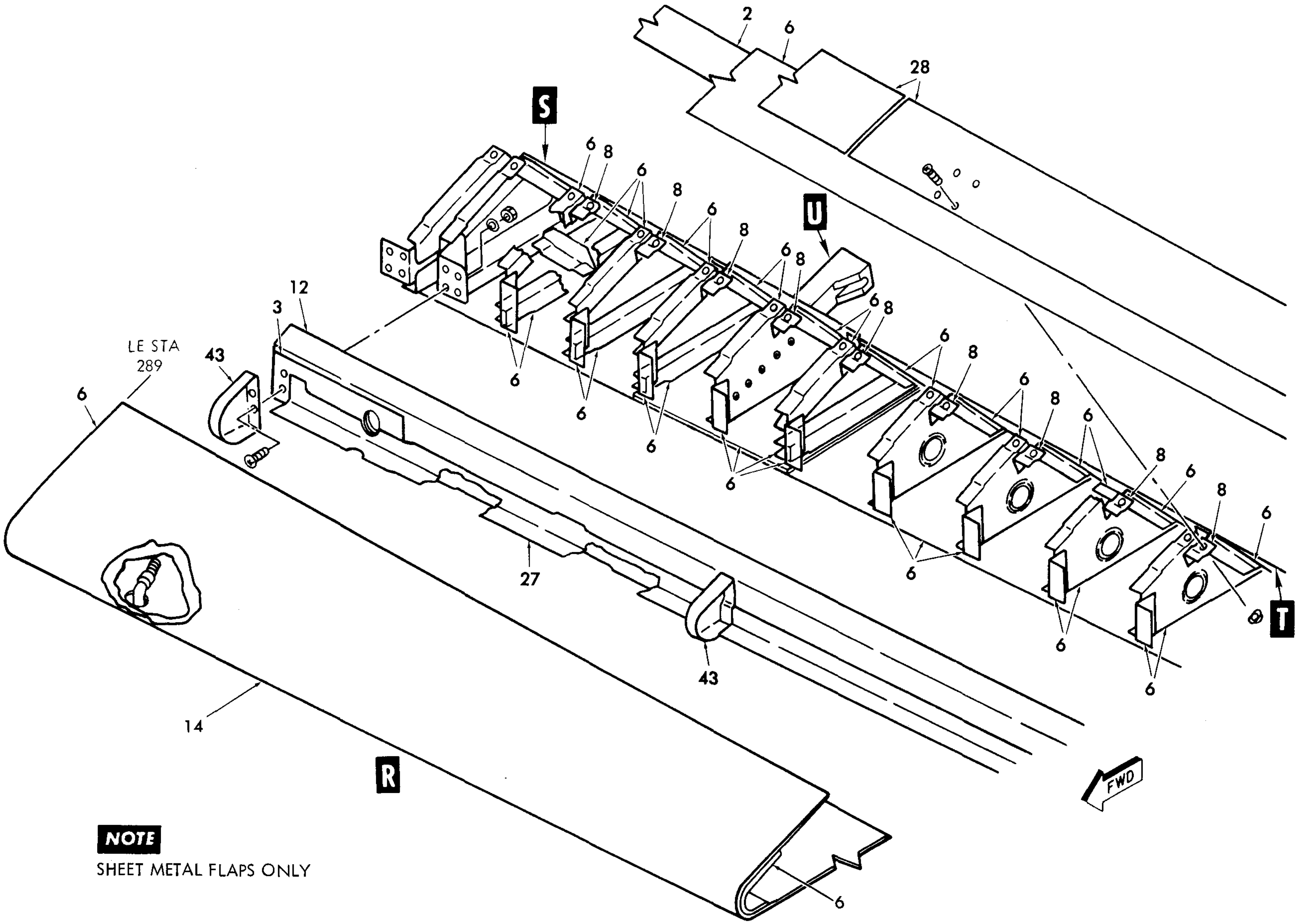


Figure 59-83. Inspection of Wing Leading Edges (Sheet 8 of 13)



NOTE
SHEET METAL FLAPS ONLY

Figure 59-83. Inspection of Wing Leading Edges (Sheet 9 of 13)

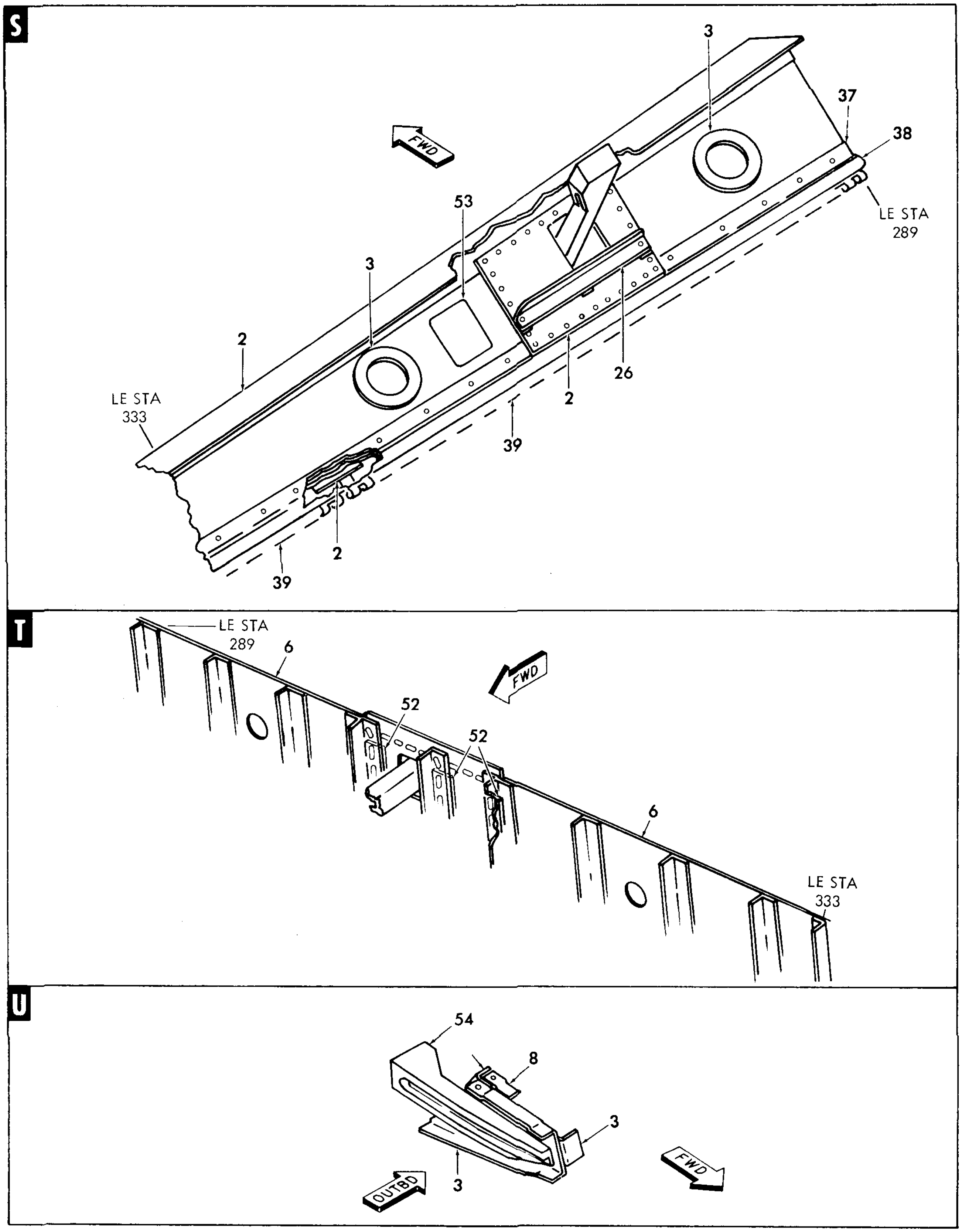
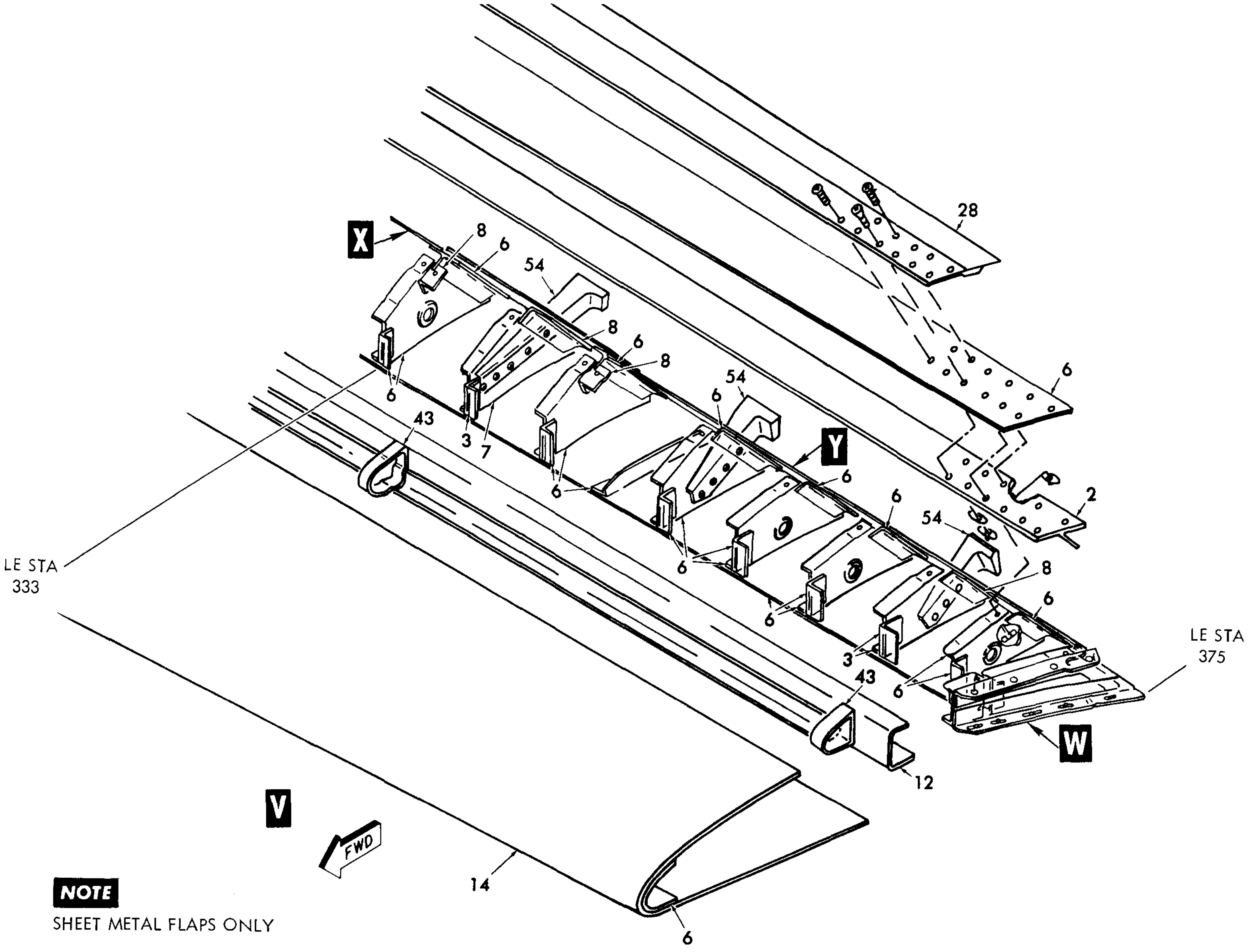


Figure 59-83. Inspection of Wing Leading Edges (Sheet 10 of 13)



NOTE
SHEET METAL FLAPS ONLY

Figure 59-83. Inspection of Wing Leading Edges (Sheet 11 of 13)

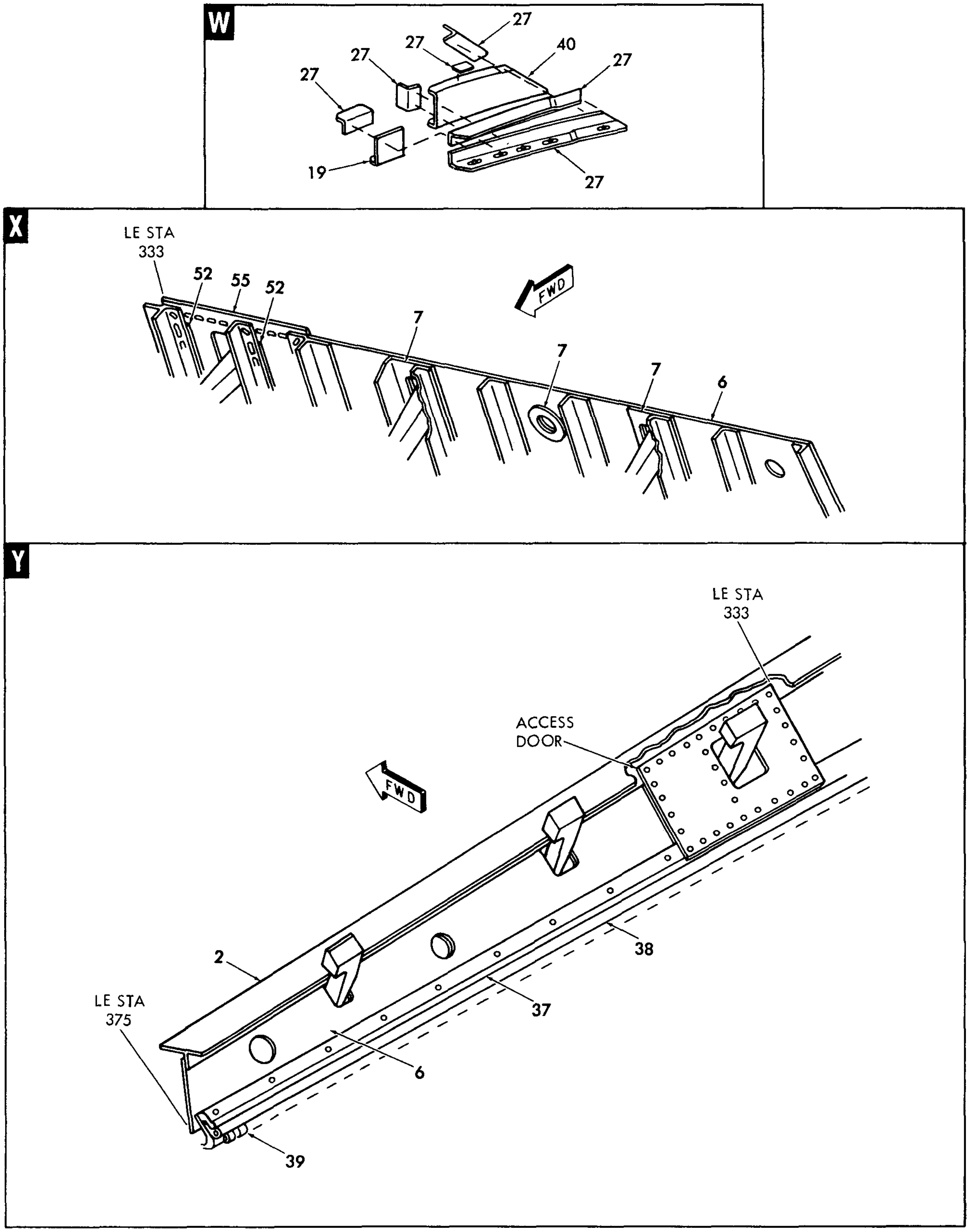


Figure 59-83. Inspection of Wing Leading Edges (Sheet 12 of 13)

ITEM	THICKNESS	MATERIAL
1	0.051	7075-T6
2	0.080	7075-T6
3	0.063	7075-T6
4	0.063	7075-T6
5	0.43 MAX (TAPERS)	2024-T4
6	0.040	7075-T6
7	0.050	7075-T6
8	0.062	7075-T6
9	0.050	7075-T6
10	0.125	PHENOLIC
11	0.049 WALL X 2.00 CD	2024-T3
12	0.040	2024-T3
13	0.083 WALL X 1.0 OD	6061
14	0.125	FABRIC AND RUBBER
15	0.100	7075-T6
16	0.20	A356-T6 AL ALLOY
17	0.050	7075-T6
18	0.125	2024-T4
19	0.050	2024-T4
20	——	AL ALLOY COUPLING
21	0.125	7075-T6
22	0.053	7075-T6
23	0.160	7075-T6
24	0.090	7075-T6
25	0.19	2024-T3
26	0.070	7075-T6
27	0.040	2024-T4
28	0.125	7075-T6

ITEM	THICKNESS	MATERIAL
29	0.750 MAX (TAPERS)	2024-T4
30	0.060	FIBERGLASS LAMINATE
31	0.032	FIBERGLASS LAMINATE
32	0.071	7075-T6
33	0.063	7075-T6
34	0.040	2024-T351
35	0.094	7075-T6
36	1.00	7075-T651
37	0.100	2024-T4
38	0.050	SILICONE RUBBER
39	0.320	2024-T4
40	0.063	2024-T4
41	0.062	7075-T6
42	0.075	7075-T6
43	0.040	2024-T351
44	0.032	7075-T6
45	1.50 MAX (TAPERS)	7075-T651
46	0.320 MAX (TAPERS)	7075-T651
47	0.040	301 1/4 H
48	0.050	301 1/4 H
49	0.063	301 1/4 H
50	0.250	7075-T651
51	0.064	7075-T6
52	0.080	2024-T3
53	——	AN7510-2
54	0.500 MAX (TAPERS)	7075-T6
55	0.063	301 1/2 H

Figure 59-83. Inspection of Wing Leading Edges (Sheet 13 of 13)

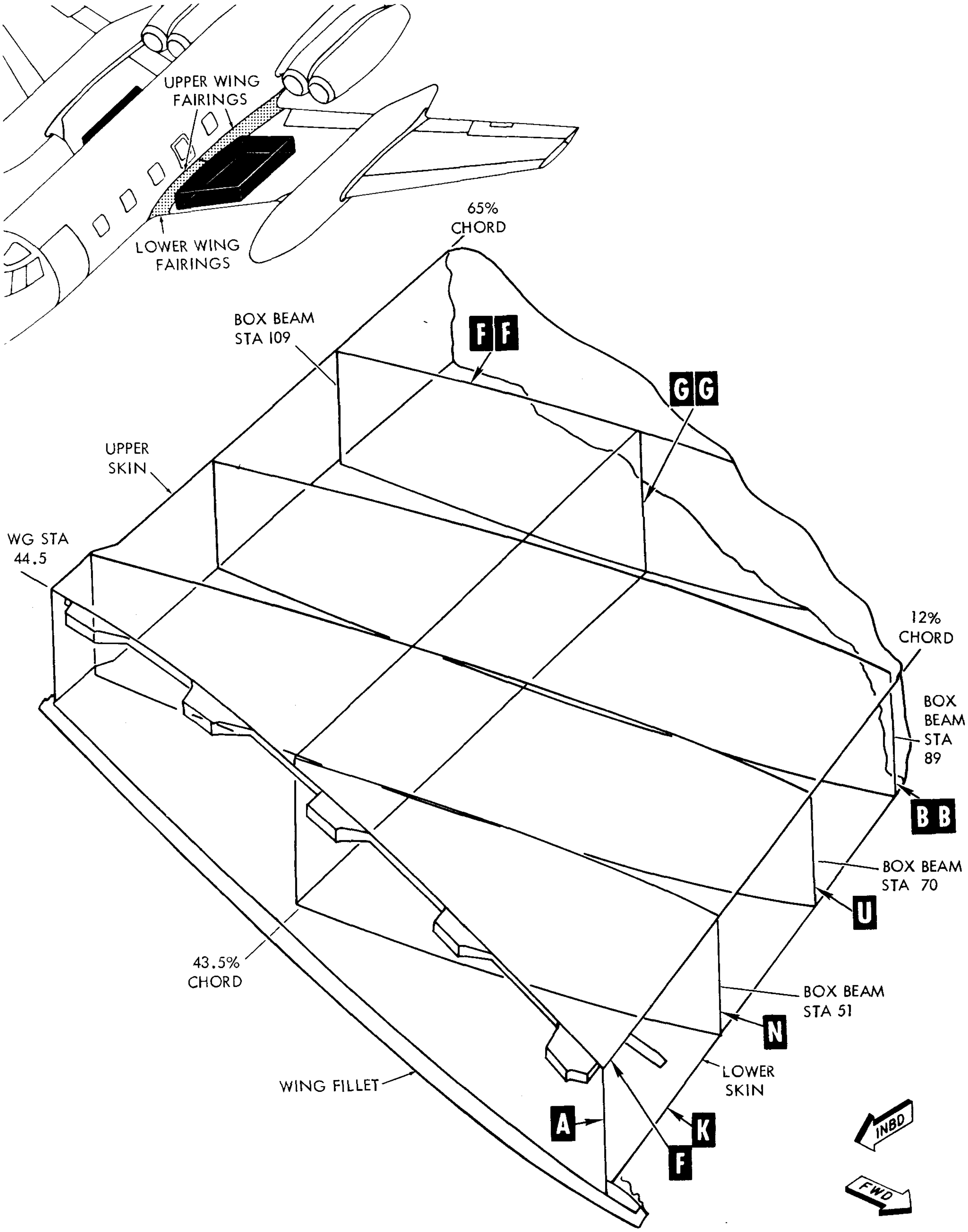


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 1 of 11)

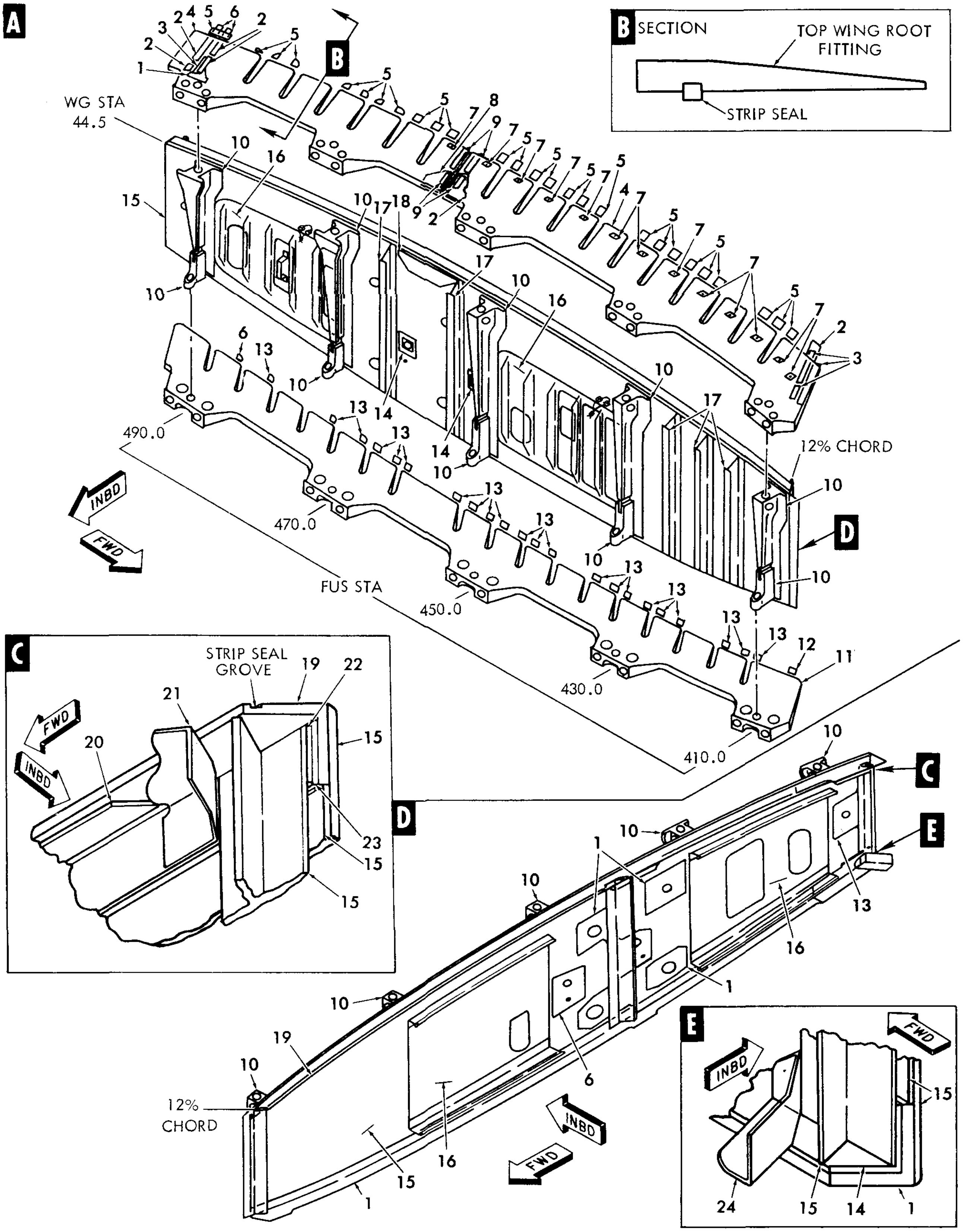


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 2 of 11)

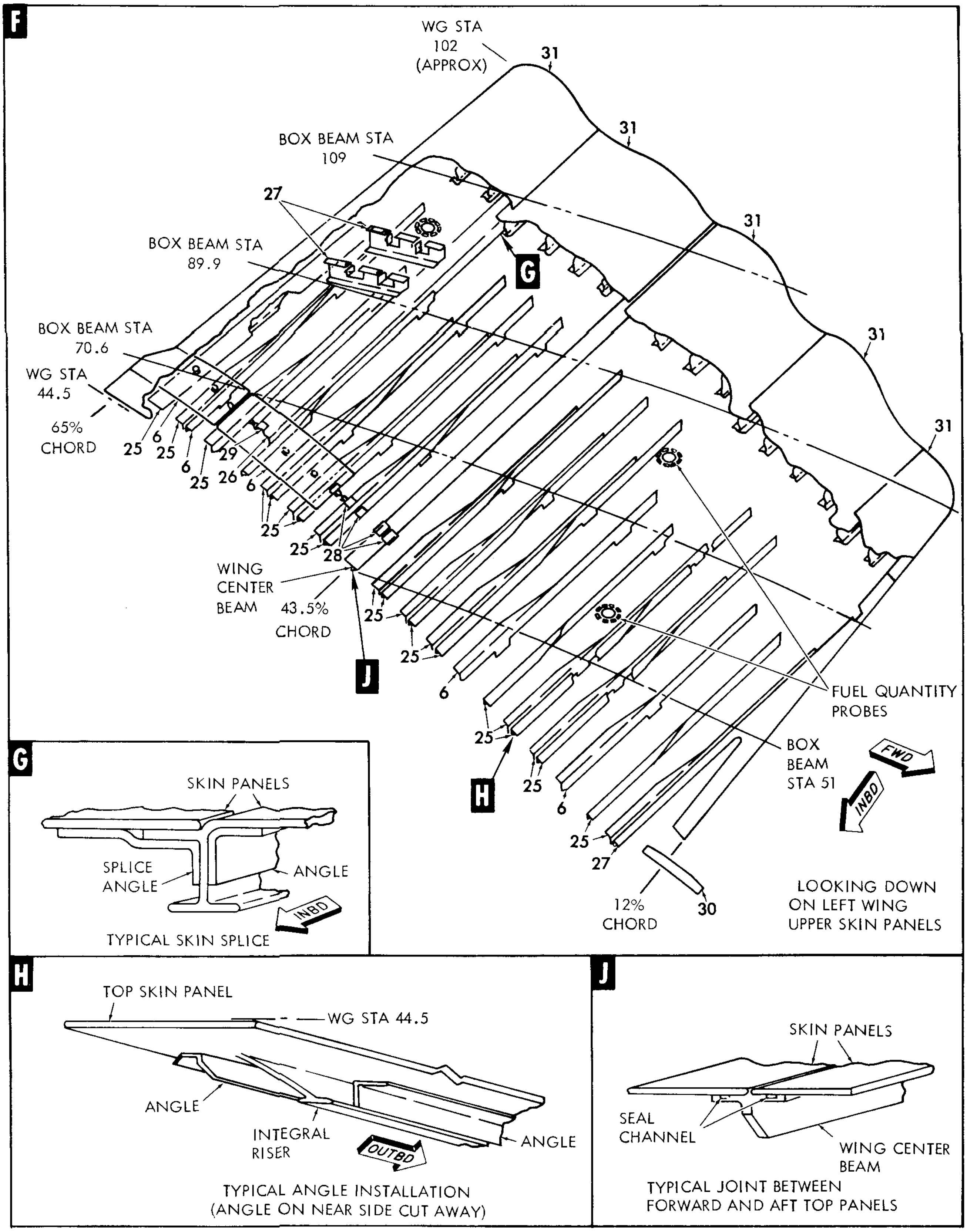


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 3 of 11)

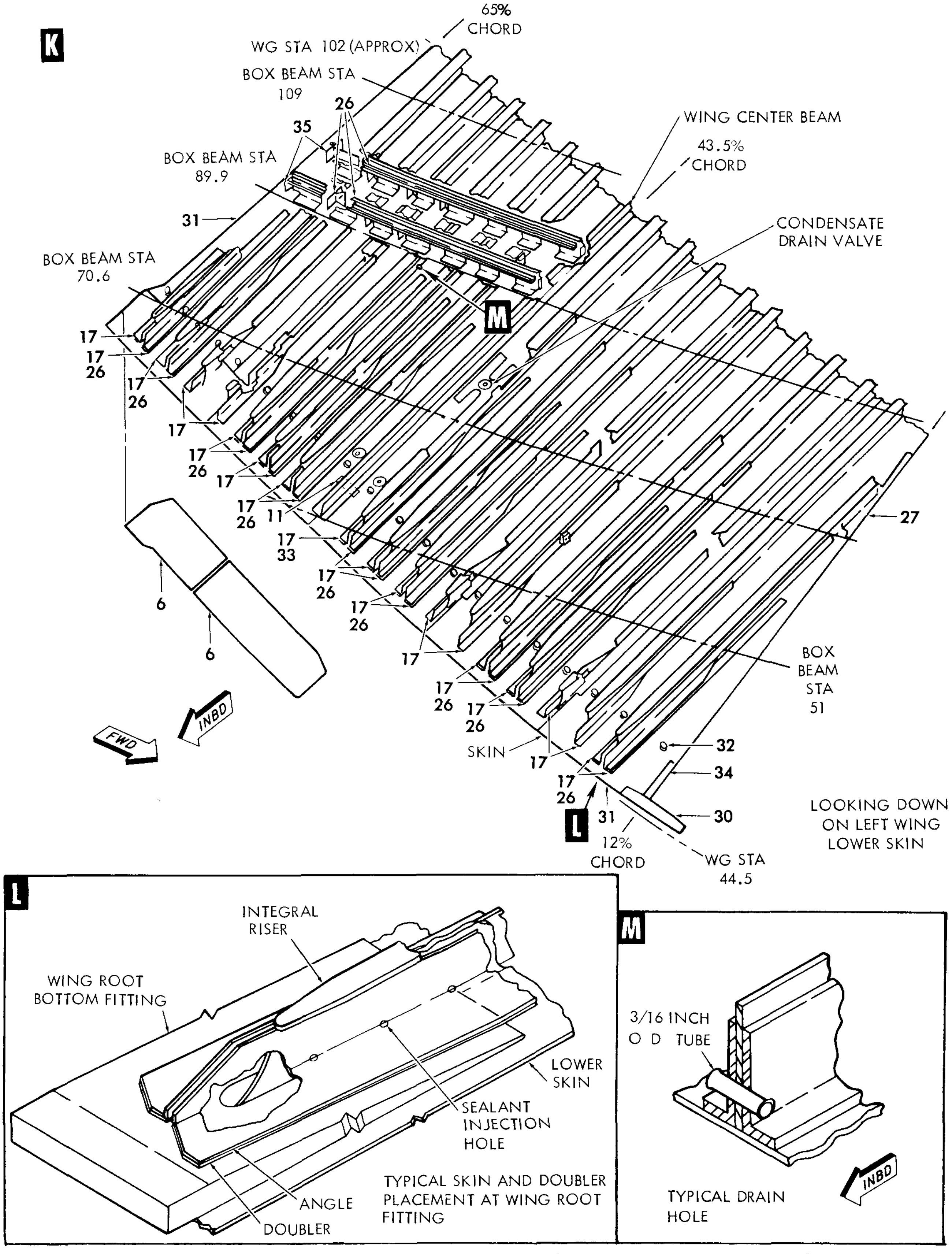


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 4 of 11)

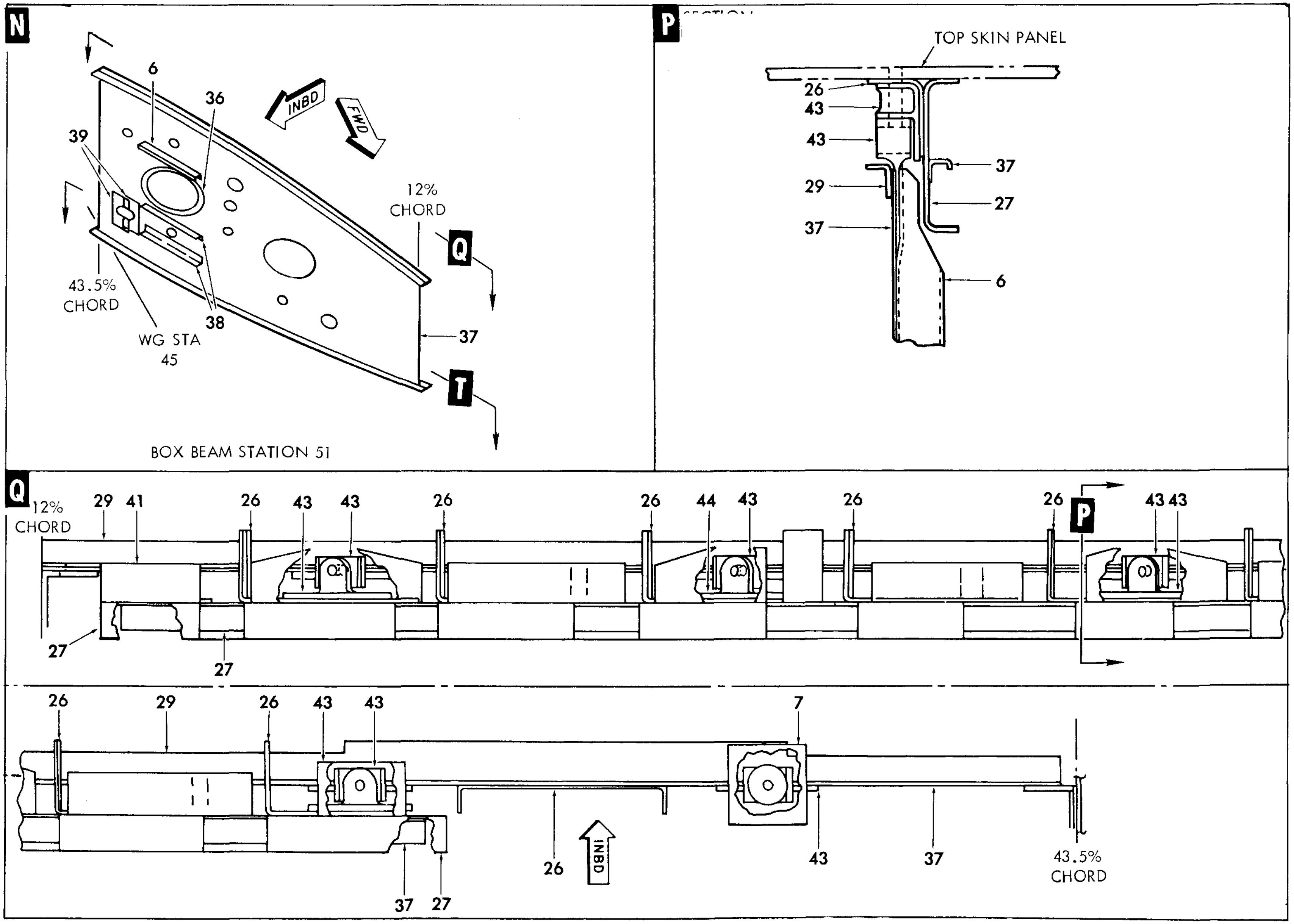


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 5 of 11)

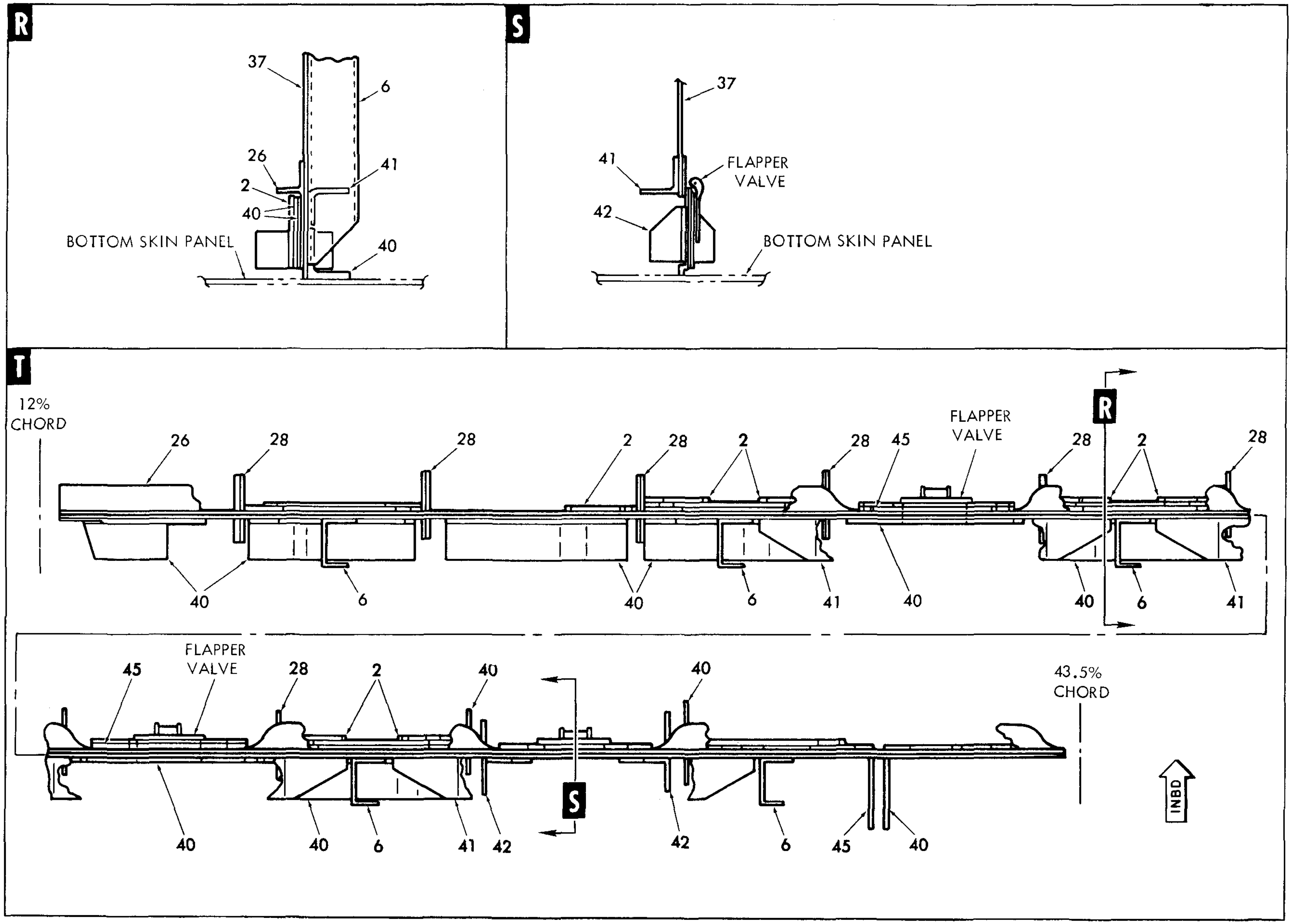


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 6 of 11)

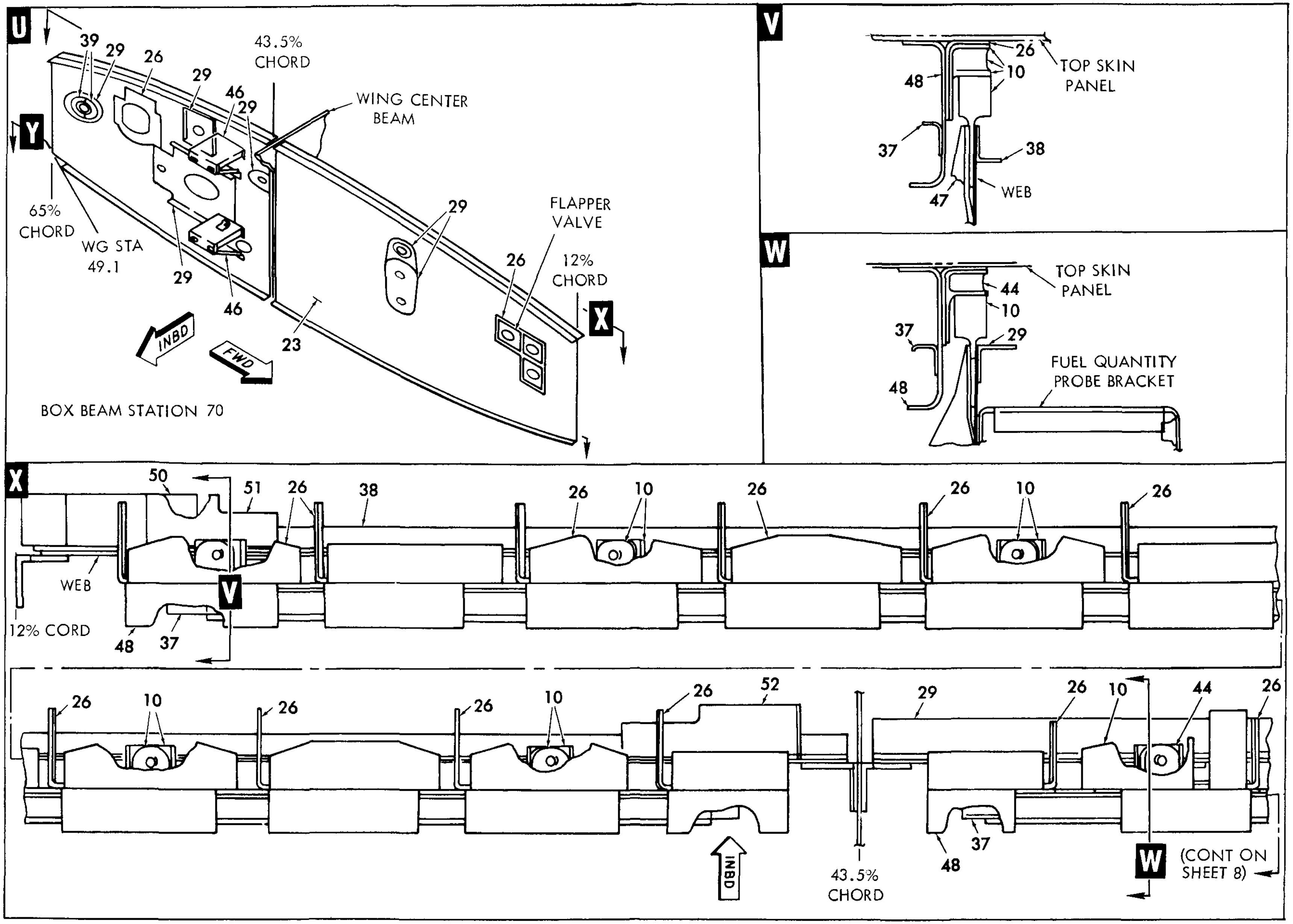


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 7 of 11)

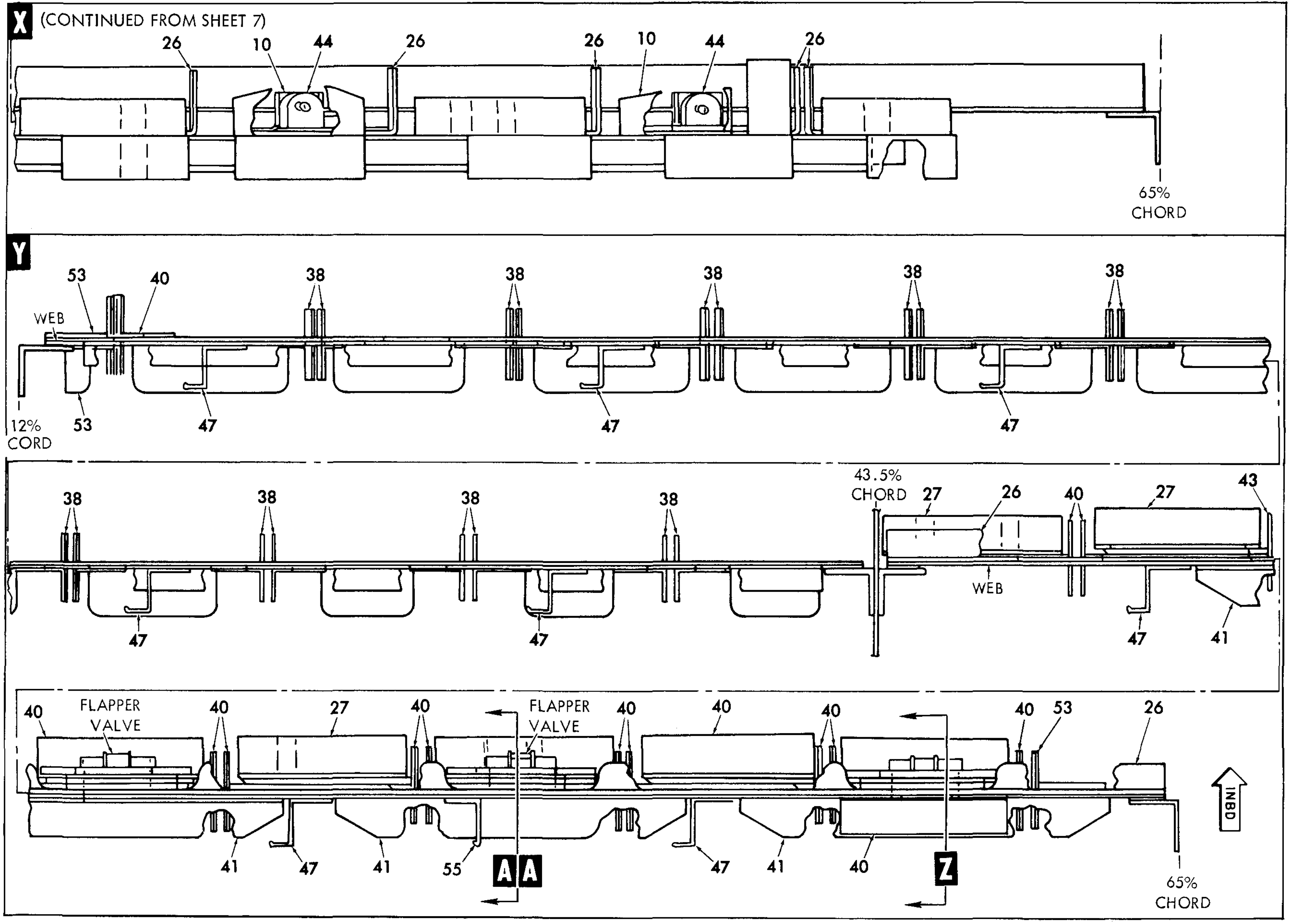


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 8 of 11)

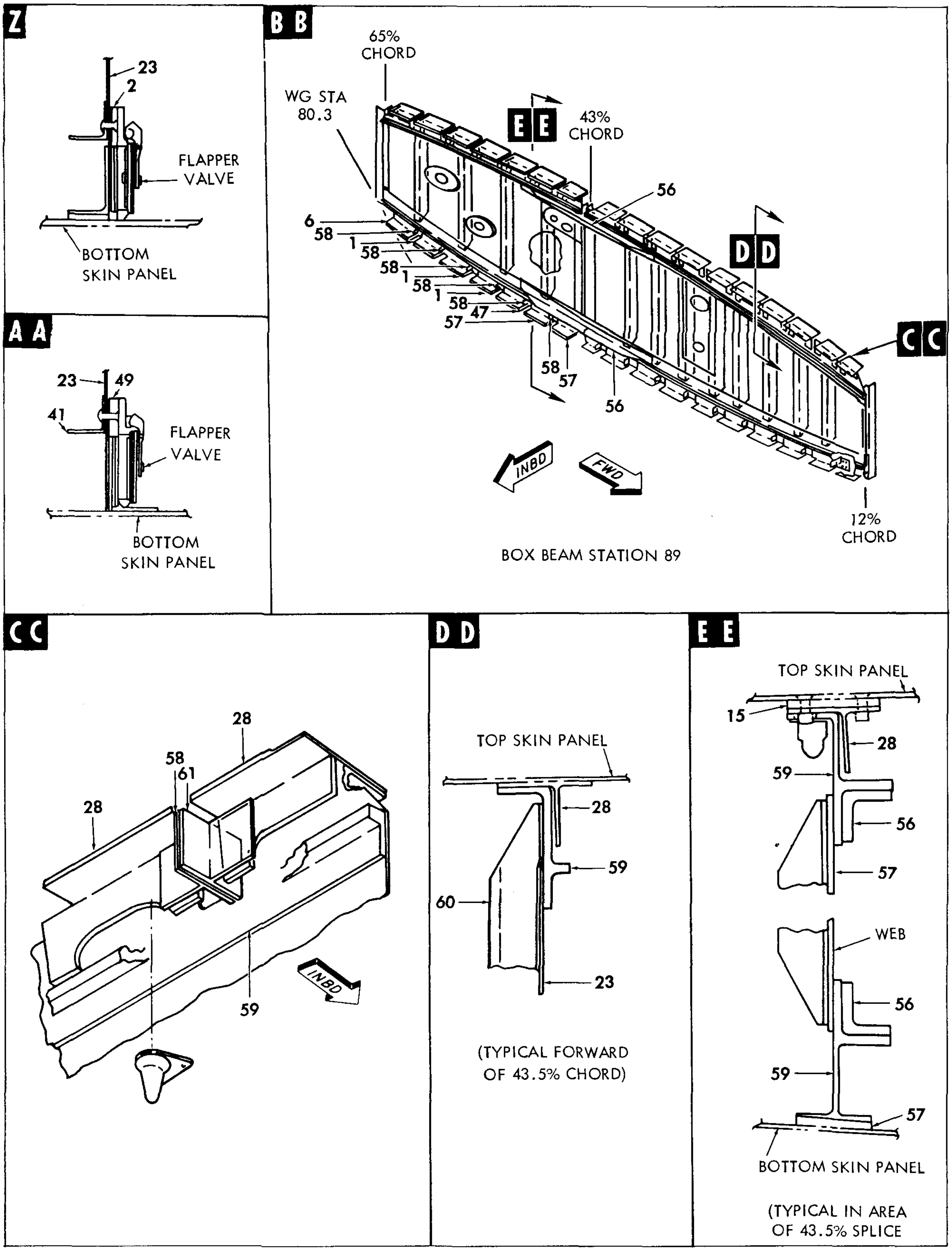
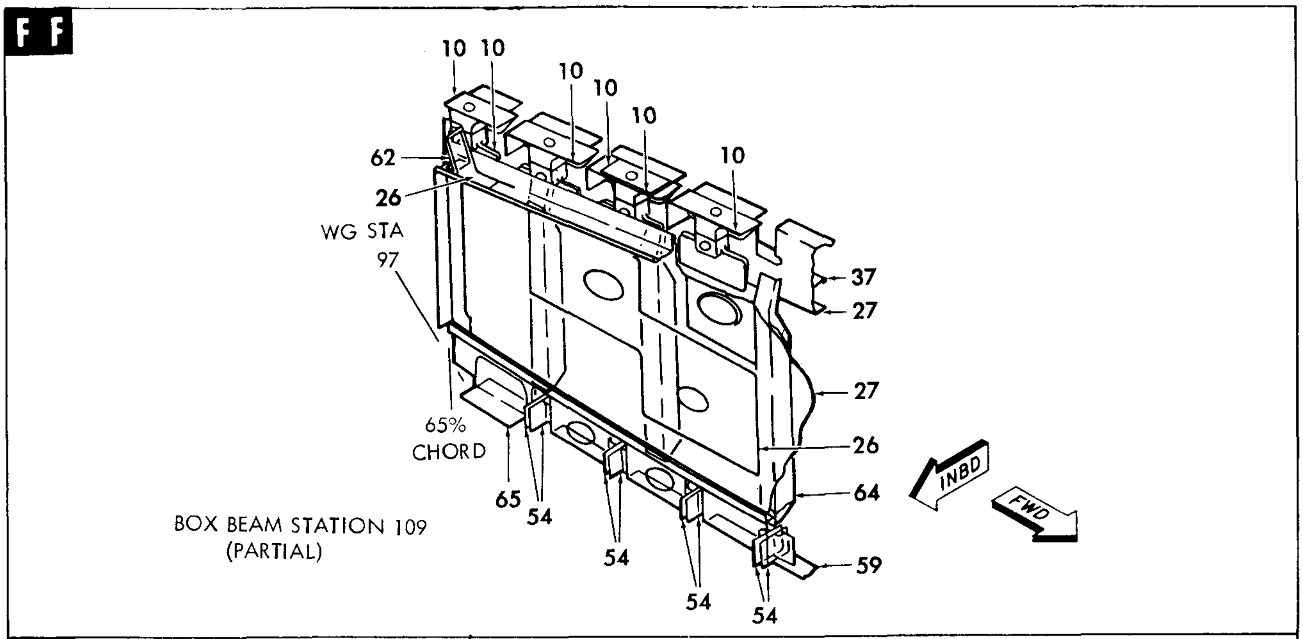


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 9 of 11)



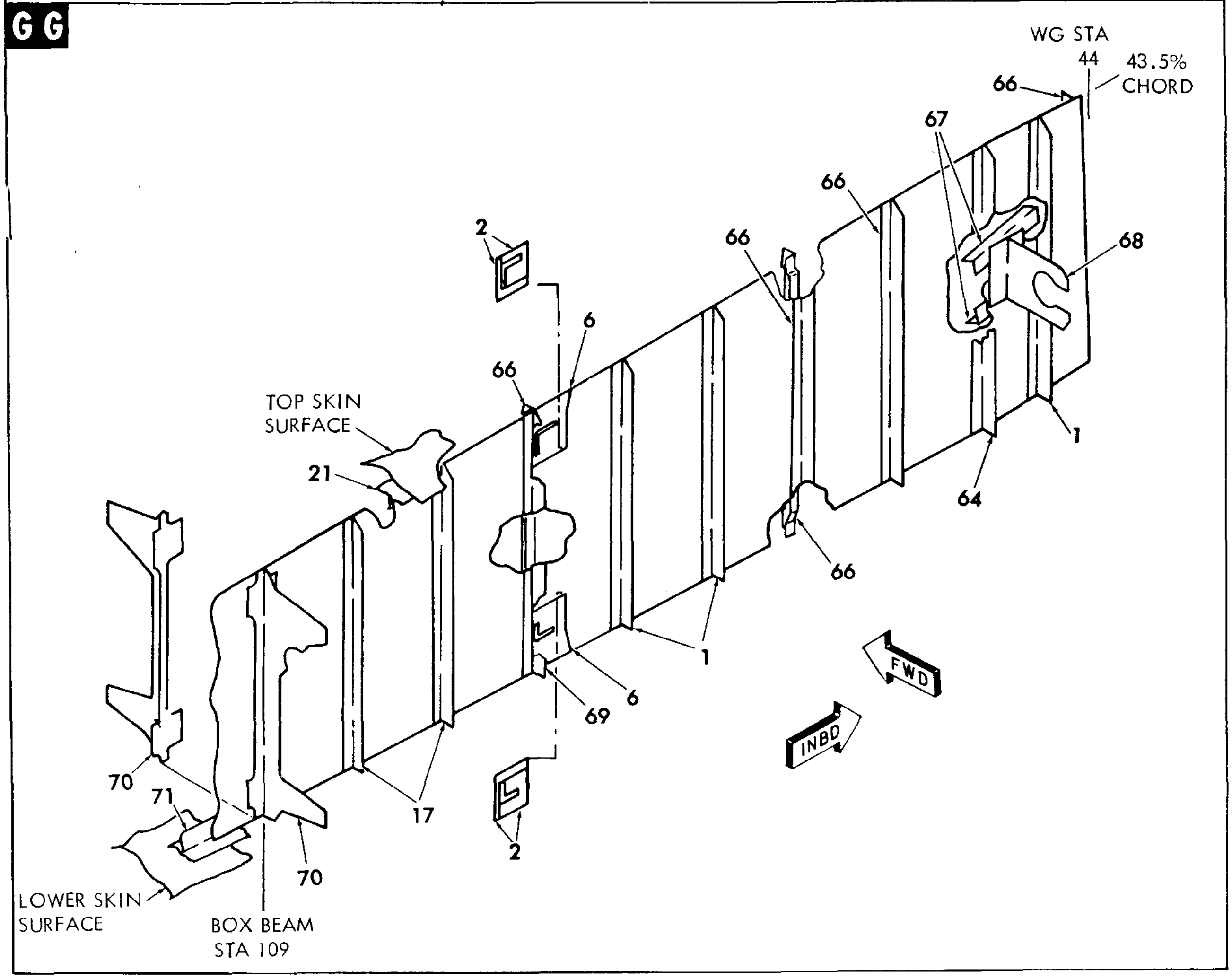
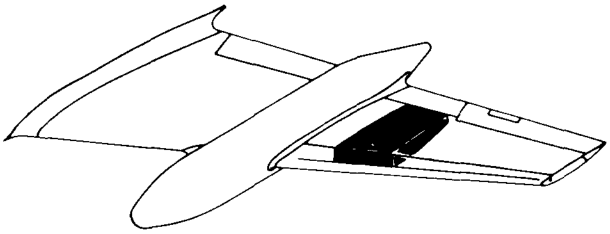
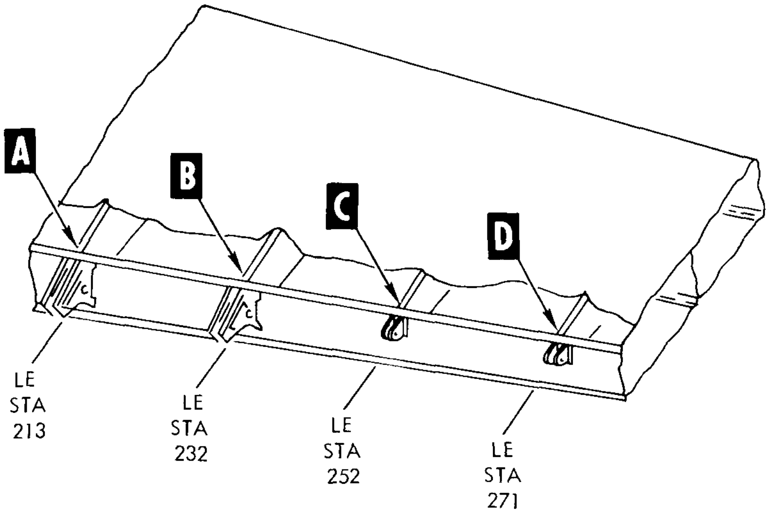


Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 10 of 11)

ITEM	THICKNESS	MATERIAL
1	0.125	7075-T6
2	0.125	2024-T3
3	0.100	2024-T3
4	1.40	7075-T6
5	0.101	2024-T4
6	0.080	7075-T6
7	VARIES	2024-T4
8	0.125	2024-T4
9	0.100	2024-T4
10	VARIES	7075-T6
11	1.50	7075-T6
12	0.160	2024-T3
13	0.100	2024-T4
14	0.063	2024-T3
15	0.190	7075-T6
16	—	DOOR ASSEMBLY
17	0.070	7075-T6
18	0.312	7075-T6
19	0.156	7075-T6
20	0.071	2024-T3
21	0.040	3003-0
22	0.032	2024-T3
23	0.032	2024-T6
24	0.050	3003-0
25	0.120	7075-T6
26	0.050	7075-T6
27	0.071	7075-T6
28	0.100	7075-T6
29	0.040	7075-T6
30	TAPERS	4130 STL
31	0.080 MAX	7075-T6
32	VARIES	2024-T6
33	0.063	7075-T6
34	0.020	7075-T6
35	0.375	7075-T6
36	0.040	2024-T4
37	0.025	7075-T6
38	0.062	7075-T6
39	0.125	PHENOLIC
40	0.060	7075-T6
41	0.051	7075-T6
42	0.062	7075-T6
43	VARIES	A360 AL ALLOY
44	0.830	7075-T6
45	0.080	2024-T3
46	0.040	2024-T3
47	0.062	2024-T4
48	0.072	7075-T6
49	0.190	2024-T3
50	VARIES	2024-T351
51	0.050	2024-T3
52	0.063	2024-T4
53	0.094	7075-T6
54	0.062	2024-T4
55	0.050	2024-T4
56	0.250	2024-T4
57	0.160	7075-T6
58	0.060	2024-T4
59	0.200	7075-T6
60	0.063	2024-T4
61	0.060	2024-T4
62	0.100	6061-74
63	0.125	LAMINATED AL ALLOY
64	0.075	7075-T6
65	0.125	7075-T6
66	0.070	7075-T6
67	0.078	2024-T4
68	0.125	7075-T6
69	0.094	2024-T4
70	0.188	7075-T6
71	0.078	7075-T6

Figure 59-84. Inspection of Wing Root Upper and Lower Surfaces (Sheet 11 of 11)





ITEM	THICKNESS	MATERIAL
1	——	7075-T6
2	0.090	7075-T6
3	0.125 (Min.), 0.200 (Max.)	7075-T6
4	0.060 (Min.), 0.080 (Max.)	7075-T6
5	0.070 (Min.), 0.100 (Max.)	7075-T6
6	0.125	7075-T6
7	0.188	7075-T6
8	0.125 (Min.), 0.195 (Max.)	7075-T6
9	0.040	7075-T6
10 △1	0.040	7075-T6
10 △2	0.060	7075-T6
11	0.025	7075-T6

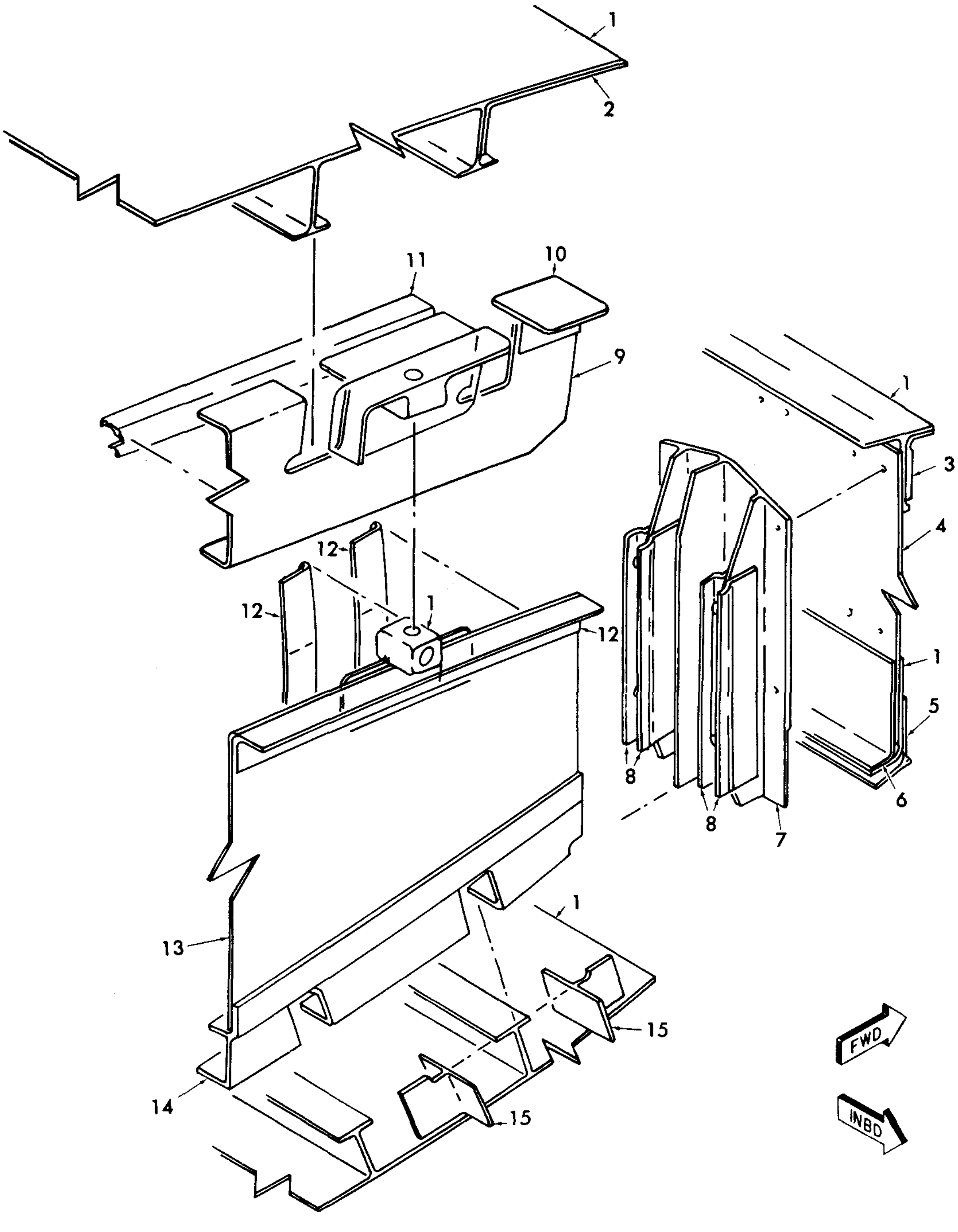
ITEM	THICKNESS	MATERIAL
12	0.050	7075-T6
13	0.032	7075-T6
14	0.040 (Min.), 0.200 (Max.)	7075-T6
15	0.062	2024-T4
16	0.060	7075-T6
17	0.071	7075-T6
18	—	2024-T351
19	—	6061-T6
20	0.125	2024-T4
21	0.062	7075-T6
22	0.125	2024-T3
23	—	8630 STEEL

NOTE

△1 AIRPLANES 5001 THRU 5022.

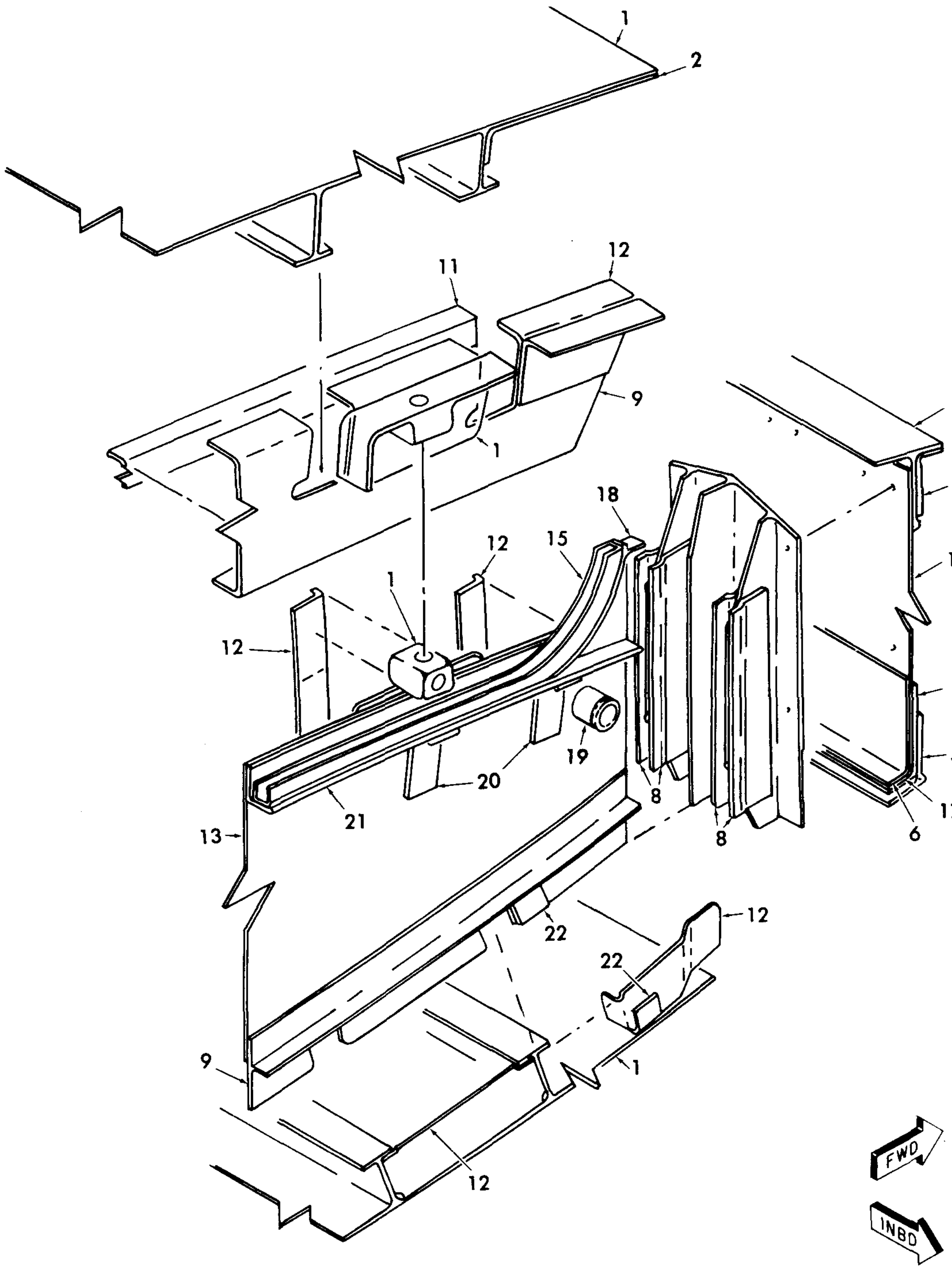
△2 AIRPLANES 5023 AND UP.

Figure 59-85. Inspection of Leading Edge Flap Actuator Region (Sheet 1 of 5)



A

Figure 59-85. Inspection of Leading Edge Flap Actuator Region (Sheet 2 of 5)



B

Figure 59-85. Inspection of Leading Edge Flap Actuator Region (Sheet 3 of 5)

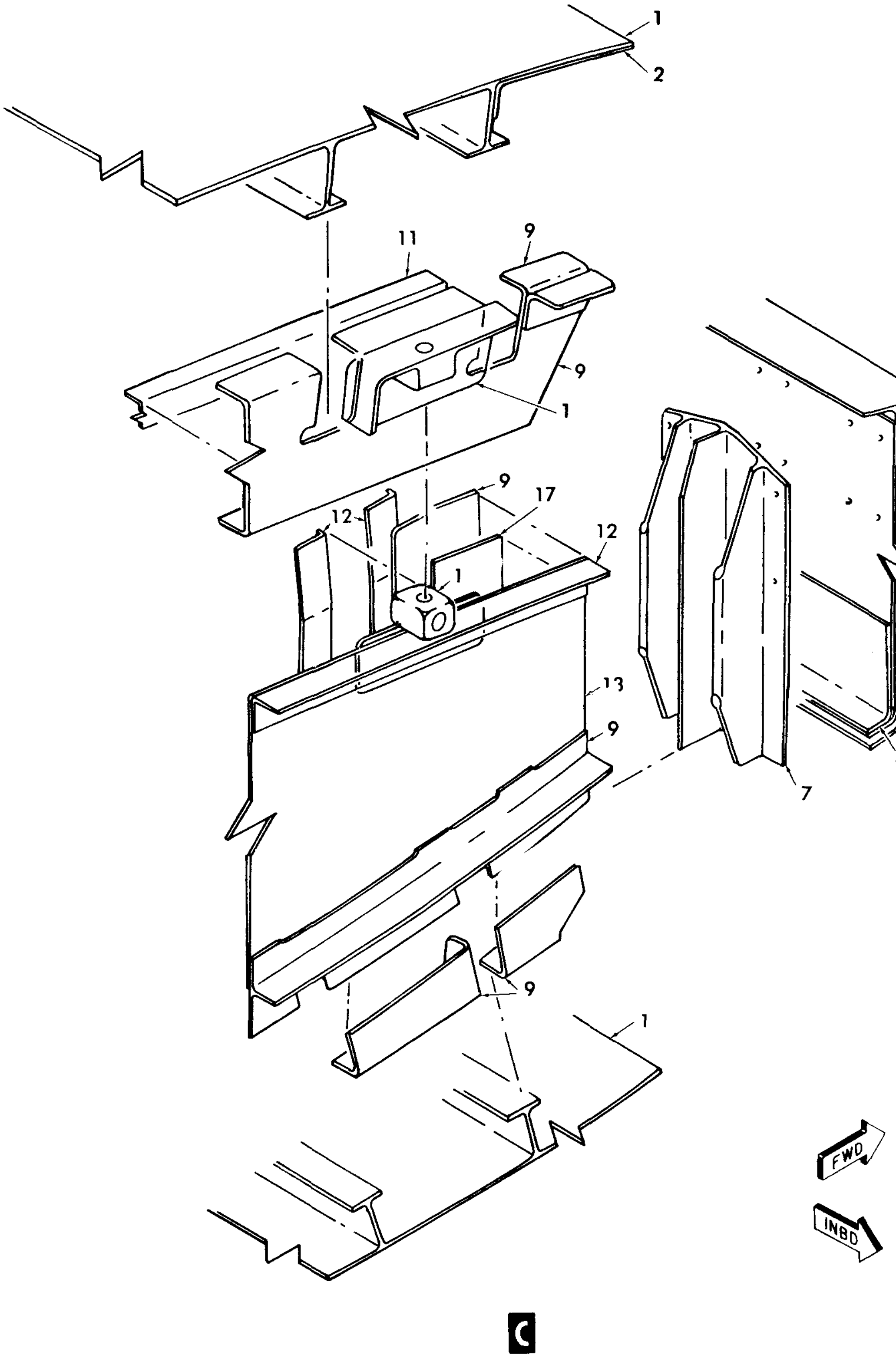


Figure 59-85. Inspection of Leading Edge Flap Actuator Region (Sheet 4 of 5)

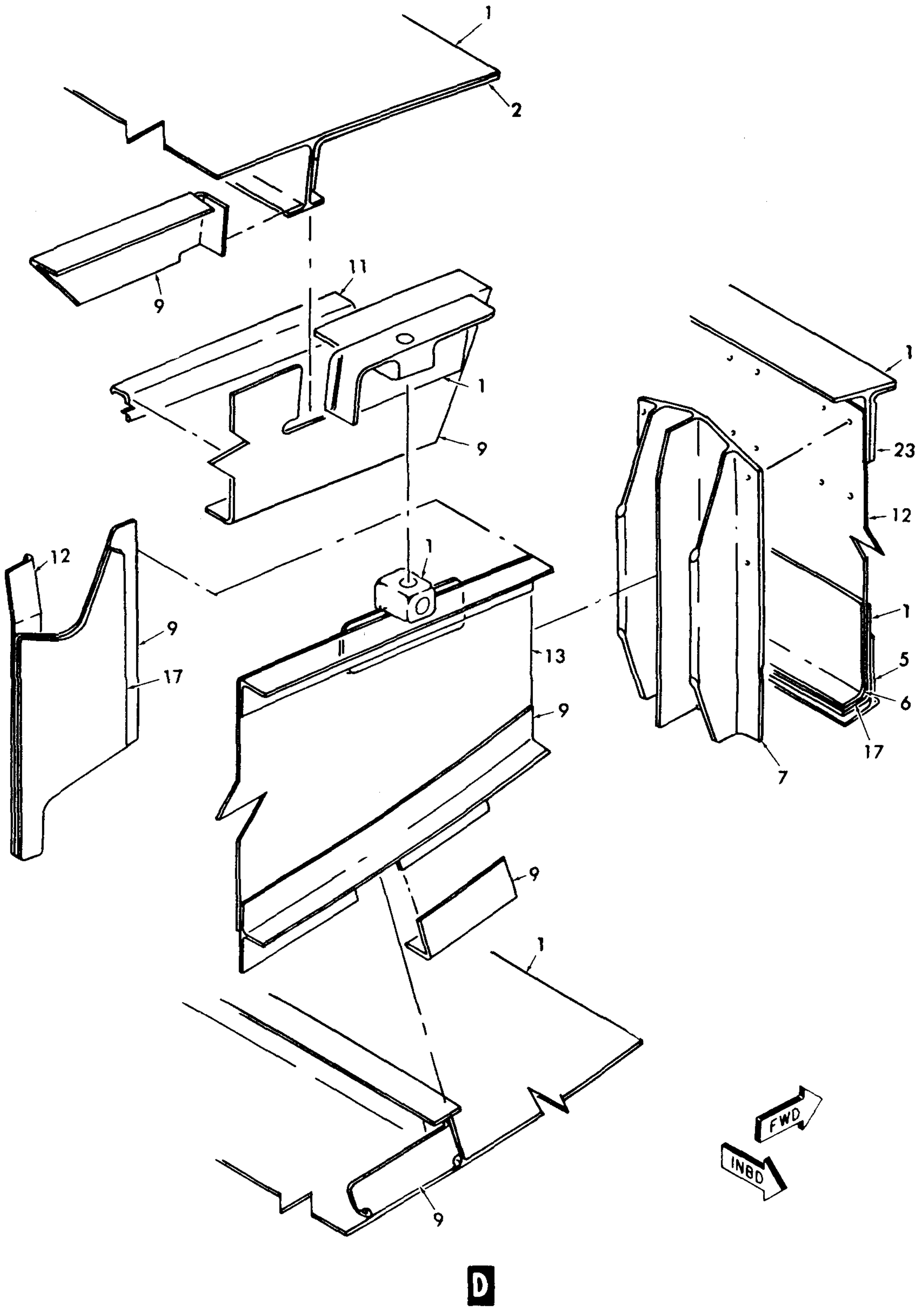
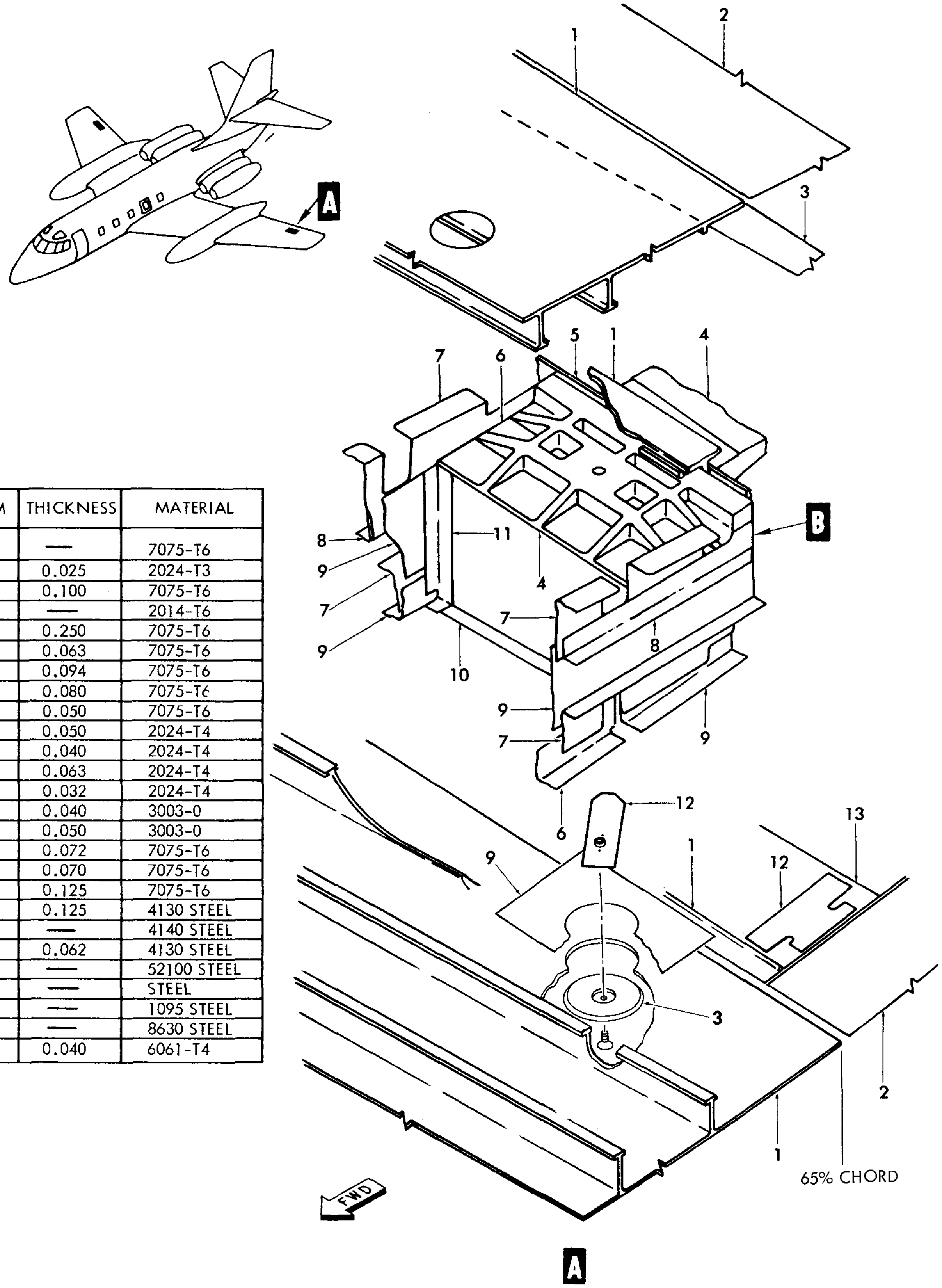
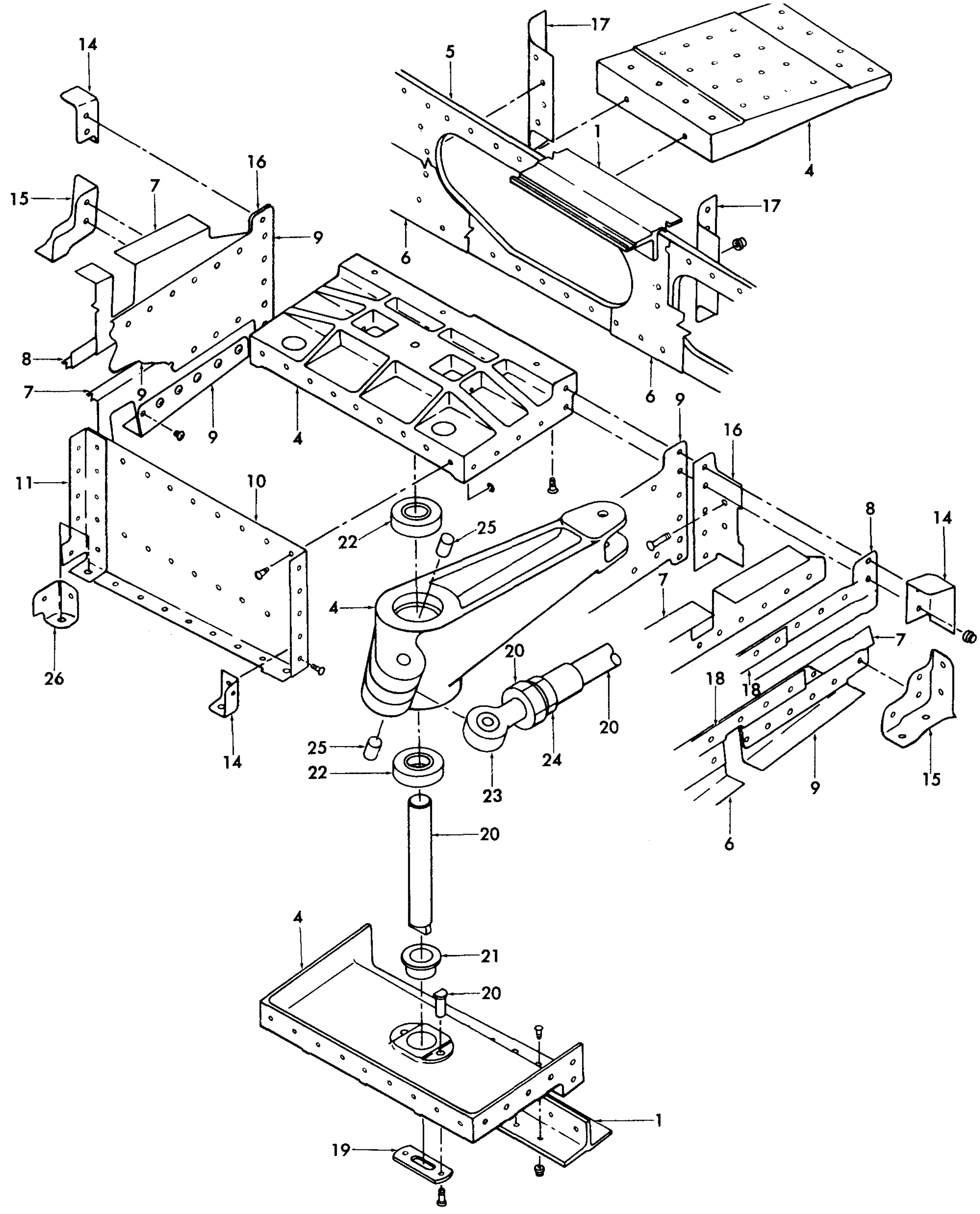


Figure 59-85. Inspection of Leading Edge Flap Actuator Region (Sheet 5 of 5)



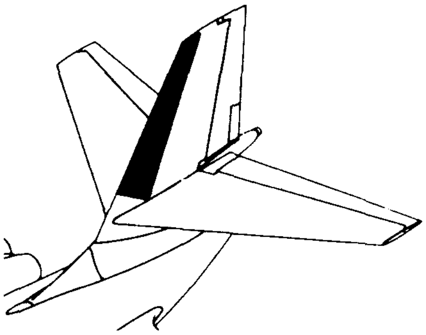
ITEM	THICKNESS	MATERIAL
1	—	7075-T6
2	0.025	2024-T3
3	0.100	7075-T6
4	—	2014-T6
5	0.250	7075-T6
6	0.063	7075-T6
7	0.094	7075-T6
8	0.080	7075-T6
9	0.050	7075-T6
10	0.050	2024-T4
11	0.040	2024-T4
12	0.063	2024-T4
13	0.032	2024-T4
14	0.040	3003-0
15	0.050	3003-0
16	0.072	7075-T6
17	0.070	7075-T6
18	0.125	7075-T6
19	0.125	4130 STEEL
20	—	4140 STEEL
21	0.062	4130 STEEL
22	—	52100 STEEL
23	—	STEEL
24	—	1095 STEEL
25	—	8630 STEEL
26	0.040	6061-T4

Figure 59-86. Inspection of Aileron Bellcrank Support Region (Sheet 1 of 2)

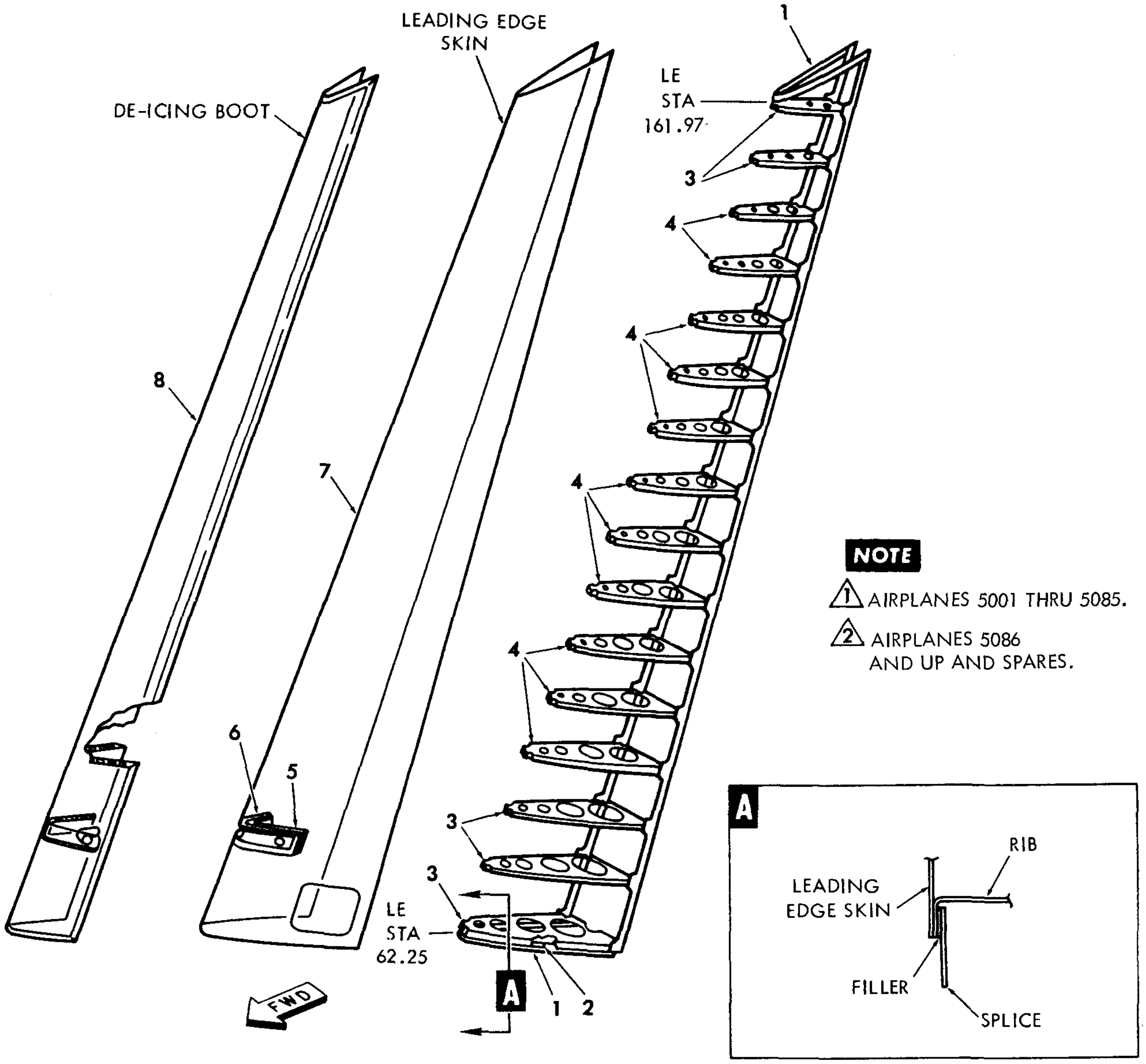


B

Figure 59-86. Inspection of Aileron Bellcrank Support Region (Sheet 2 of 2)



	ITEM	THICKNESS	MATERIAL
	1	0.040	2024-T3
	2	0.025	2024-T3
	3	0.040	2024-T4
	4	0.032	2024-T4
	5	0.125	2024-T3
	6	0.032	2024-T3
△1	7	0.032	7075-T6
△2	7	0.040	7075-T6 OR 2024-T3
	8	0.125	RUBBER AND FABRIC



NOTE

△1 AIRPLANES 5001 THRU 5085.

△2 AIRPLANES 5086 AND UP AND SPARES.

Figure 59-87. Inspection of Vertical Stabilizer Leading Edge

ITEM	THICKNESS	MATERIAL
1	0.063	7075-T6
2	0.160 (MAX)	7075-T6
3	0.170 (MAX)	7075-T6
4	0.050	7075-T6
5	0.040	2024-T4
6	0.120 (MAX)	7075-T6
7	0.063	2024-T4
8	0.032	2024-T4
9	0.050	2024-T4
10	0.078	7075-T6
11	0.160	2024-T3
12	0.050	2024-T3
13	0.109	2024-T4
14	——	8630 STEEL
15	0.125 WALL	NITRALLOY 135
16	0.032	1100-H19
17	0.094 WALL	NITRALLOY135
18	——	7075-T6
19	0.561 DIA	NITRALLOY 135
20	0.062	1100-H19
21	0.020	302 CRES

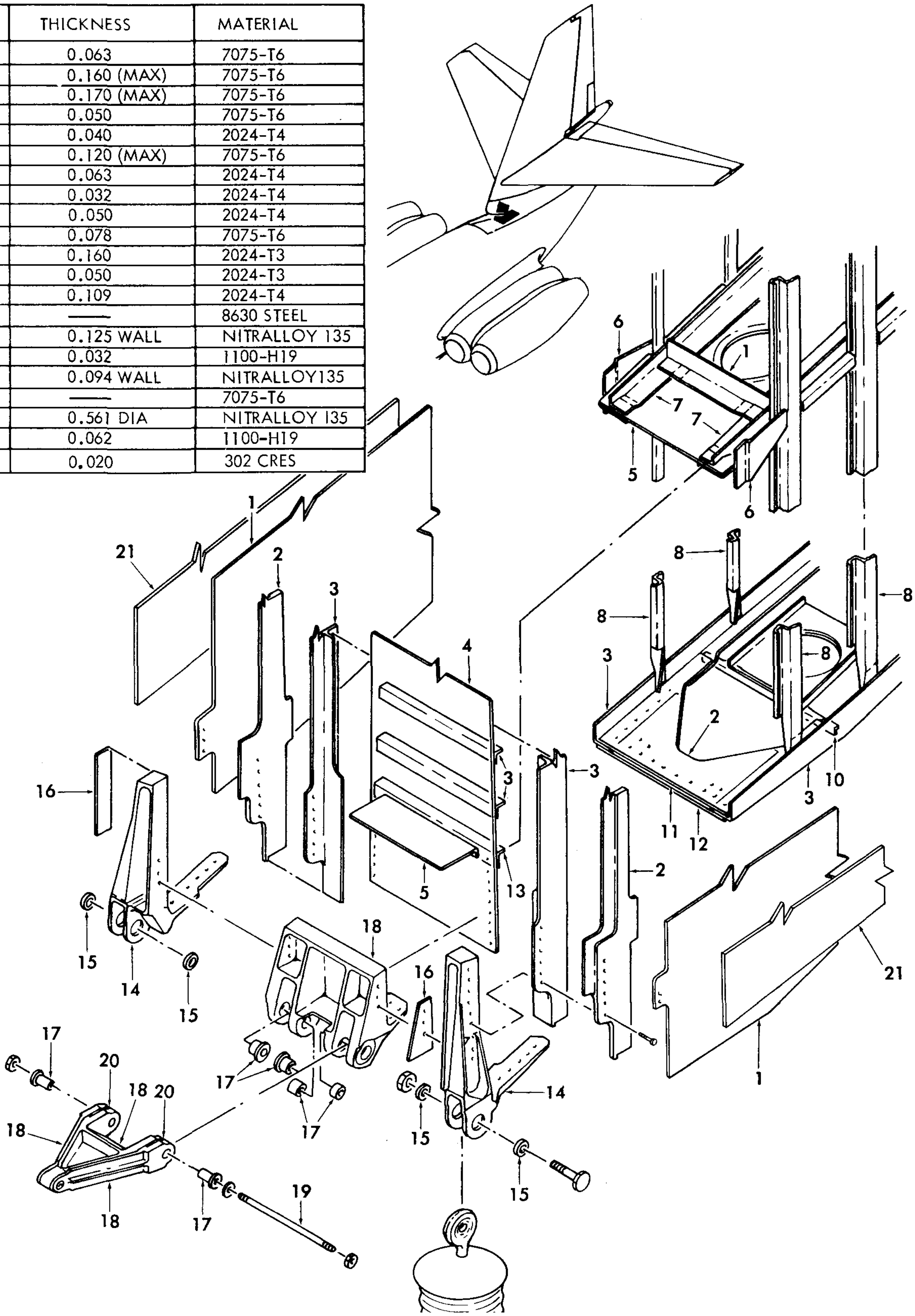
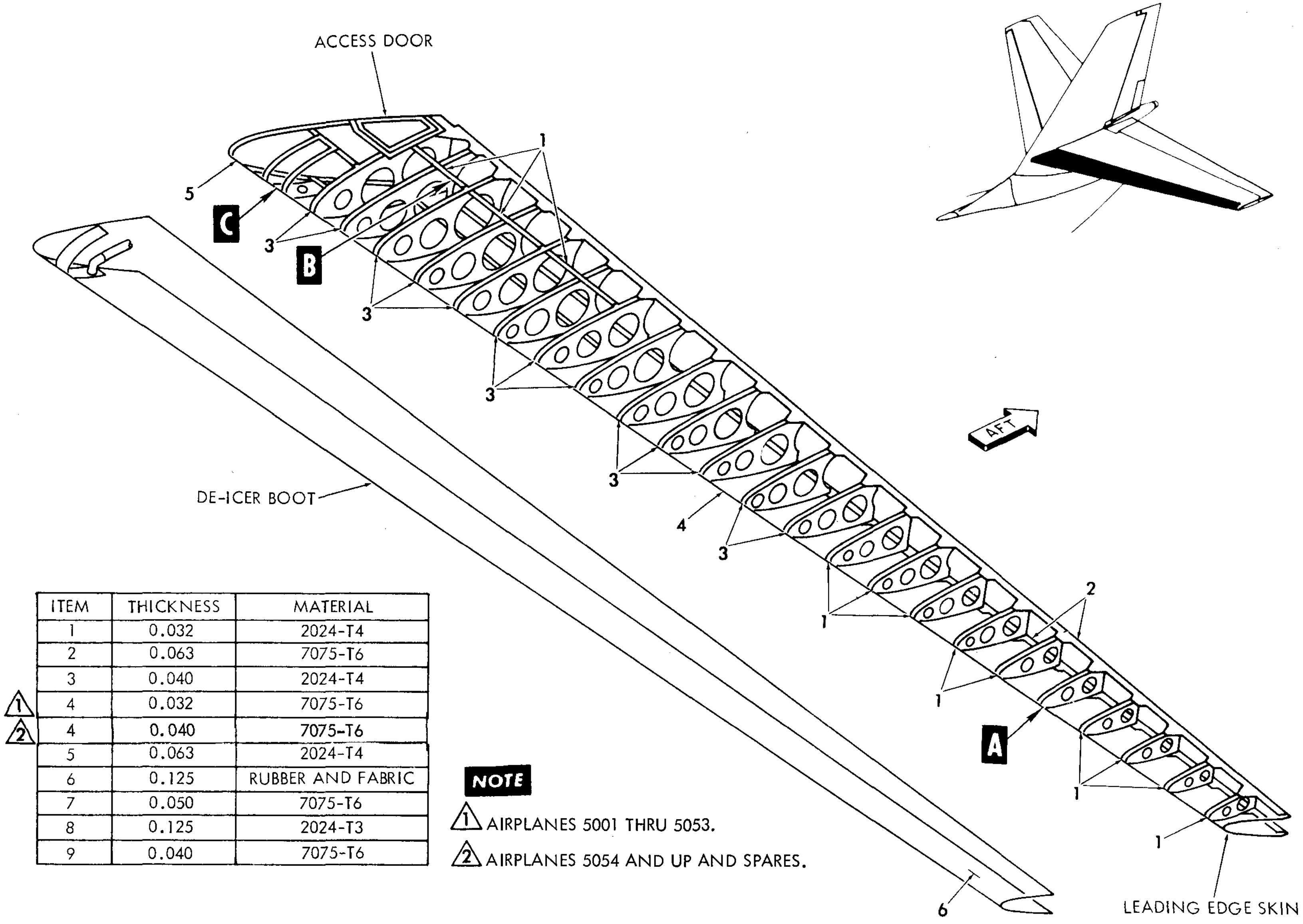


Figure 59-88. Inspection of Vertical Stabilizer Forward Support Region



ITEM	THICKNESS	MATERIAL
1	0.032	2024-T4
2	0.063	7075-T6
3	0.040	2024-T4
△1 4	0.032	7075-T6
△2 4	0.040	7075-T6
5	0.063	2024-T4
6	0.125	RUBBER AND FABRIC
7	0.050	7075-T6
8	0.125	2024-T3
9	0.040	7075-T6

NOTE

△1 AIRPLANES 5001 THRU 5053.

△2 AIRPLANES 5054 AND UP AND SPARES.

Figure 59-89. Inspection of Horizontal Stabilizer Leading Edge (Sheet 1 of 2)

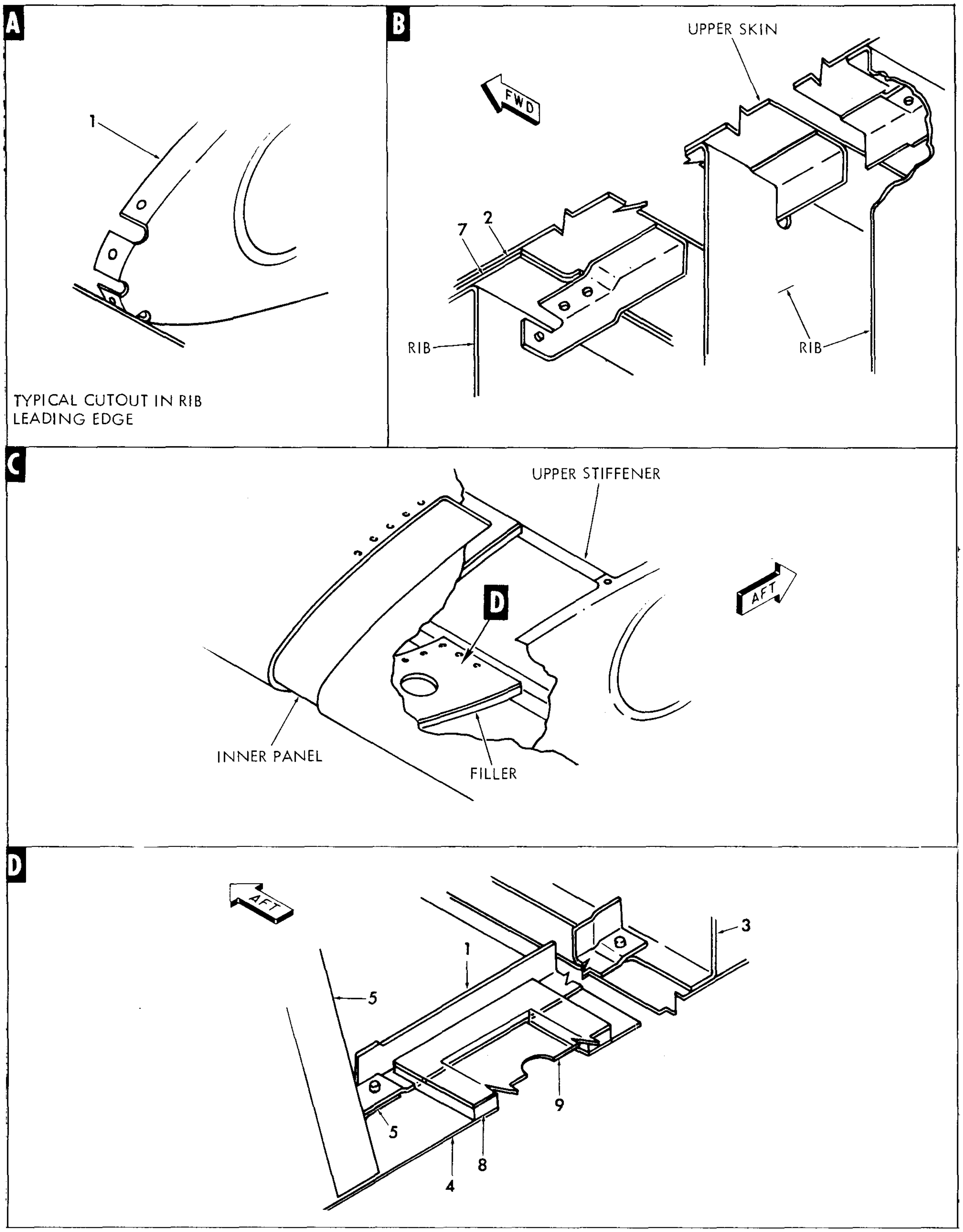
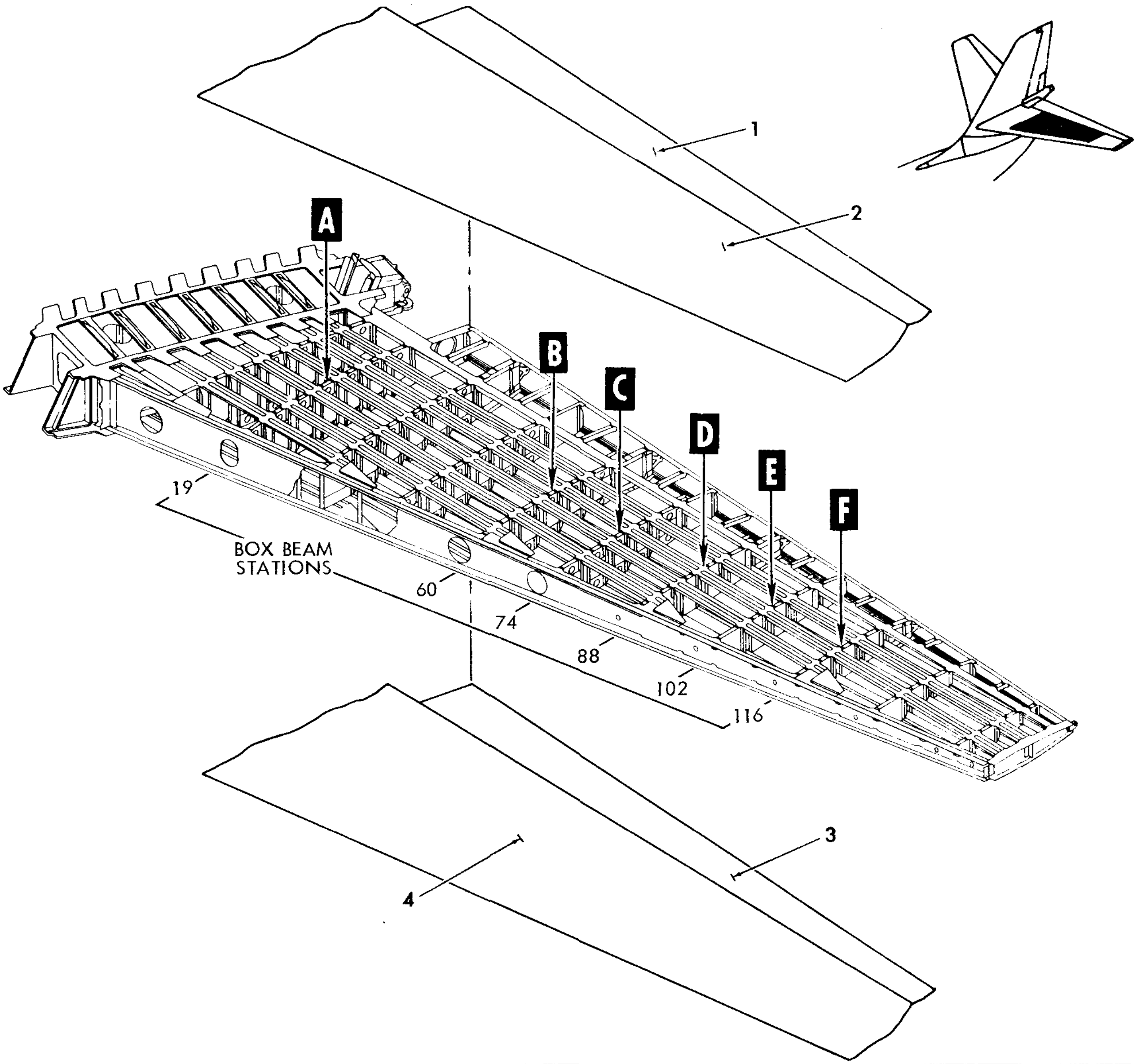


Figure 59-89. Inspection of Horizontal Stabilizer Leading Edge (Sheet 2 of 2)



ITEM	THICKNESS	MATERIAL	ITEM	THICKNESS	MATERIAL
1	0.032	7075-T6	14	0.050	2024-T4
2	0.030 (MIN), 0.050 (MAX.)	7075-T6	15	0.040	2024-T4
3	0.030 (MIN), 0.050 (MAX.)	2024-T3	16	0.050 (MIN), 0.190 (MAX.)	7075-T6
4	0.050	7075-T6	17	0.070	7075-T6
5	0.066 (MIN), 0.190 (MAX.)	7075-T6	18	0.040	7075-T6
6	0.027 (MIN), 0.063 (MAX.)	7075-T6	19	0.040	2024-T3
7	0.030 (MIN), 0.105 (MAX.)	2024-T4	20	0.032	2024-T3
8	0.063	7075-T6	21	0.063	2024-T3
9	0.071	2024-T4	22	0.065	2024-T4
10	0.040 (MIN), 0.060 (MAX.)	2024-T4	23	0.065 (MIN), 0.312 (MAX.)	7075-T6
11	0.064 (MIN), 0.089 (MAX.)	7075-T6	24	0.070	2024-T4
12	0.005 (MIN), 0.050 (MAX.)	2024-T4	25	0.050 (MIN), 0.061 (MAX.)	7075-T6
13	0.050	2024-T3	26	0.005 (MIN), 0.050 (MAX.)	2024-T4

Figure 59-90. Inspection of Horizontal Stabilizer Box Section (Sheet 1 of 5)

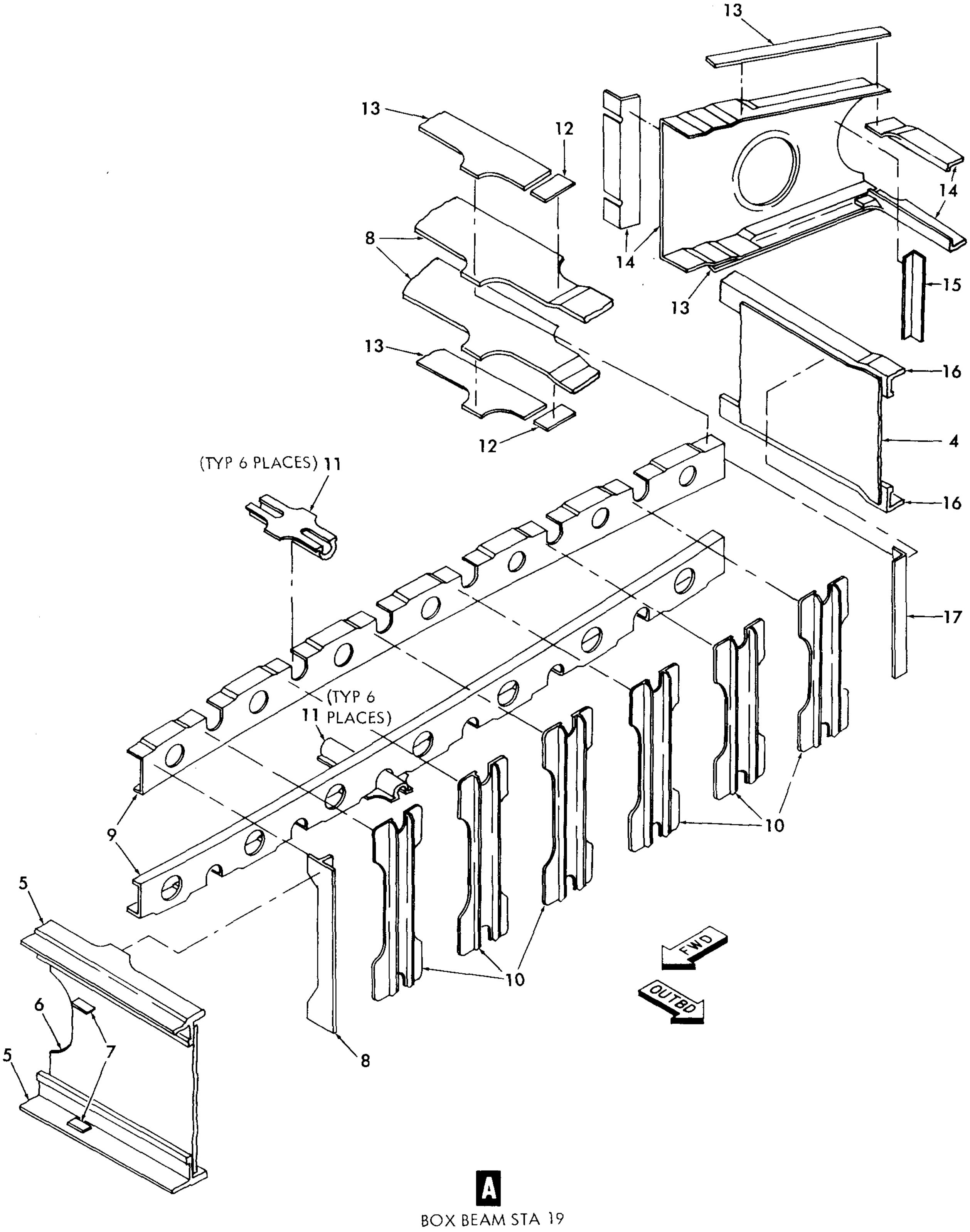


Figure 59-90. Inspection of Horizontal Stabilizer Box Section (Sheet 2 of 5)

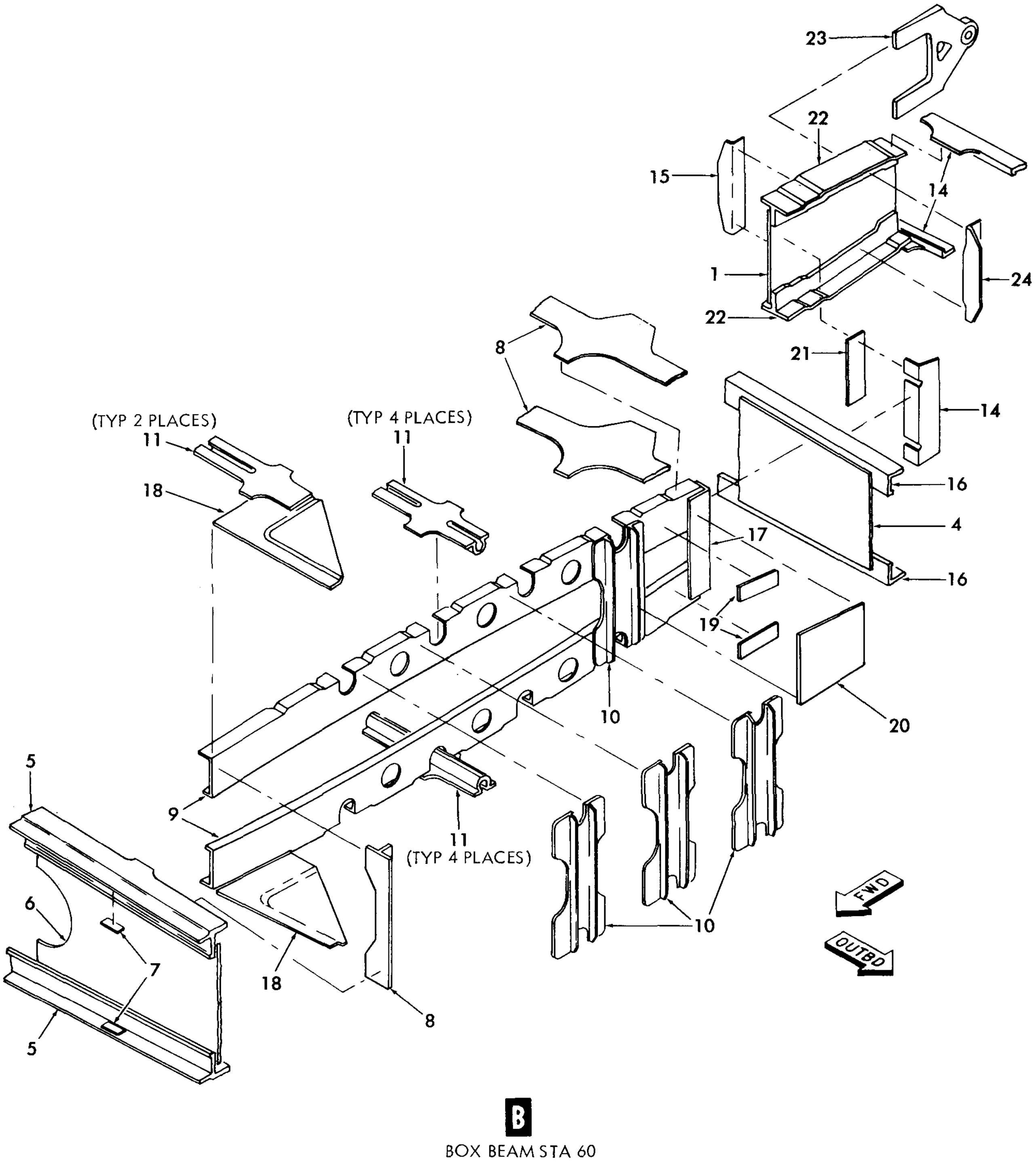


Figure 59-90. Inspection of Horizontal Stabilizer Box Section (Sheet 3 of 5)

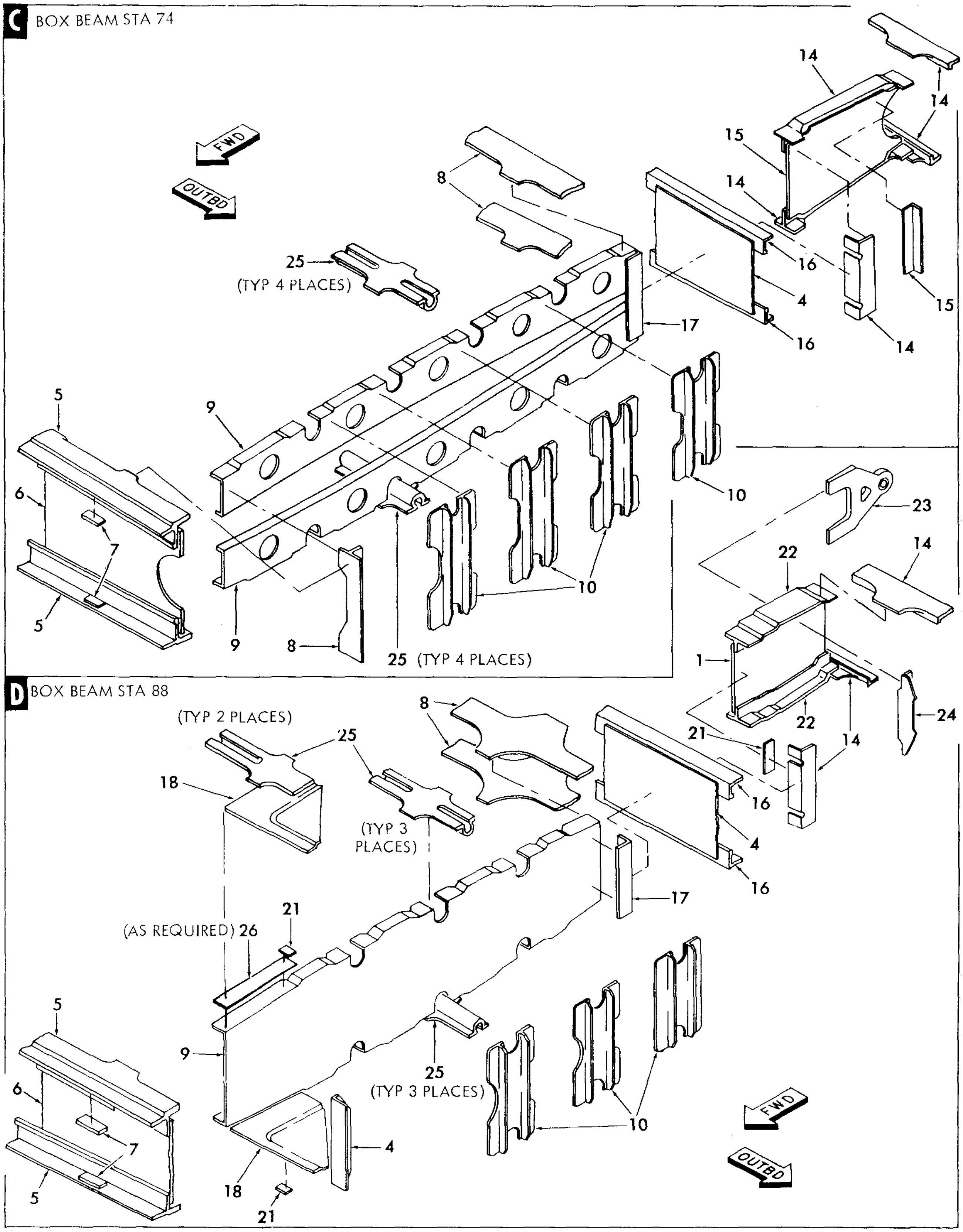
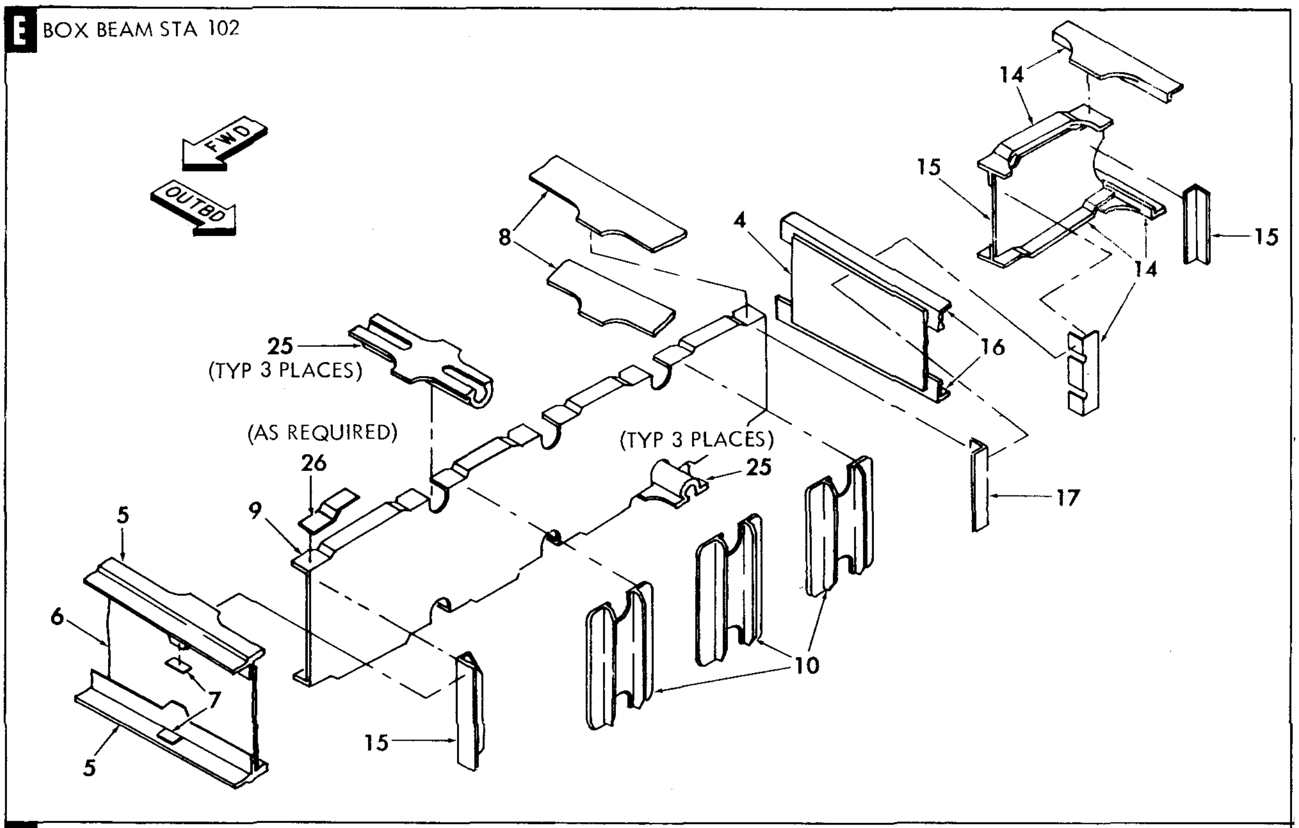


Figure 59-90. Inspection of Horizontal Stabilizer Box Section (Sheet 4 of 5)

E BOX BEAM STA 102



F BOX BEAM STA 116

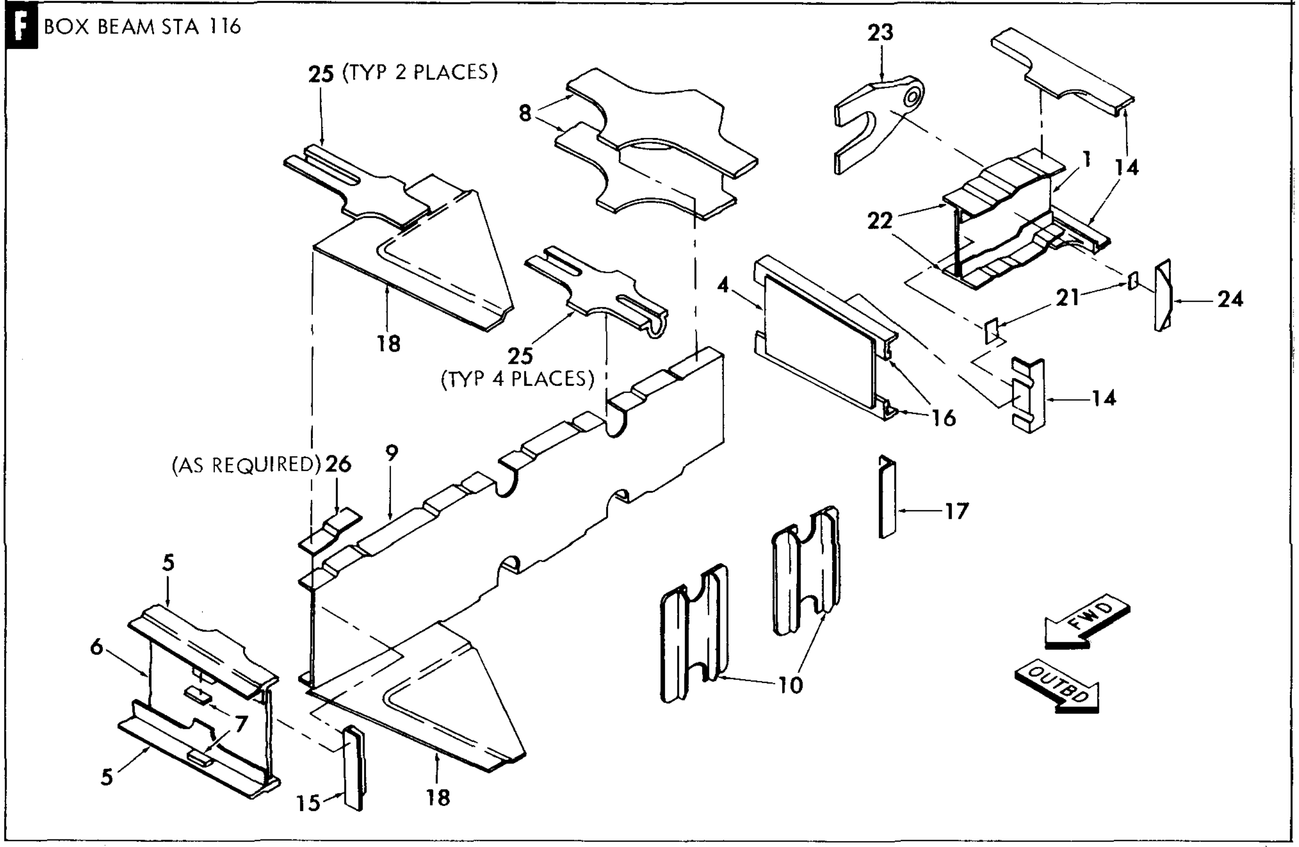
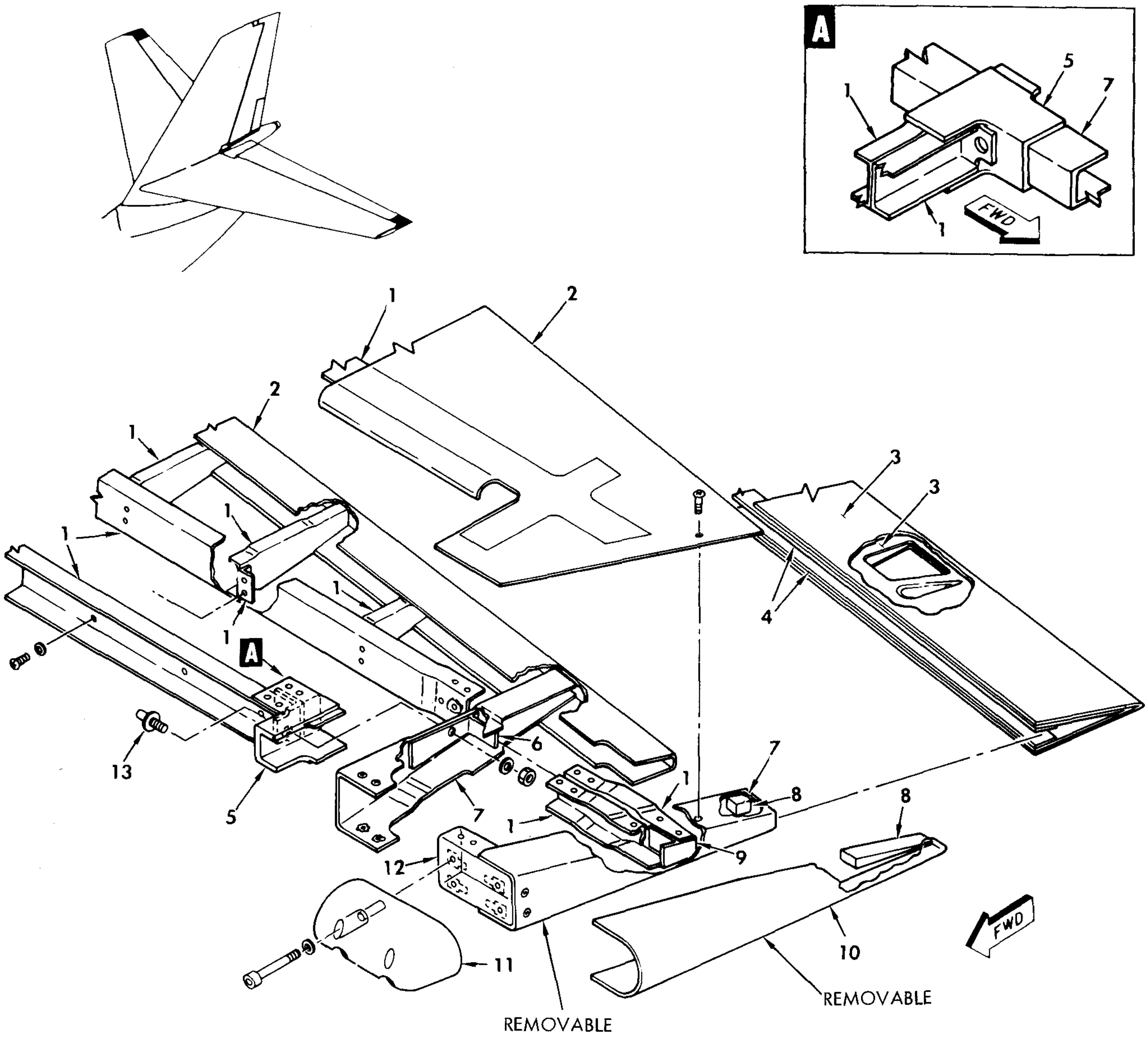


Figure 59-90. Inspection of Horizontal Stabilizer Box Section (Sheet 5 of 5)



INDEX NO.	THICKNESS	MATERIAL
1	0.040	2024-T4
2	0.050	7075-T6
3	0.012	2024-T4
4	0.025	2024-T4
5	0.063	7075-T6
6	0.094	2024-T4
7	0.063	2024-T4
8	———	COTTON PHENOLIC
9	0.094	7075-T6
10	0.040	6061-T4
11	2.55 (Max.)	DEPLETED URANIUM
12	0.060	2024-T4
13	0.500	4130 STEEL

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Figure 59-91. Inspection of Elevator Outboard Counterweight Region

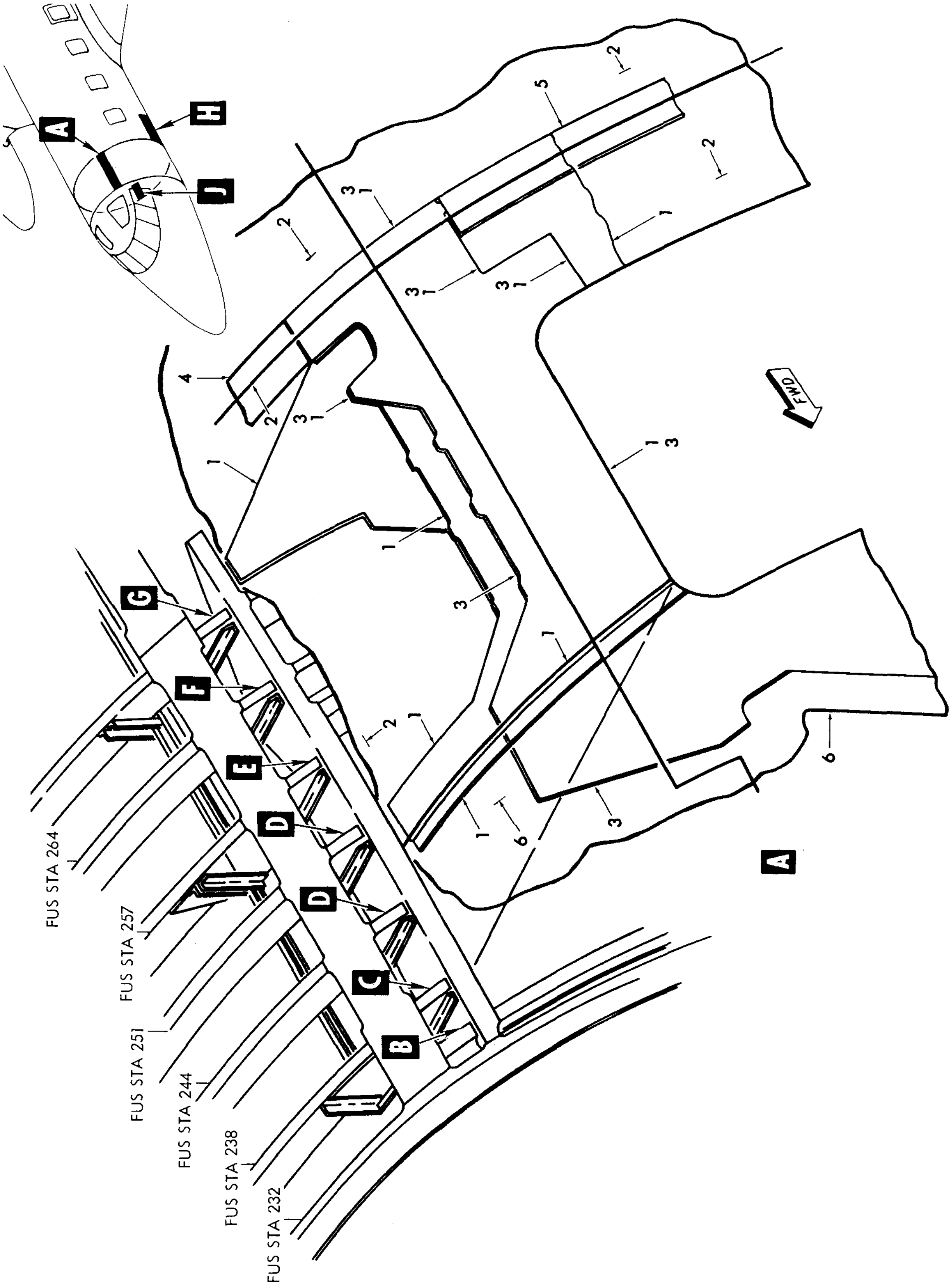


Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 1 of 7)

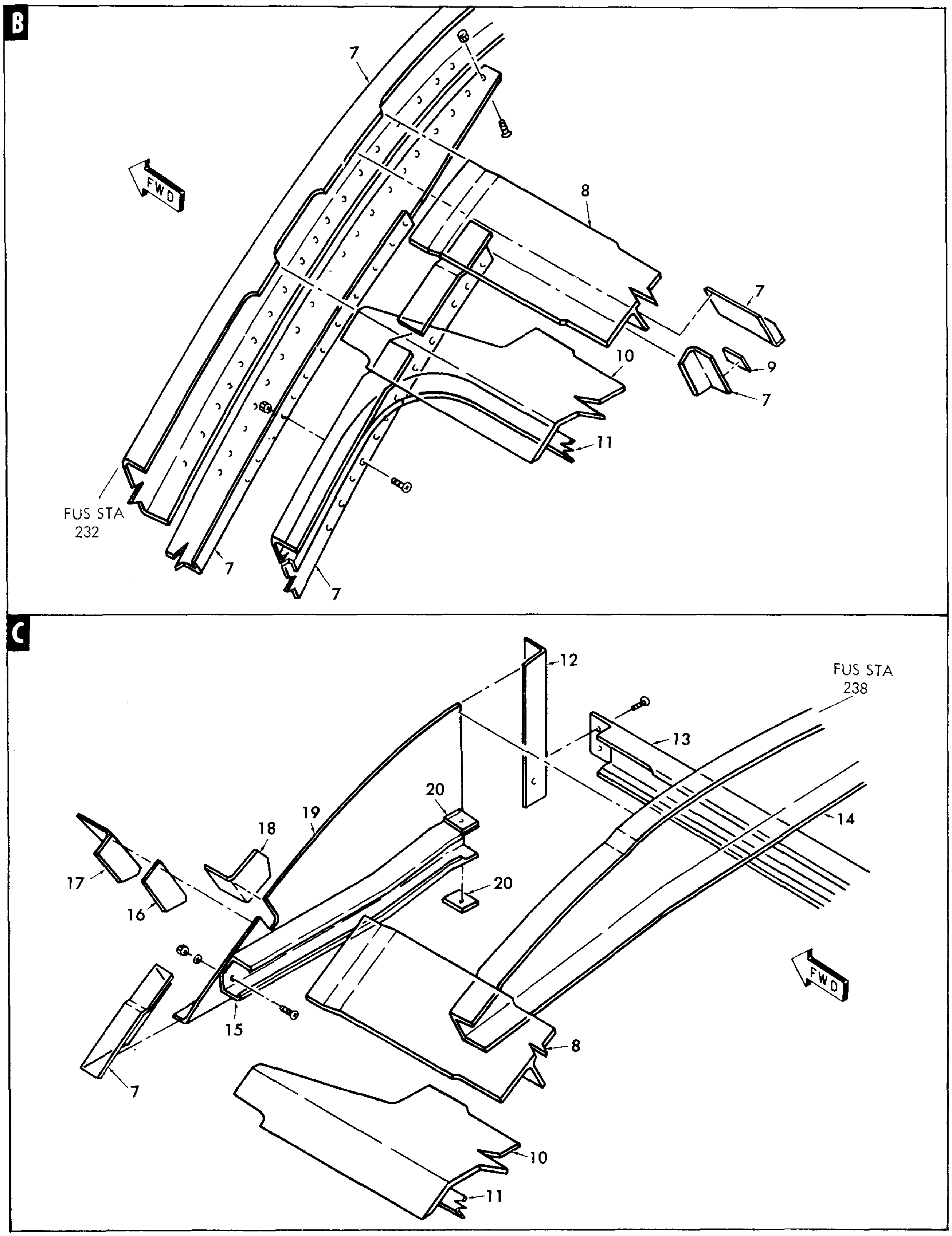
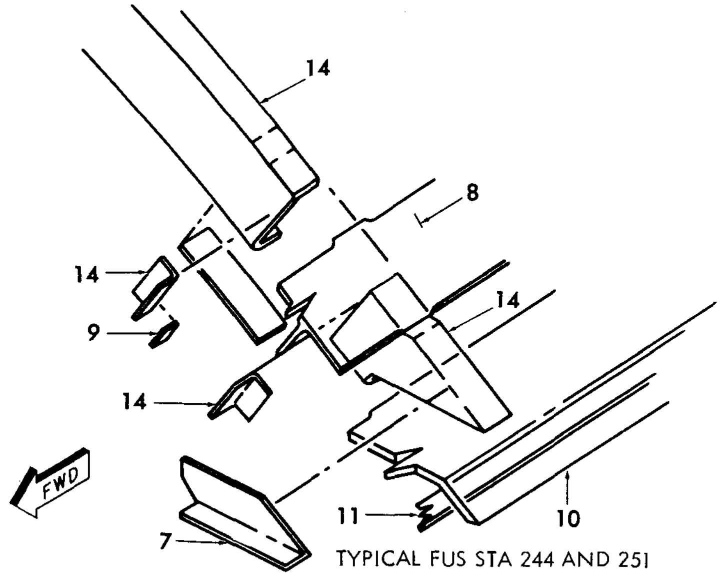


Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 2 of 7)



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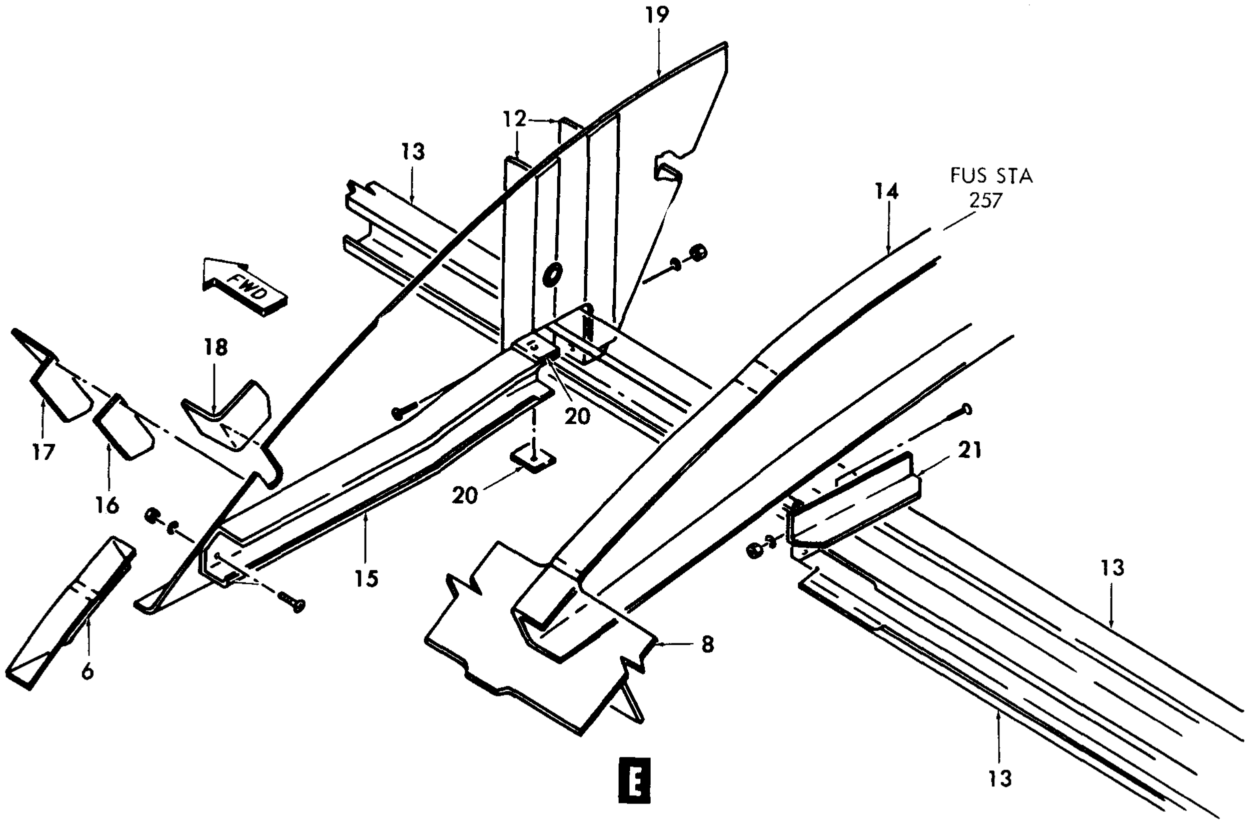

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Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 3 of 7)

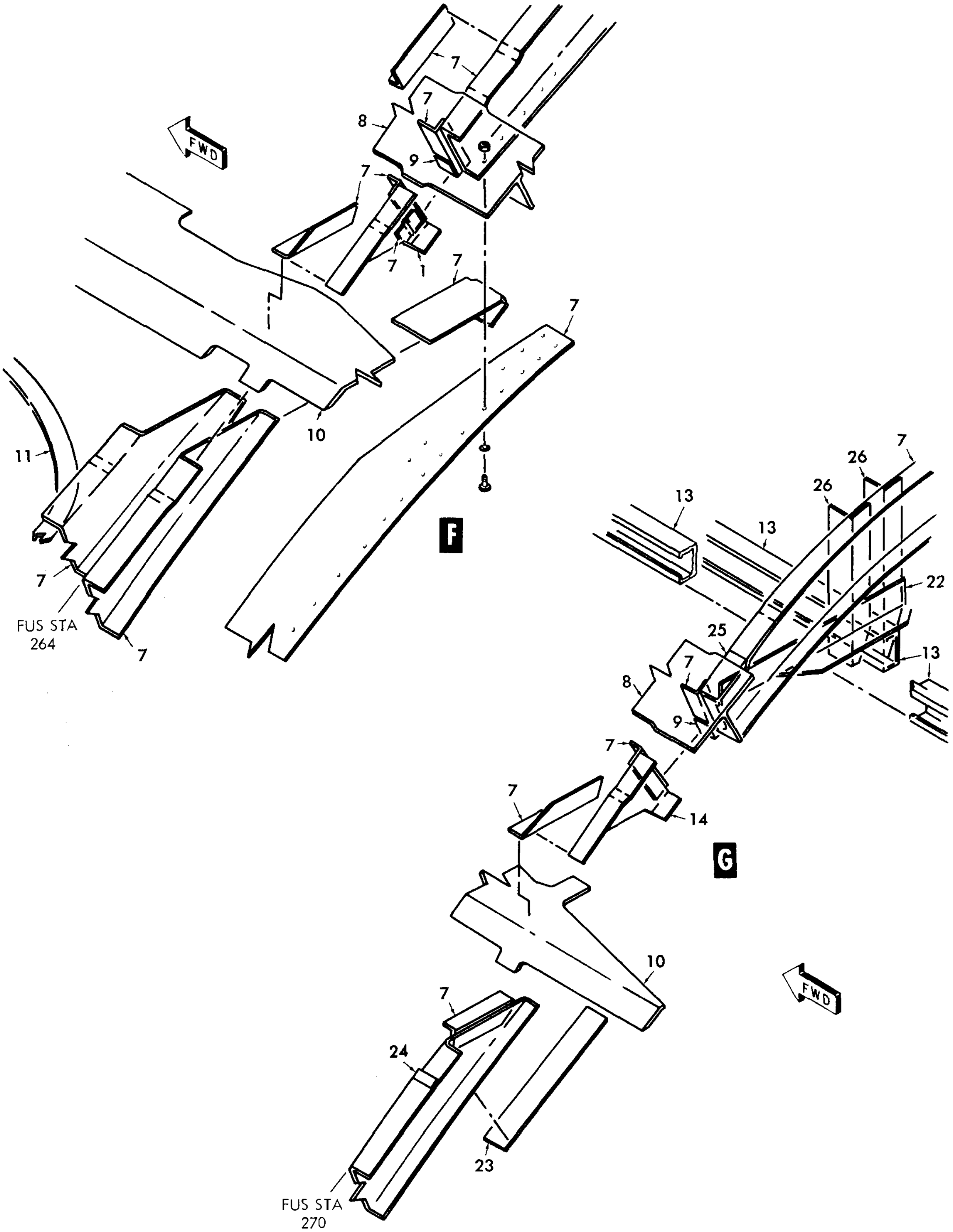


Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 4 of 7)

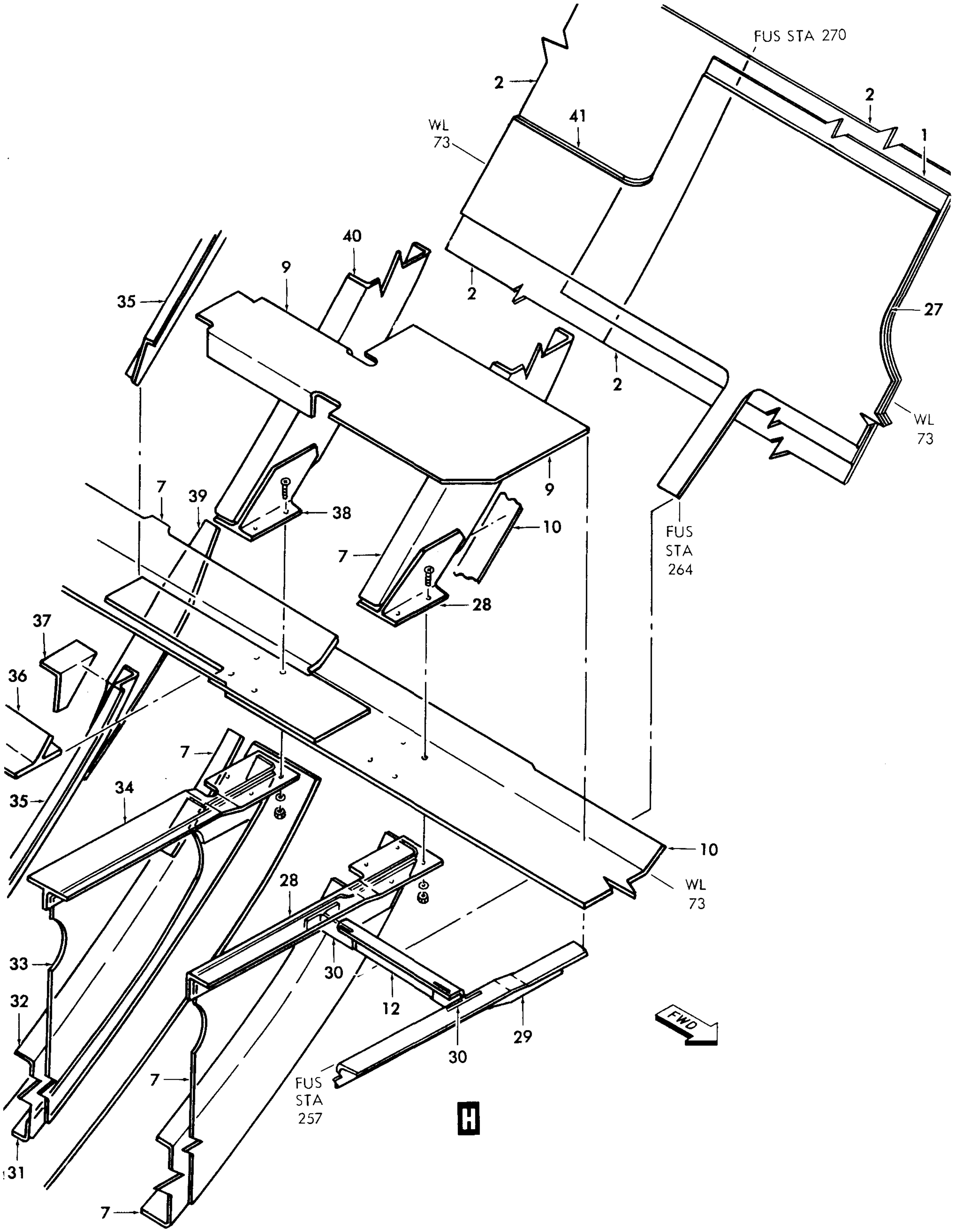


Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 5 of 7)

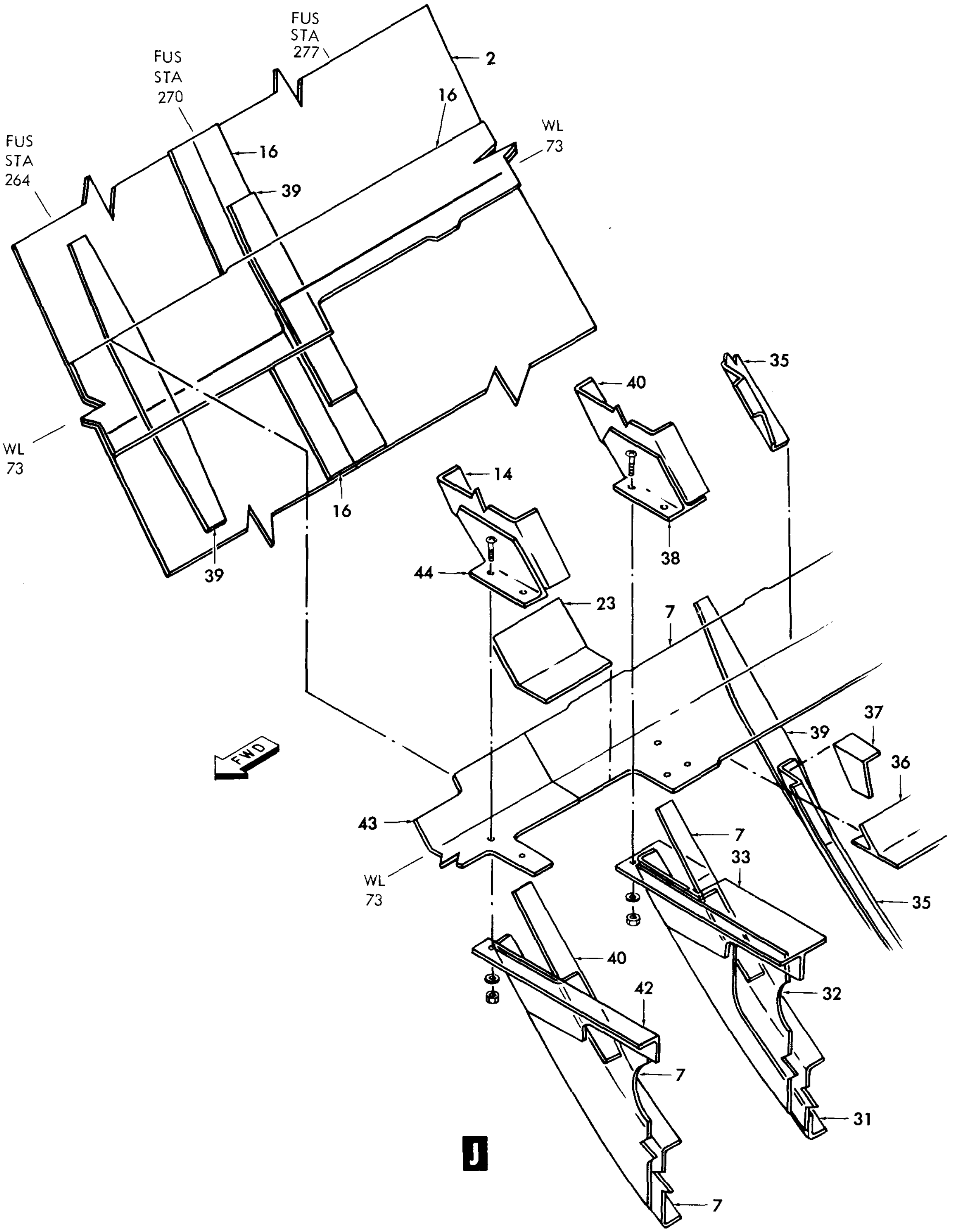
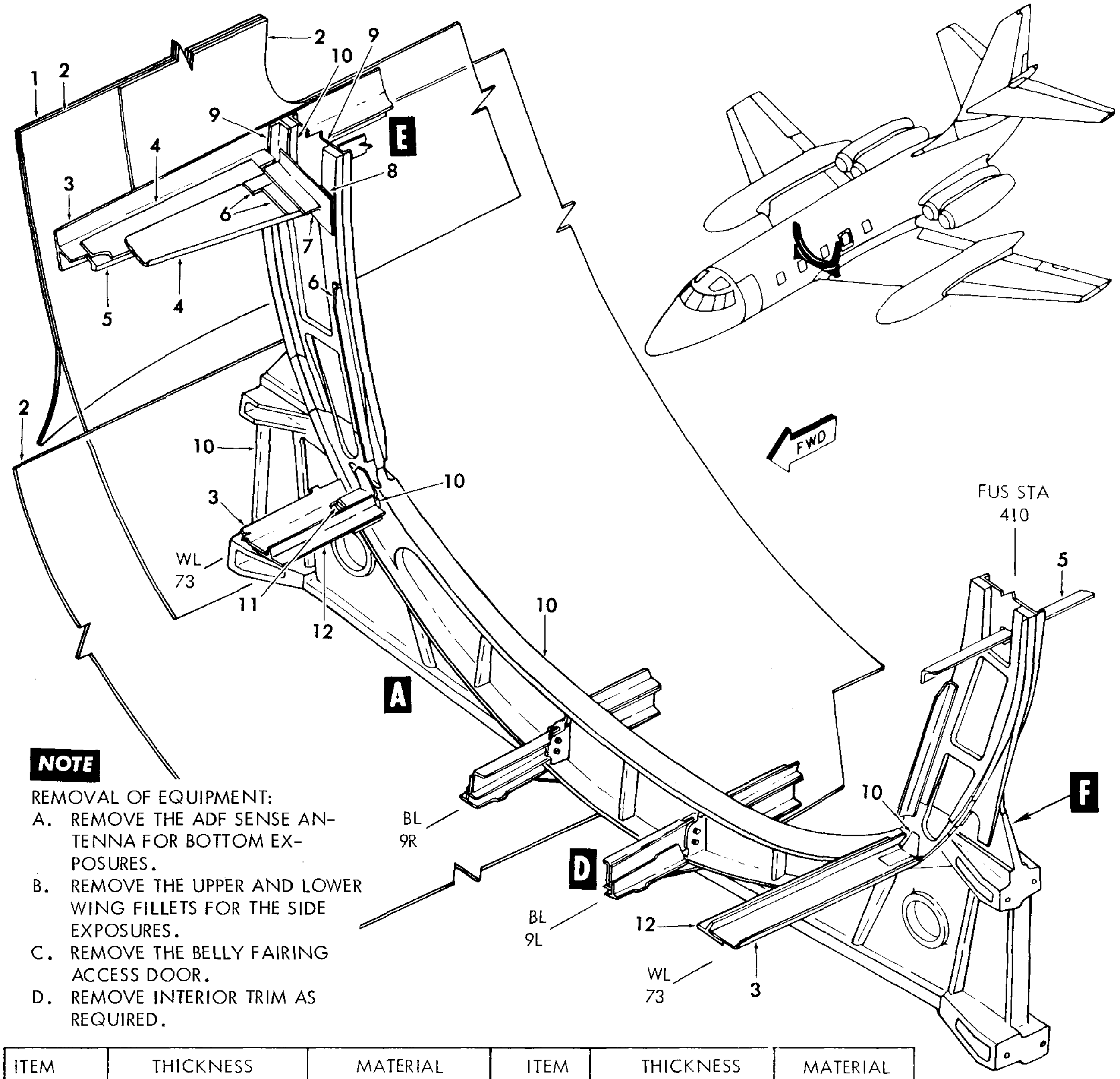


Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 6 of 7)

ITEM	THICKNESS	MATERIAL
1	0.025	2024-T3
2	0.032	2024-T3
3	0.080	2024-T4
4	0.040	2024-T3
5	0.032	2024-T4
6	0.040	2024-T4
7	0.063	7075-T6
8	0.080	7075-T6
9	0.050	2024-T3
10	0.125	7075-T6
11	0.100	2024-T42
12	0.062	2024-T4
13	0.100	2024-T4
14	0.040	7075-T6
15	0.125	2024-T4
16	0.040	2024-T3
17	0.063	2024-T4
18	0.064	2024-T4
19	0.050	2024-T4
20	0.125	ALLUMINUM LAMINATE
21	0.062	2024-T4
22	0.065	2024-T4
23	0.080	7075-T6
24	WEDGE	2024-T3
25	WEDGE	2024-T3
26	0.062	7075-T6
27	0.063	2024-T3
28	0.125	7075-T6
29	0.156	7075-T6
30	0.081	2024-T6
31	0.100	7075-T6
32	0.060	7075-T6
33	0.064	7075-T6
34	0.100	7075-T6
35	0.062	7075-T6
36	0.100	7075-T6
37	0.062	2024-T4
38	0.125	TITANIUM
39	0.016	TITANIUM
40	0.071	7075-T6
41	0.025	2024-T4
42	0.125	7075-T6
43	0.063	7075-T6
44	0.125	7075-T6

Figure 59-92. Inspection of Fuselage Station 232 to 277 Skin Splice (Sheet 7 of 7)



NOTE

REMOVAL OF EQUIPMENT:

A. REMOVE THE ADF SENSE ANTENNA FOR BOTTOM EXPOSURES.

B. REMOVE THE UPPER AND LOWER WING FILLETS FOR THE SIDE EXPOSURES.

C. REMOVE THE BELLY FAIRING ACCESS DOOR.

D. REMOVE INTERIOR TRIM AS REQUIRED.

ITEM	THICKNESS	MATERIAL	ITEM	THICKNESS	MATERIAL
1	0.040	2024-T4	14	0.135 (Max)	2024-T4
2	0.040	2024-T3	15	0.016	2024-T3
3	0.062	7075-T6	16	0.190 (Max)	2024-T4
4	0.050	7075-T6	17	0.250 (Max)	7075-T6
5	0.220	7075-T6	18	0.060	7075-T6
6	0.090	7075-T6	19	0.063	2024-T3
7	0.120	7075-T6	20	0.060	7075-T6
8	0.280 (Max)	2024-T4	21	0.125	7075-T6
9	0.080	7075-T6	22	0.100 (Max)	7075-T6
10	——	7075-T6	23	0.38 (Max)	2024-T4
11	0.102	2024-T3	24	——	2014-T6
12	0.100 (Max)	7075-T6	25	0.032	2024-T3
13	0.133 (Max)	2024-T4			

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Figure 59-93. Inspection of Fuselage Station 410 Skin Splices (Sheet 1 of 3)

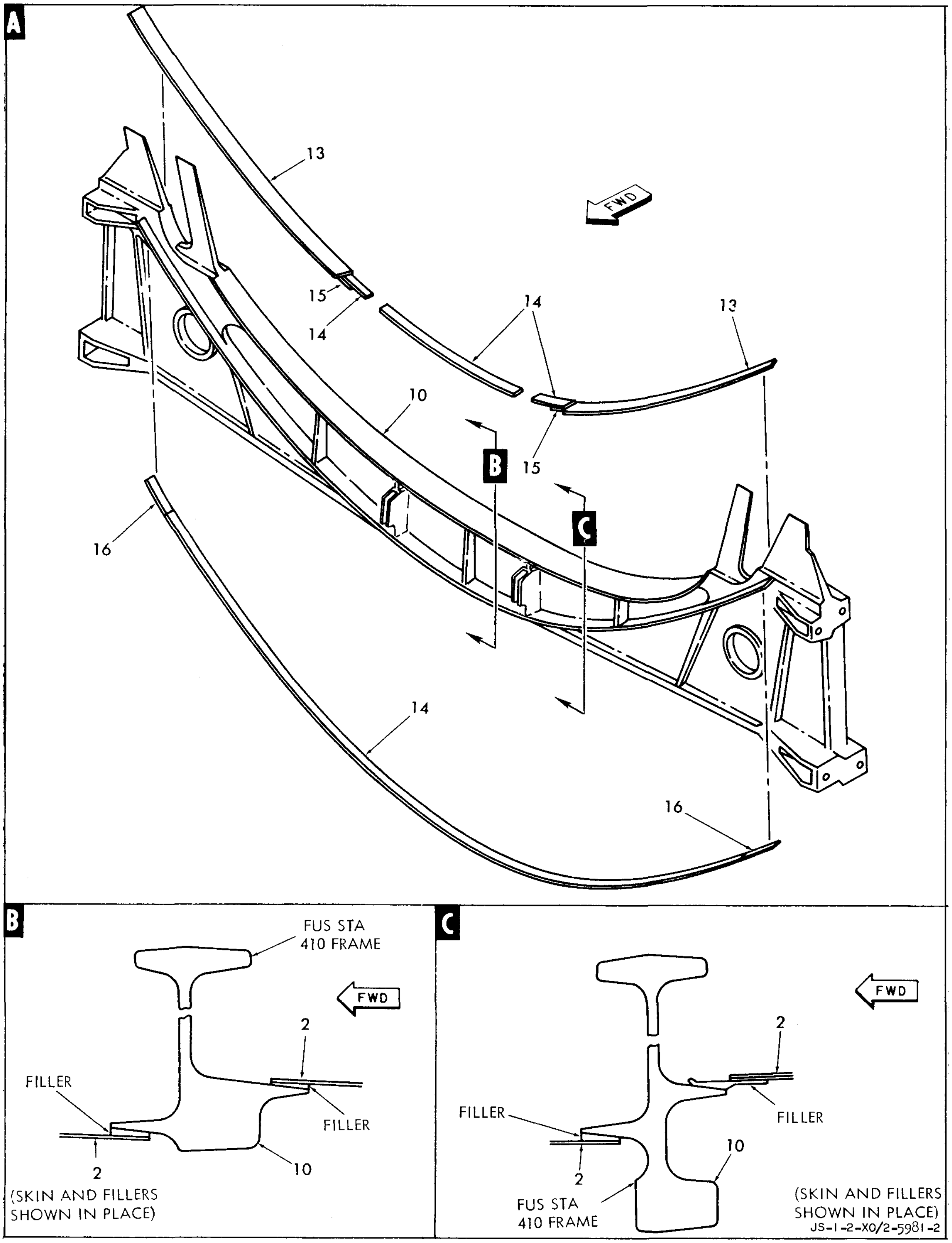


Figure 59-93. Inspection of Fuselage Station 410 Skin Splices (Sheet 2 of 3)

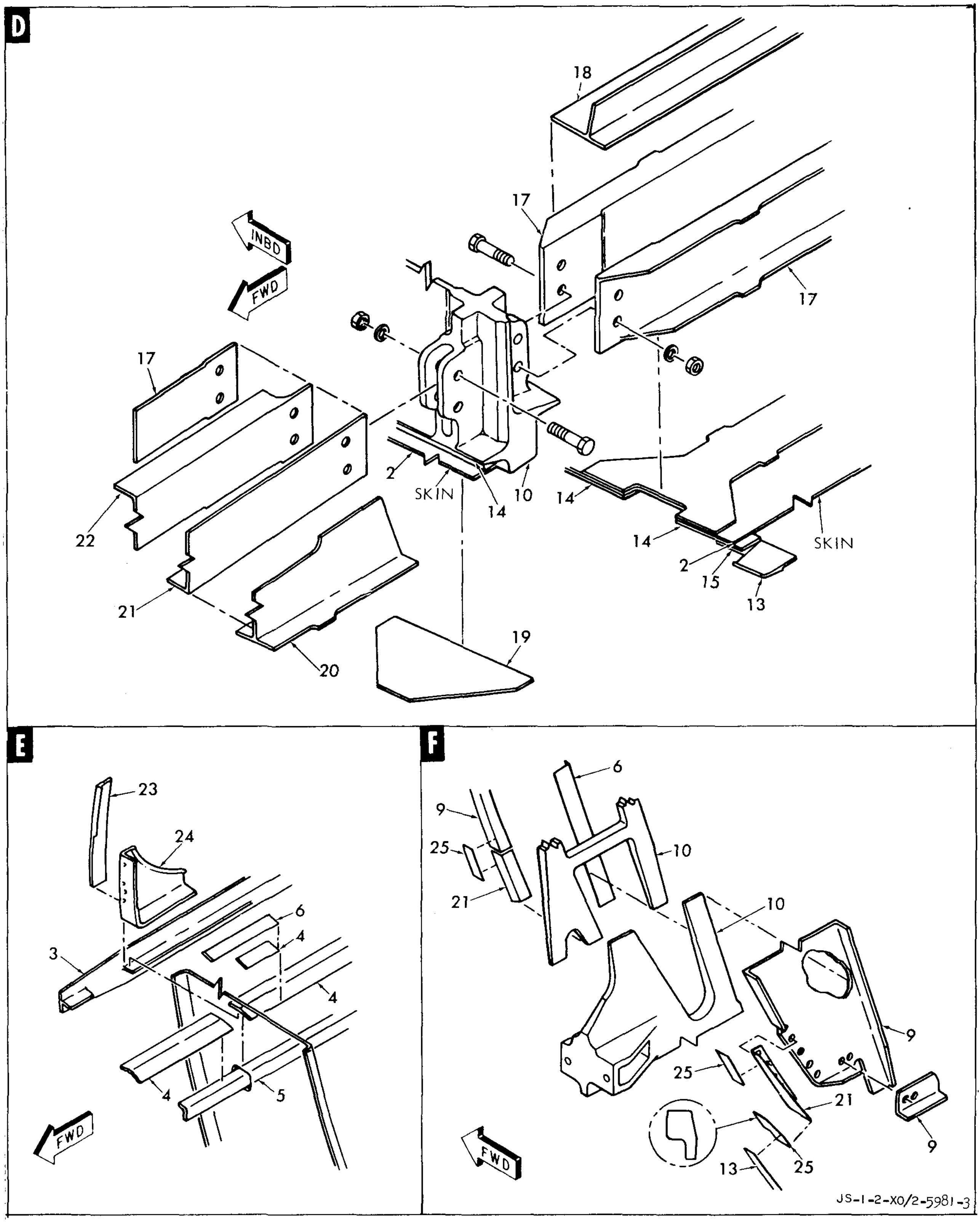
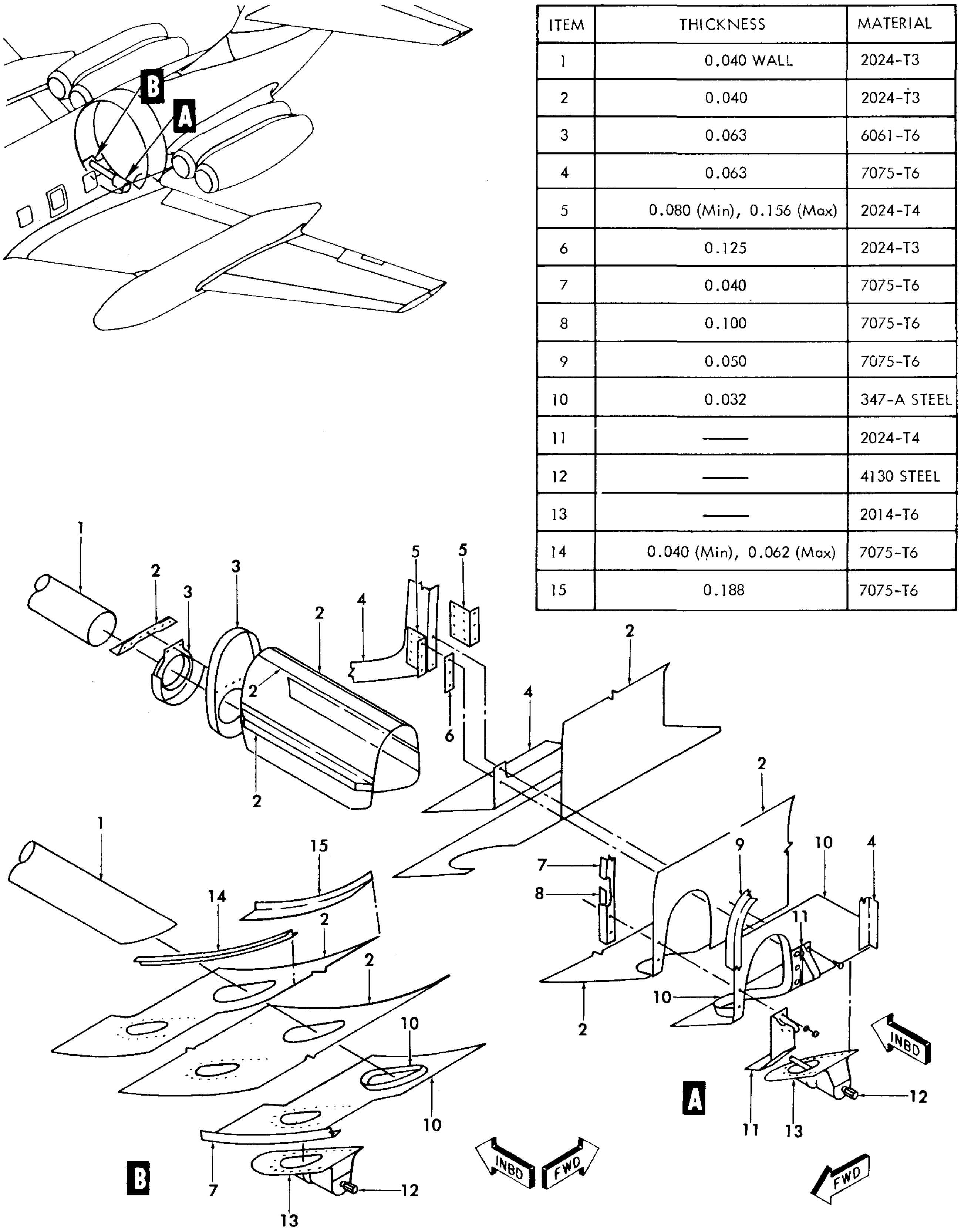


Figure 59-93. Inspection of Fuselage Station 410 Skin Splices (Sheet 3 of 3)



ITEM	THICKNESS	MATERIAL
1	0.040 WALL	2024-T3
2	0.040	2024-T3
3	0.063	6061-T6
4	0.063	7075-T6
5	0.080 (Min), 0.156 (Max)	2024-T4
6	0.125	2024-T3
7	0.040	7075-T6
8	0.100	7075-T6
9	0.050	7075-T6
10	0.032	347-A STEEL
11	——	2024-T4
12	——	4130 STEEL
13	——	2014-T6
14	0.040 (Min), 0.062 (Max)	7075-T6
15	0.188	7075-T6

Figure 59-94. Inspection of Fuselage Station 526 to 539 Skin Splices

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CHAPTER 71

POWER PLANT

Contents Page

GENERAL 71-1

ENGINE BUILDUP 71-1

ENGINE MOUNTING 72-2

RUBBER ENGINE MOUNT INSPECTION AND MAINTENANCE 71-2

METAL ENGINE MOUNT INSPECTION AND MAINTENANCE 71-2

FIGURES
71-1 Engine Buildup — Firewall, Tailpipe, Blanket, and Manifold 71-4
71-2 Engine Buildup — Drain Bracket Subassembly (Engines without Drain Sump Modification) 71-5
71-2A Engine Buildup— Drain Bracket Subassembly (Engines with Drain Sump Modification 71-6A
71-3 Engine Buildup — Hydraulic System Installation 71-7
71-4 Engine Buildup — Breathers, Tubing and Ducts Installation 71-8
71-5 Engine Buildup — Instrumentation and Air Inlet Duct Installation 71-9
71-6 Engine Buildup — Engine Controls and Electrical Installation 71-11
71-7 Engine Buildup — Engine Mounts and Ignition Exciter Heat Shields Installation 71-13
71-8 Rubber Engine Mount Inspection Data 71-15
71-9 Metal Engine Mount Inspection Data 71-16

71.1 GENERAL. The JetStar is powered by four Pratt and Whitney JT12A-6 (aircraft 5001 through 5045), JT12A-6A (5046 thru 5092, 5094 thru 5096 and 5121) or JT12A-8 (5093, 5097 thru 5120, 5122 and up) turbo-jet engines mounted in pairs in a nacelle on each side of the aft fuselage. The JT12A engine is a continuous flow, gas turbine power plant using a two-stage, reaction turbine to drive a single rotor, nine stage, axial flow compressor. Each engine is equipped with a clamshell-type thrust reverser. Refer to the appropriate chapters of this manual for detailed information regarding the following power plant provisions:

NACELLE	CHAPTER 54
ENGINE	CHAPTER 72
ENGINE FUEL AND CONTROL	CHAPTER 73
ENGINE CONTROLS	CHAPTER 76
ENGINE STARTING AND IGNITION	CHAPTER 74
ENGINE INDICATING	CHAPTER 77
ENGINE FIRE PROTECTION	CHAPTER 26
EXHAUST	CHAPTER 78
ENGINE ICE PROTECTION	CHAPTER 30
FUEL SYSTEM	CHAPTER 28

71.2 ENGINE BUILDUP.

71.2.1 GENERAL. This chapter contains instructions for building up a bare engine for installation in the JetStar. These instructions apply to both the JT12A-6, the JT12A-6A, and the JT12A-8 engines.

71.2.2 PREPARATION.

a. Remove the engine from the shipping box, and install it on a suitable buildup stand or dolly in accordance with the instructions in the Pratt & Whitney Maintenance Manual.

b. Depreserve the engine in accordance with the instructions in the Pratt and Whitney Maintenance Manual.

Note

If desired, the engine may be depreserved after installation.

71.2.2 (Cont)

CAUTION

Throughout buildup operations, protect the engine from entry of foreign matter by installing suitable covers over all openings, including engine air inlet, exhaust outlet, all ports, and any open lines or fittings which may result from buildup sequence. Inspect at each stage of buildup to ensure that foreign matter is not left in the engine or systems. Perform the buildup operation in a clean area protected from wind-blown dust, and permit access only by personnel actually engaged in the buildup operation. Keep the bleed valve ports in the compressor inlet case covered at all times.

71.2.3 BUILDUP SEQUENCE. Refer to figures 71-1 thru 71-7. For buildup of a complete engine, these figures should be followed in sequence.

71.3 ENGINE MOUNTING. The main engine mounts are supported by a structure which extends down from the box beam. The front mounting bracket is attached to the front frame of the box beam. All engine mounts incorporate vibration isolators between the fittings attached to the engine and the supporting structure.

Engine side loads are absorbed by the left main mount, and the right main mount is free to move laterally as the engine expands.

CAUTION

Engine mounts are provided with isolators of either all metal or bonded rubber construction. It is not recommended that these two types of mount be mixed on any one engine installation.

71.4 RUBBER ENGINE MOUNT INSPECTION AND MAINTENANCE.

Note

Two different types of rubber main mount isolators have been supplied. The older type has a black phenolic inner bore and the later type has a metal inner bore. The phenolic bore isolators require a polished aluminum heat shield (figure 71-7, Item 6) used with a .040-inch thick phenolic shim. The metal bore isolators do not require a heat shield and are to be used with an .075-inch thick phenolic shim. Metal shims should not be used with rubber isolators.

a. The rubber parts may be cleaned by using a clean dry cloth. The metal parts can be cleaned by using a dry cleaning solvent, Stoddard Solvent or equivalent.

b. The most important check of the main mounts is the distance between the inner and the outer metal shells. The nominal distance is .090 inch with a minimum of .045 inch with no load on the mount. This check can be made by using a metal rod or drill of three sixty-fourth-inch diameter. This tool must be accepted at any point between the inner and outer shell (see figure 71-8 for this and other inspection requirements).

c. The forward mount need not be removed and replaced if the surface crack and the bond separation depth does not exceed the limits of figure 71-8. The dimensions at the center of the bonded bracket should be within the limits indicated on figure 71-8.

d. Refer to Lord Manufacturing Company Report No. 03326-5 for maintenance information on the older, phenolic bore mount, and to their Report No. 03326-6 for information on the newer metal bore mount.

71.5 METAL ENGINE MOUNT INSPECTION AND MAINTENANCE. (Aircraft 5060 and up-Optional on earlier aircraft).

a. The gap in the main engine mount should be measured as shown in Figure 71-9. Locate the point of greatest clearance between the OD of the washer and the ID of the ball flange (M2). Measure this clearance within an accuracy of .004, using a round wire gage, and record. In the same manner, measure the clearance 180° opposite (M1). Subtract the small clearance from the large. If the difference is greater than .055 inch, replace the assembly. No inspection for the lateral direction is required, since the mount will always wear out first in the vertical-thrust direction.

b. The forward mount should be checked to assure that the resilient material is not extruding out between the inner housing and the outer link, as shown in figure 71-9. Also, there should be no evidence that the inner housing has been in contact with the outer link. The part should be replaced if either condition is apparent.

c. The main engine mount requires a force of from 1 to 60 pounds for installation or removal from the trunnion mounting base.

d. Parts should be replaced if rusted, cracked, or plating is coming off.

e. Any parts of the metal engine mount removed in accordance with the above may be overhauled in accordance with Robintech Overhaul Manual TK1846-7.

71.6 MINOR REPAIRS FOR THE JETSTAR ENGINE TAILPIPE ASSEMBLIES. The JetStar tailpipe assemblies (part number JP1067-1 or -501 for use with JT12A-6, -6A engines and JP1067-502 for use with JT12A-8 engines) are all made of Inconel "X" material. It will be satisfactory to make minor repairs on any of these parts of fusion, spot or seam welding in accordance with the processes recommended for this material. Small patches of the same material may be fabricated and welded in place.

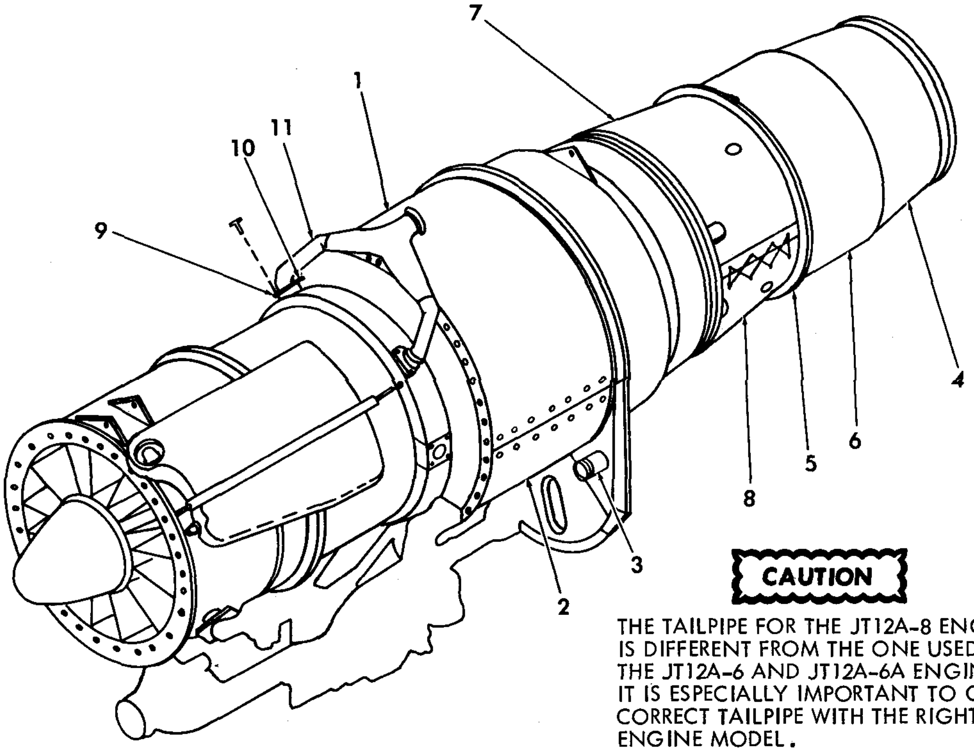
In repairing tailpipe assemblies, it would be well to remember the following points:

a. No repair should change the exhaust nozzle area.

b. The mating flange should not be modified in any way that will affect the clamping or sealing when the tailpipe is attached to the engine.

c. No repairs should be made that can cause undue turbulence in the exhaust stream.

d. Any repairs should maintain the structural integrity of the part and be capable of withstanding the exhaust pressure and velocity both during normal operation and while the engine is in reverse thrust.



CAUTION

THE TAILPIPE FOR THE JT12A-8 ENGINE IS DIFFERENT FROM THE ONE USED ON THE JT12A-6 AND JT12A-6A ENGINES. IT IS ESPECIALLY IMPORTANT TO GET THE CORRECT TAILPIPE WITH THE RIGHT ENGINE MODEL.

FIREWALL AND TAIL PIPE INSTALLATION

STEP 1

POSITION THE FIREWALL UPPER SHELL (1) AND THE LOWER SHELL (2) ON THE ENGINE, AND SECURE THEM WITH 30 EACH SCREWS AND WASHERS ON THE FORWARD EDGE OF THE RING.

STEP 2

CONNECT THE UPPER AND LOWER SHELLS BY INSTALLING 30 SCREWS AND 8 EACH SCREWS AND NUTS.

STEP 3

APPLY PRO-SEAL NO. 700 SEALANT COMPOUND (COAST PAINT AND CHEMICAL COMPANY) OR AN APPROVED EQUIVALENT SEALANT AROUND THE ENGINE AND FIREWALL RING TO CLOSE ALL OPENINGS BETWEEN THE ZONES SEPARATED BY THE FIREWALL. USE A 1/8-INCH BEAD OF SEALANT AT ALL LOCATIONS. SEALANT CURE TIME IS APPROXIMATELY 24 HOURS.

STEP 4

INSTALL THE THERMOCOUPLE HARNESS CONNECTOR (3) ON THE FIREWALL WITH FOUR EACH SCREWS, WASHERS, AND NUTS.

STEP 5

INSTALL THE TAILPIPE ASSEMBLY (4) ON THE ENGINE TURBINE EXHAUST CASE, ATTACHING IT WITH THE CLAMP ASSEMBLY (5). POSITION THE CLAMP BOLTS ON THE VERTICAL CENTERLINE OF THE ENGINE. TAP ALONG THE CLAMP AS THE BOLTS ARE TIGHTENED. TIGHTEN BOLTS ON PART NO. JP 1074-2 CLAMPS TO A TORQUE VALUE OF 80 INCH-POUNDS. BACK OFF BOLTS TO 40 INCH-POUNDS, AND DO NOT RETORQUE. TIGHTEN BOLTS ON ALTERNATE CLAMPS, PART NO. JP 1074-3, TO A TORQUE VALUE OF 160 INCH-POUNDS. BACK OFF BOLTS TO 50 INCH-POUNDS, AND THEN RETORQUE TO 85 INCH-POUNDS.

INSULATION BLANKET INSTALLATION

STEP 6

INSTALL THE TAILPIPE BLANKET (6) ON THE TAILPIPE (4) BY CROSS-LACING WITH 0.032 INCH DIAMETER INCONEL SAFETY WIRE. POSITION THE LACING ON THE RIGHT SIDE OF THE ENGINE, WITH THE CENTERLINE OF THE LACING APPROXIMATELY 45 DEGREES BELOW THE HORIZONTAL CENTERLINE OF THE ENGINE.

STEP 7

INSTALL THE UPPER ENGINE BLANKET (7) AND THE LOWER ENGINE BLANKET (8) AROUND THE TURBINE SECTION OF THE ENGINE BY CROSS-LACING WITH 0.032 INCH DIAMETER INCONEL SAFETY WIRE. POSITION THE LACING SYMMETRICALLY ON EACH SIDE OF THE ENGINE.

BLEED AIR MANIFOLD INSTALLATION

STEP 8

REMOVE THE TRIANGULAR SHIPPING COVER FROM EACH OF THE TWO BLEED AIR MANIFOLD PORTS (9) BY DISASSEMBLING THREE EACH WASHERS AND NUTS FROM EACH PORT.

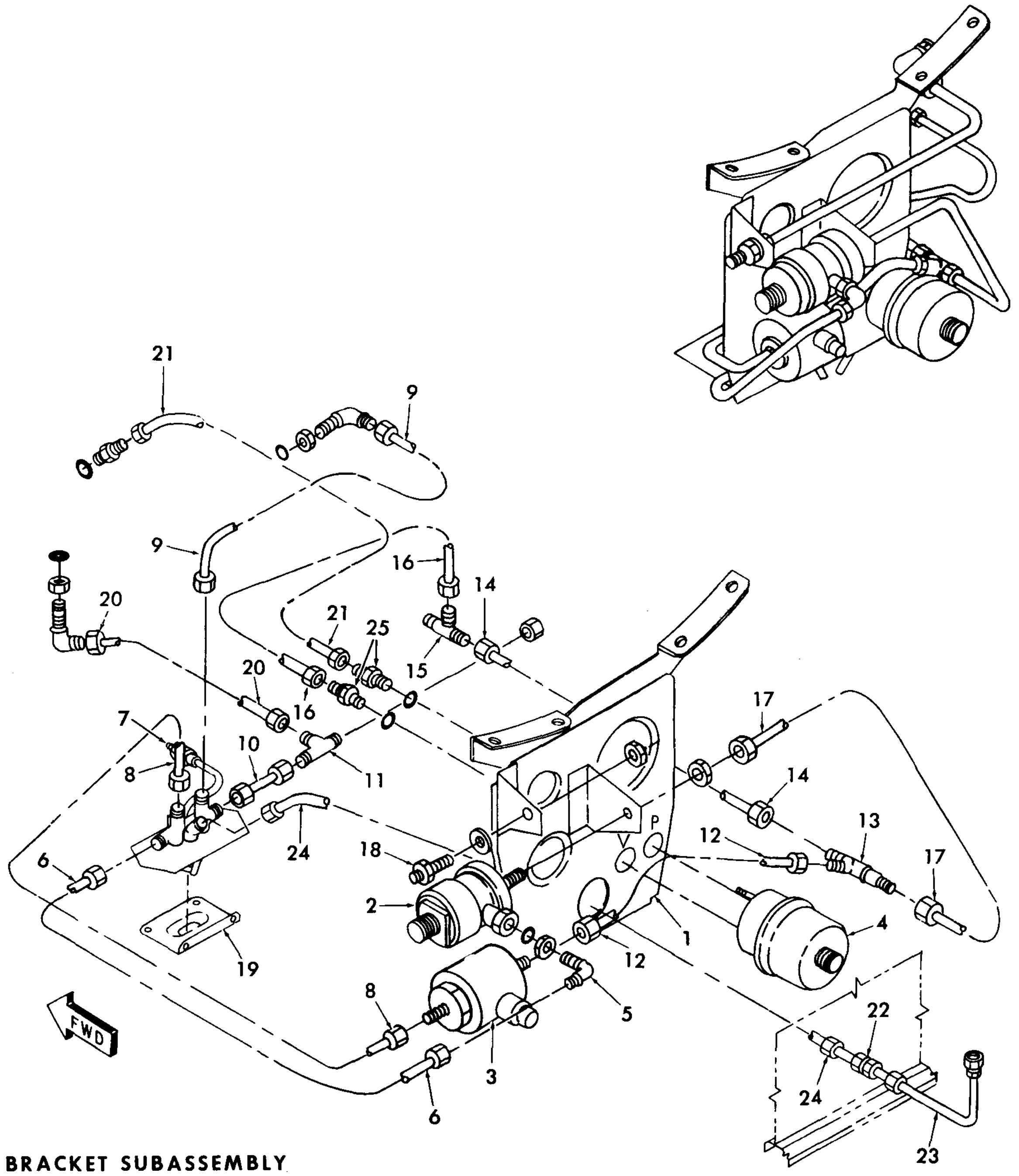
STEP 9

PLACE A GASKET (10), FURNISHED WITH THE ENGINE, ON EACH BLEED AIR PORT, AND INSTALL THE BLEED AIR MANIFOLD (11) ON THE PORTS, SECURING IT WITH THE BOLTS AND WASHERS PREVIOUSLY REMOVED FROM THE SHIPPING COVERS.

STEP 10

ATTACH THE BLEED AIR MANIFOLD TO THE SUPPORT BRACKET ON THE FIREWALL RING ASSEMBLY WITH ONE EACH BOLT, WASHER, AND NUT.

Figure 71-1. Engine Buildup—Firewall, Tailpipe, Blanket, and Manifold



DRAIN BRACKET SUBASSEMBLY

STEP 1

INSTALL THE FUEL PRESSURE SWITCH (2) SECURING IT WITH ONE NUT.

STEP 2

INSTALL THE FUEL LINE PURGE VALVE (3) WITH TWO EACH CLAMPS, SCREWS, AND WASHERS.

STEP 3

INSTALL THE FUEL DIFFERENTIAL PRESSURE SWITCH (4) WITH TWO EACH RESTRICTORS (25) PACKINGS, WASHERS, NUTS. MATCH THE HOLES IN THE DRAIN BRACKET MARKED "P" AND "V" WITH THE SIMILARLY MARKED SWITCH PORTS.

STEP 4

INSTALL THE FUEL PRESSURE SWITCH ELBOW (5) IN THE FUEL PRESSURE SWITCH (2) WITH ONE EACH NUT AND PACKING.

STEP 5

CONNECT THE TUBE ASSEMBLY (6) TO THE FUEL PRESSURE SWITCH (2). CONNECT THE OTHER END OF THE TUBE ASSEMBLY (6) TO THE DRAIN OUTLET ASSEMBLY (7).

STEP 6

CONNECT THE TUBE ASSEMBLY (8) BETWEEN THE FUEL LINE PURGE VALVE (3) AND THE DRAIN OUTLET ASSEMBLY (7).

Figure 71-2. Engine Buildup-Drain Bracket Subassembly (Engines Without Engine Drain Sump Modification)

STEP 7

CONNECT THE TUBE ASSEMBLY (9) TO THE DRAIN OUTLET ASSEMBLY (7) ON ENGINES IN COMPLIANCE WITH PWA SERVICE BULLETIN NO. 4178, THE INSTALLATION OF TUBE ASSEMBLY (9) IS OMITTED AND CAP PART NO. AN929-4 IS INSTALLED AT THE DRAIN OUTLET ASSEMBLY (7) IN THE POSITION WHERE THIS TUBE WOULD BE INSTALLED.

STEP 8

CONNECT THE TUBE ASSEMBLY (10) TO THE DRAIN OUTLET ASSEMBLY (7). CONNECT THE TEE FITTING (11) TO THE OTHER END OF THE TUBE ASSEMBLY (10).

STEP 9

CONNECT THE TUBE ASSEMBLY (12) TO THE FUEL LINE PURGE VALVE (3). CONNECT THE TEE FITTING (13) TO THE OTHER END OF THE TUBE ASSEMBLY (12).

STEP 10

CONNECT THE TUBE ASSEMBLY (14) TO THE TEE FITTING (13). CONNECT THE TEE FITTING (15) TO THE OTHER END OF THE TUBE ASSEMBLY (14).

STEP 11

INSTALL CAP ON THE END OF THE TEE FITTING (11) FACING TOWARDS THE RIGHT OF THE ENGINE.

STEP 12

CONNECT THE TUBE ASSEMBLY (16) BETWEEN THE RESTRICTOR (25) IN THE LEFT "V"PORT OF THE DIFFERENTIAL PRESSURE SWITCH (4) AND THE TEE FITTING (15).

STEP 13

CONNECT THE TUBE ASSEMBLY (17) BETWEEN THE FUEL PRESSURE SWITCH (2) AND THE TEE (13).

STEP 14

ON ENGINES BEING BUILT UP FOR NO. 3 POSITION ONLY, INSTALL THE UNION (18) IN THE HOLE IN THE ANGLE OF THE BRACKET ASSEMBLY (1), ABOVE THE FUEL PRESSURE SWITCH (2). SECURE THE UNION WITH ONE EACH WASHER AND NUT.

STEP 15

INSTALL THE BUILT-UP DRAIN BRACKET SUBASSEMBLY ON THE ENGINE WITH FOUR EACH SCREWS AND WASHERS.

STEP 16

INSTALL THE GUIDE (19) ON THE DRAIN BRACKET WITH FOUR EACH SCREWS AND WASHERS, AND TWO NUTS.

STEP 17

CONNECT THE DRAIN TUBE ASSEMBLY (20) BETWEEN THE TEE FITTING (11) AND THE ENGINE-DRIVEN FUEL PUMP DRAIN PORT, WITH ONE EACH ELBOW, NUT, AND PACKING AT THE DRAIN PORT.

STEP 18

CONNECT THE TUBE ASSEMBLY (21) BETWEEN THE RESTRICTOR (25) IN THE RIGHT ("P") PORT OF THE DIFFERENTIAL PRESSURE SWITCH (4) AND THE PORT ON THE ENGINE-DRIVEN PUMP, WITH ONE EACH UNION AND GASKET AT THE PUMP PORT.

STEP 19

CONNECT THE DRAIN TUBE ASSEMBLY (9) TO THE ENGINE FUEL PRESSURIZATION DUMP VALVE, WITH ONE EACH ELBOW, NUT, AND PACKING AT THE DUMP VALVE OUTLET PORT.

STEP 20

INSTALL A UNION (22) IN THE HOLE NEAR THE LOWER CORNER OF THE FIREWALL, SECURING IT WITH A NUT. CONNECT THE TUBE ASSEMBLY (23) BETWEEN THE UNION (22) AND THE COMBUSTION CHAMBER DRAIN PORT ON THE ENGINE. CONNECT THE TUBE ASSEMBLY (24) BETWEEN THE UNION (22) AND THE ANGLED CONNECTION ON THE DRAIN OUTLET ASSEMBLY (7).

NOTE

THIS FIGURE IS APPLICABLE FOR ENGINES WITHOUT DRAIN SUMP MODIFICATION. SEE FIGURE 71-2A FOR ENGINE BUILDUP INCORPORATING THE ENGINE DRAIN SUMP MODIFICATION.

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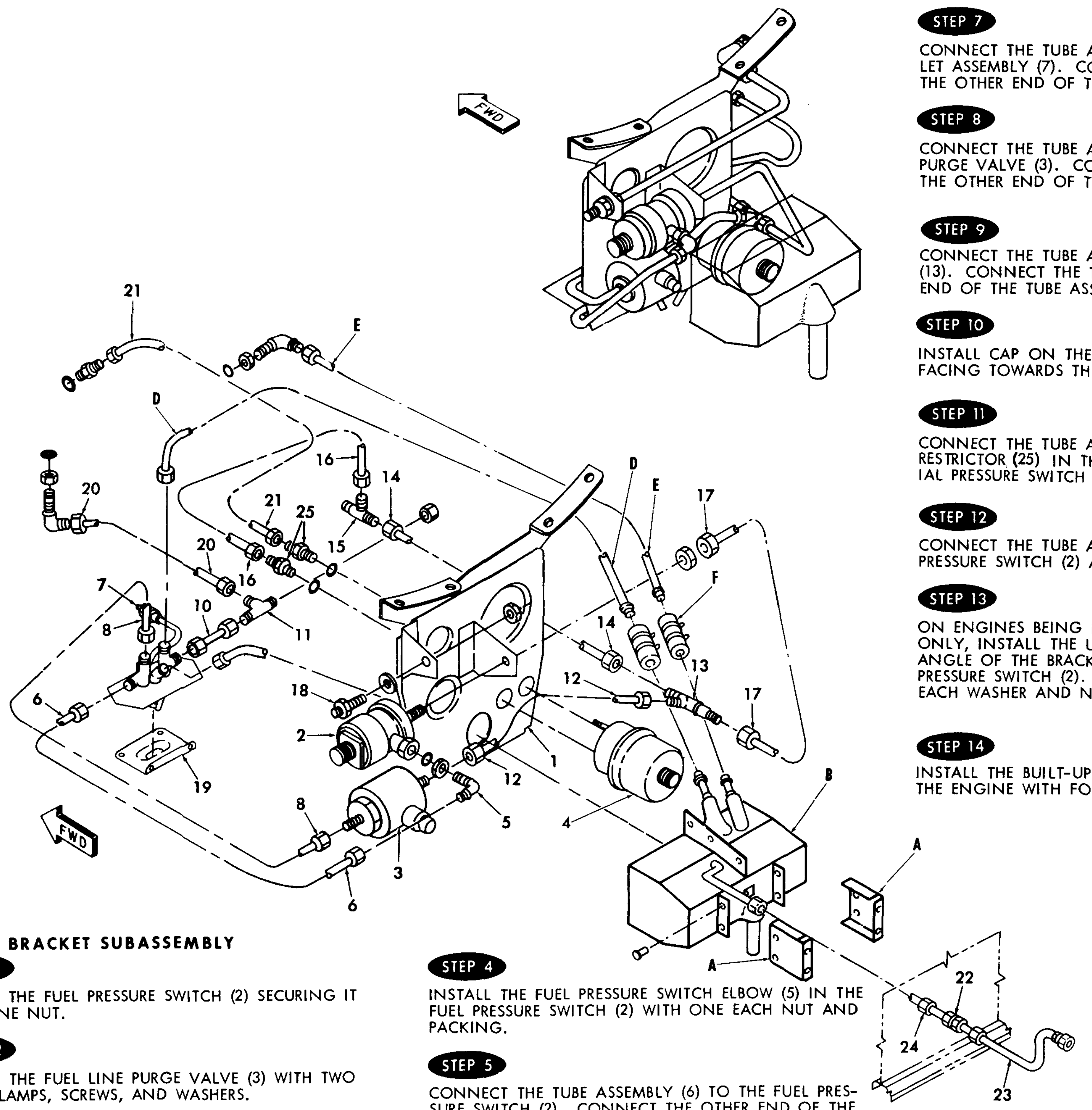


Figure 71-2A. Engine Buildup-Drain Bracket Subassembly (Engines With Engine Drain Sump Modification)

DRAIN BRACKET SUBASSEMBLY

STEP 1

INSTALL THE FUEL PRESSURE SWITCH (2) SECURING IT WITH ONE NUT.

STEP 2

INSTALL THE FUEL LINE PURGE VALVE (3) WITH TWO EACH CLAMPS, SCREWS, AND WASHERS.

STEP 3

INSTALL THE FUEL DIFFERENTIAL PRESSURE SWITCH (4) WITH TWO EACH RESTRICTORS (25) PACKINGS, WASHERS, AND NUTS. MATCH THE HOLES IN THE DRAIN BRACKET MARKED "P" AND "V" WITH THE SIMILARLY MARKED SWITCH PORTS.

STEP 4

INSTALL THE FUEL PRESSURE SWITCH ELBOW (5) IN THE FUEL PRESSURE SWITCH (2) WITH ONE EACH NUT AND PACKING.

STEP 5

CONNECT THE TUBE ASSEMBLY (6) TO THE FUEL PRESSURE SWITCH (2). CONNECT THE OTHER END OF THE TUBE ASSEMBLY (6) TO THE DRAIN OUTLET ASSEMBLY(7).

STEP 6

CONNECT THE TUBE ASSEMBLY (8) BETWEEN THE FUEL LINE PURGE VALVE (3) AND THE DRAIN OUTLET ASSEMBLY (7).

STEP 7

CONNECT THE TUBE ASSEMBLY (10) TO THE DRAIN OUTLET ASSEMBLY (7). CONNECT THE TEE FITTING (11) TO THE OTHER END OF THE TUBE ASSEMBLY (10).

STEP 8

CONNECT THE TUBE ASSEMBLY (12) TO THE FUEL LINE PURGE VALVE (3). CONNECT THE TEE FITTING (13) TO THE OTHER END OF THE TUBE ASSEMBLY (12).

STEP 9

CONNECT THE TUBE ASSEMBLY (14) TO THE TEE FITTING (13). CONNECT THE TEE FITTING (15) TO THE OTHER END OF THE TUBE ASSEMBLY (14).

STEP 10

INSTALL CAP ON THE END OF THE TEE FITTING (11) FACING TOWARDS THE RIGHT OF THE ENGINE.

STEP 11

CONNECT THE TUBE ASSEMBLY (16) BETWEEN THE RESTRICTOR (25) IN THE LEFT "V"PORT OF THE DIFFERENTIAL PRESSURE SWITCH (4) AND THE TEE FITTING (15).

STEP 12

CONNECT THE TUBE ASSEMBLY (17) BETWEEN THE FUEL PRESSURE SWITCH (2) AND THE TEE (13).

STEP 13

ON ENGINES BEING BUILT UP FOR NO. 3 POSITION ONLY, INSTALL THE UNION (18) IN THE HOLE IN THE ANGLE OF THE BRACKET ASSEMBLY (1), ABOVE THE FUEL PRESSURE SWITCH (2). SECURE THE UNION WITH ONE EACH WASHER AND NUT.

STEP 14

INSTALL THE BUILT-UP DRAIN BRACKET SUBASSEMBLY ON THE ENGINE WITH FOUR EACH SCREWS AND WASHERS.

STEP 15

INSTALL THE GUIDE (19) ON THE DRAIN BRACKET WITH FOUR EACH SCREWS AND WASHERS, AND TWO NUTS.

STEP 16

ATTACH BRACKETS (A) TO FIREWALL USING TWO BOLTS, NUTS AND WASHERS EACH.

STEP 17

INSTALL UNION (22) IN THE HOLE JUST BELOW THE CYLINDRICAL SECTION, SECURING IT WITH A NUT. CONNECT THE TUBE ASSEMBLY (23) BETWEEN THE UNION (22) AND THE COMBUSTION CHAMBER DRAIN PORT IN THE ENGINE.

STEP 18

INSTALL DRAIN TANK ASSEMBLY (B) TO BRACKETS (A) AND FIREWALL USING SEVEN BOLTS, NUTS AND WASHERS. MAKE CONNECTION WITH UNION (22).

STEP 19

INSTALL TUBE (D) BETWEEN THE DRAIN OUTLET ASSEMBLY (7) AND THE DRAIN TANK ASSEMBLY (B) WITH HOSE AND CLAMPS BETWEEN TUBE (D) AND TANK (B).

STEP 20

CONNECT THE DRAIN TUBE ASSEMBLY (20) BETWEEN THE TEE FITTING (11) AND THE ENGINE-DRIVEN FUEL PUMP DRAIN PORT, WITH ONE EACH ELBOW, NUT, AND PACKING AT THE DRAIN PORT.

STEP 21

CONNECT THE TUBE ASSEMBLY (21) BETWEEN THE RESTRICTOR (25) IN THE RIGHT ("P") PORT OF THE DIFFERENTIAL PRESSURE SWITCH (4) AND THE PORT ON THE ENGINE-DRIVEN PUMP, WITH ONE EACH UNION AND GASKET AT THE PUMP PORT.

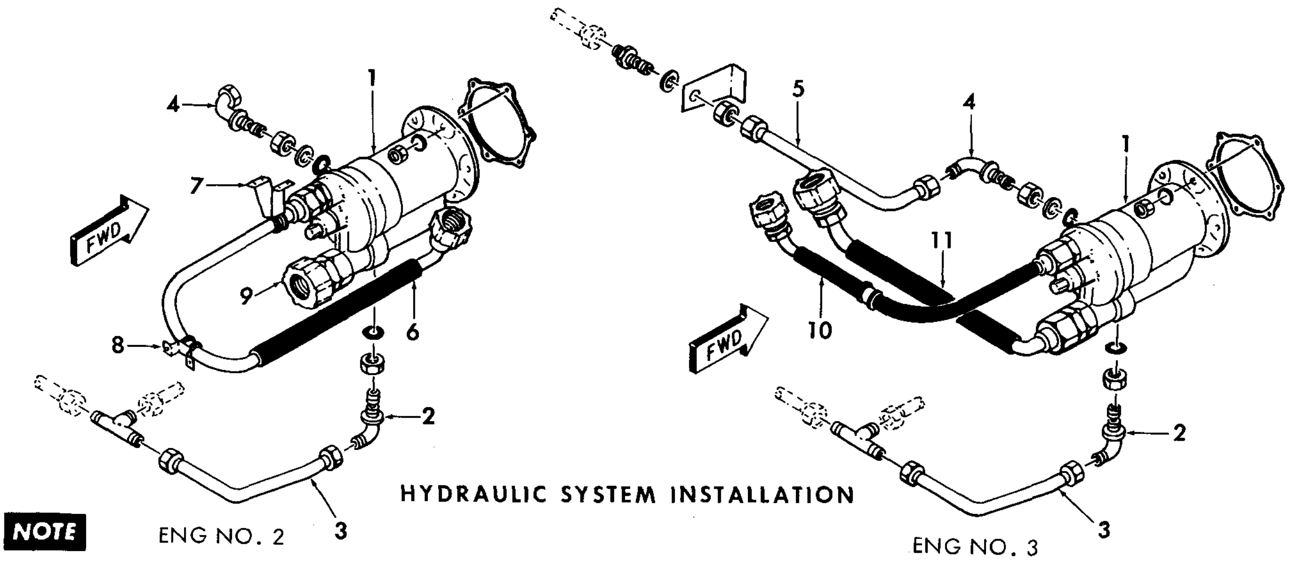
STEP 22

CONNECT THE DRAIN TUBE (E) BETWEEN THE ENGINE FUEL PRESSURIZATION DUMP VALVE AND THE DRAIN TANK ASSEMBLY (B). CONNECTION IS TO BE MADE AT THE DUMP VALVE OUTLET PART WITH ONE EACH ELBOW, NUT, AND PACKING. HOSE AND CLAMP (F) IS INSTALLED AT TANK END OF TUBE.

NOTE

THIS FIGURE IS APPLICABLE FOR ENGINES WITH ENGINE DRAIN SUMP MODIFICATION (JPK18). SEE FIGURE 71-2 FOR ENGINE BUILDUP WITHOUT THE ENGINE DRAIN SUMP MODIFICATION.

JS-1-2-X0/2-7102A



NOTE

1. THE HYDRAULIC SYSTEM IS INSTALLED ONLY ON ENGINES BEING PREPARED FOR POSITIONS NO. 2 AND NO. 3.
2. AIRCRAFT 5001-5055 WERE DELIVERED WITHOUT HYDRAULIC PUMP INSULATING GASKETS AND WASHERS. USE OF INSULATING GASKETS IS RECOMMENDED ON ALL AIRCRAFT WHEN PARTS BECOME AVAILABLE.

STEP 1

INSTALL THE HYDRAULIC PUMP (1) ON THE ENGINE PAD WITH SIX NUTS, SIX INSULATING WASHERS, AND ONE INSULATING GASKET. TIGHTEN THE NUTS TO A TORQUE VALUE BETWEEN 225 AND 300 INCH-POUNDS.

NOTE

COAT THE PUMP DRIVE SPLINE LIGHTLY WITH PLASTILUBE MOLY NO. 3 (WARREN REFINING CO., CLEVELAND, OHIO) PRIOR TO INSTALLATION.

STEP 2

INSTALL ONE EACH ELBOW (2) NUT, AND PACKING RING IN THE SHAFT SEAL DRAIN PORT OF THE HYDRAULIC PUMP. THE ELBOW SHOULD FACE AFT AND TOWARDS THE ENGINE CENTERLINE.

STEP 3

REMOVE CAP INSTALLED BY FIG 71-2 (STEP 11). CONNECT THE TUBE ASSEMBLY (3) BETWEEN THE ELBOW (2) AND THE TEE FITTING (11, FIGURE 71-2), PREVIOUSLY INSTALLED WITH THE DRAIN BRACKET SUBASSEMBLY.

STEP 4

INSTALL THE ELBOW (4) IN THE CASE DRAIN PORT OF THE HYDRAULIC PUMP, USING ONE EACH GASKET, PACKING, AND NUT.

STEP 5

ON NO. 3 ENGINE ONLY, CONNECT THE TUBE ASSEMBLY (5) BETWEEN THE ELBOW (4) AND THE UNION (18, FIG. 71-2) PREVIOUSLY INSTALLED WITH THE DRAIN BRACKET SUBASSEMBLY. CAP THE OPEN END OF THE UNION. (A HOSE WILL BE CONNECTED TO THE UNION ON ENGINE INSTALLATION.)

STEP 6

ON NO. 2 ENGINE ONLY, CAP THE OPEN END OF THE ELBOW (4). (A HOSE WILL BE CONNECTED TO THE ELBOW ON ENGINE INSTALLATION.)

STEP 7

ON NO. 2 ENGINE ONLY, CONNECT THE HOSE ASSEMBLY (6) TO THE HYDRAULIC PUMP WITH ONE EACH GASKET, RING, UNION, AND NUT. CONNECT THE COUPLING HALF TO THE FREE END OF THE HOSE ASSEMBLY UNTIL A SHARP INCREASE IN TIGHTENING TORQUE IS NOTED; THEN CONTINUE TIGHTENING FOR ONE-SIXTH TO ONE-THIRD OF A TURN MORE (ONE TO TWO FLATS ON THE HEXAGON NUT).

STEP 8

ON NO. 2 ENGINE ONLY, SUPPORT THE HOSE ASSEMBLY (6) WITH THE BRACKET (7) AND THE CLAMP (8).

STEP 9

ON NO. 2 ENGINE ONLY, INSTALL THE COUPLING HALF (9) ON THE HYDRAULIC PUMP WITH ONE EACH GASKET, RING, AND NUT.

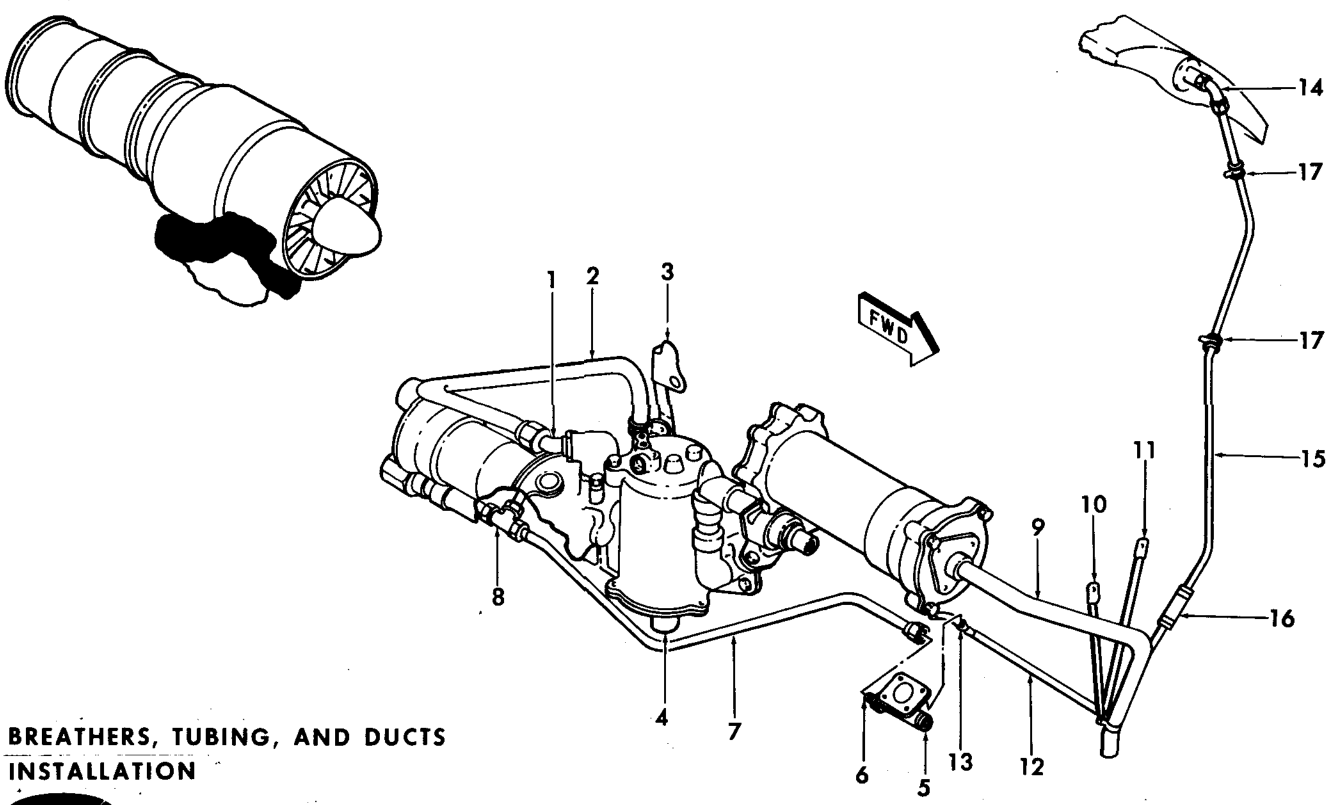
STEP 10

ON NO. 3 ENGINE ONLY, INSTALL THE HOSE ASSEMBLY (10) ON THE HYDRAULIC PUMP WITH ONE EACH GASKET AND REDUCER. INSTALL THE COUPLING HALF ON THE FREE END OF THE HOSE ASSEMBLY. TIGHTEN THE NUT ON EACH END OF THE HOSE ASSEMBLY UNTIL A SHARP INCREASE IN TIGHTENING TORQUE IS NOTED; THEN CONTINUE TIGHTENING FOR ONE-SIXTH TO ONE-THIRD OF A TURN MORE (ONE TO TWO FLATS ON THE HEXAGON NUT). SUPPORT THE HOSE ASSEMBLY WITH ONE CLAMP AT THE ENGINE CENTERLINE.

STEP 11

ON NO. 3 ENGINE ONLY, INSTALL THE HOSE ASSEMBLY (11) ON THE HYDRAULIC PUMP WITH ONE EACH GASKET AND REDUCER. INSTALL THE COUPLING HALF ON THE FREE END OF THE HOSE ASSEMBLY.

Figure 71-3. Engine Buildup—Hydraulic System Installation



BREATHERS, TUBING, AND DUCTS INSTALLATION

STEP 1

INSTALL THE BREATHER OUTLET ELBOW (1) IN THE ACCESSORY GEARBOX BREATHER PORT WITH ONE EACH GASKET AND NUT.

STEP 2

CONNECT THE BREATHER TUBE ASSEMBLY (2) TO THE ELBOW (1). ASSEMBLE THE BREATHER TUBE BRACKET (3) TO THE BREATHER TUBE NEAR ITS FREE END.

STEP 3

INSTALL THE FUEL FILTER DRAIN VALVE (4) IN THE BOTTOM COVER OF THE ENGINE FUEL FILTER WITH ONE PACKING RING.

STEP 4

INSTALL THE FUEL INLET FITTING (5) ON THE FUEL INLET PORT WITH ONE GASKET AND FOUR EACH WASHERS AND NUTS. DO NOT TIGHTEN THE NUTS.

STEP 5

INSTALL THE UNION (6) IN THE FUEL INLET FITTING (5) WITH ONE PACKING RING.

STEP 6

CONNECT THE DRAIN TUBE ASSEMBLY (7) BETWEEN THE UNION (6) AND THE TEE FITTING (8) AND (15, FIG 71-2) PREVIOUSLY INSTALLED WITH THE DRAIN BRACKET SUBASSEMBLY.

STEP 7

INSTALL THE FUEL HEATER EXHAUST DUCT ASSEMBLY (9) ON THE FUEL HEATER EXHAUST PORT WITH ONE GASKET AND THREE BOLTS.

STEP 8

ASSEMBLE THE FUEL HEATER EXHAUST DUCT SUPPORTS (10, 11, AND 12) TO THE FUEL HEATER EXHAUST DUCT (9) WITH ONE EACH CLAMP, SCREW, AND NUT.

STEP 9

INSTALL THE CLIP (13) ON THE END OF THE SUPPORT (12) WITH ONE EACH SCREW AND NUT.

STEP 10

REMOVE TWO OF THE ATTACHING NUTS FROM THE FUEL INLET FITTING (5) AND POSITION THE CLIP (13) ON THE TWO EXPOSED STUDS. TIGHTEN ALL FOUR ATTACHING NUTS TO A TORQUE VALUE BETWEEN 60 AND 85 INCH-POUNDS.

STEP 11

LOOSELY FASTEN THE SUPPORTS (10 AND 11) WITH TWO EACH BOLTS, WASHERS, AND NUTS. (THESE WILL LATER BE SECURED TO THE ENGINE AIR INLET DUCT FLANGE.)

STEP 12

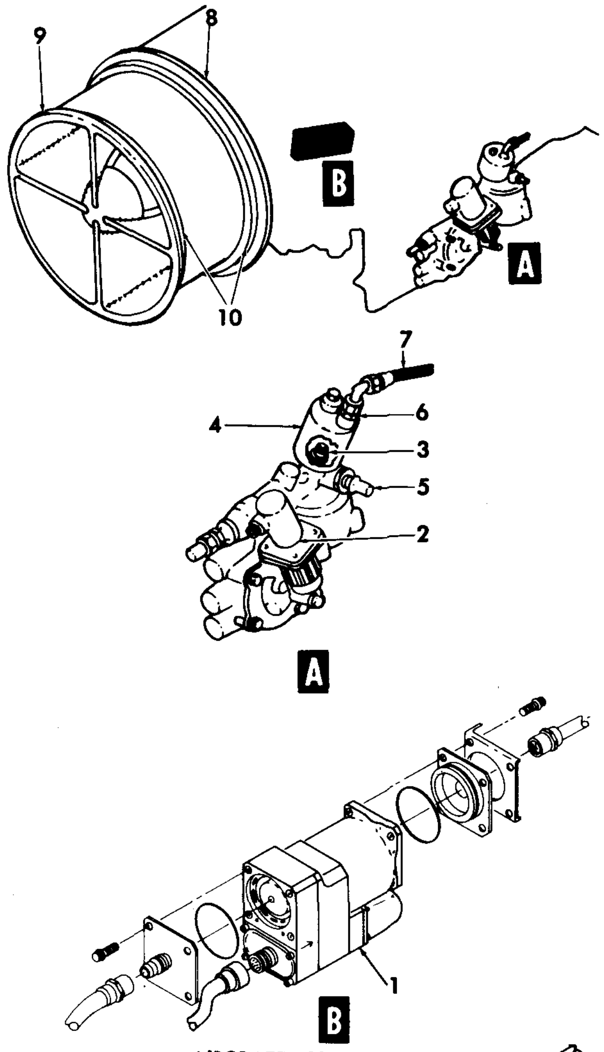
INSTALL THE ELBOW (14) IN THE OIL TANK BREATHER BOSS WITH ONE EACH GASKET AND NUT.

STEP 13

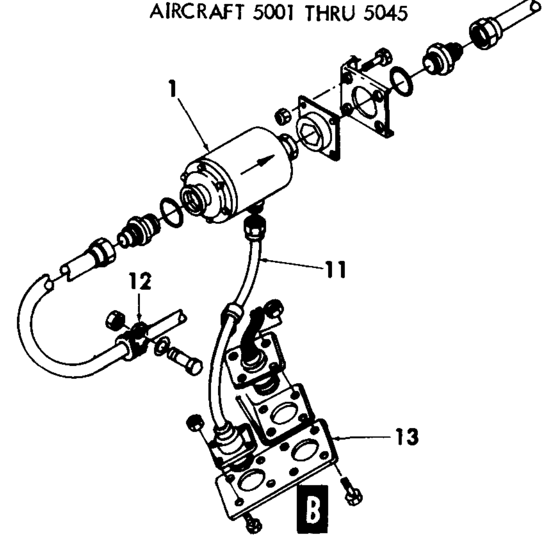
CONNECT THE OIL TANK BREATHER TUBE ASSEMBLY (15) BETWEEN THE ELBOW (14) AND THE WELDED CONNECTION ON THE FUEL HEATER EXHAUST DUCT (9) WITH THE HOSE (16). SECURE THE HOSE WITH TWO CLAMPS. (THE TUBE ASSEMBLY WILL LATER BE SECURED TO THE ENGINE AIR INLET DUCT FLANGE WITH TWO CLAMPS (17) WHICH MAY NOW BE INSTALLED ON THE TUBE ASSEMBLY WITH THE HARDWARE LEFT LOOSE.)

JS-2-2-7105

Figure 71-4. Engine Buildup—Breathers, Tubing and Ducts Installation



AIRCRAFT 5001 THRU 5045



AIRCRAFT 5046 AND UP (EXCEPT AS NOTED IN STEP 1)

INSTRUMENTATION AND AIR INLET DUCT INSTALLATION

STEP 1

INSTALL THE FUEL FLOW TRANSMITTER (1) WITH THE ADAPTERS FURNISHED WITH THE ENGINE. INSTALL DISCONNECT HARNESS (11). CLAMP TO ENGINE FUEL TUBE (12) WITH A PLASTIC CLAMP AROUND THE HARNESS AND A WASHER BETWEEN IT AND THE SCREW HEAD. ON AIRCRAFT 5086 AND UP; INSTALL BRACKET (13) USING THE ENGINE CONNECTOR MOUNT SCREWS AND NUTS AND SECURE DISCONNECT HARNESS TO BRACKET.

STEP 2

INSTALL THE TACHOMETER GENERATOR (2) ON THE ENGINE DRIVE PAD WITH ONE GASKET AND FOUR EACH WASHERS AND NUTS. TIGHTEN THE NUTS TO A TORQUE VALUE BETWEEN 65 AND 85 INCH-POUNDS.

NOTE

COAT THE TACHOMETER GENERATOR DRIVE SHAFT LIGHTLY WITH PLASTILUBE MOLY NO. 3 (WARREN REFINING CO., CLEVELAND OHIO) PRIOR TO INSTALLATION.

STEP 3

INSTALL THE RESTRICTOR (3) IN THE PRESSURE PORT OF THE OIL PRESSURE TRANSMITTER (4). TIGHTEN THE RESTRICTOR TO RETAIN IT SECURELY IN THE TRANSMITTER. RESTRICTORS INSTALLED ON TRANSMITTERS MADE BY COURTER, INC. SHOULD BE TIGHTENED SO THAT THEY PROTRUDE EXACTLY 0.10 INCH PLUS OR MINUS 0.01 INCH. THIS REQUIREMENT WILL PREVENT DAMAGE TO THE TRANSMITTER AND WILL ASSURE ADEQUATE ENGAGEMENT OF THE NYLON LOCKING PLUG TO SECURE THE RESTRICTOR IN THE TRANSMITTER.

STEP 4

INSTALL THE OIL PRESSURE TRANSMITTER (4) IN THE ENGINE OIL PRESSURE PORT WITH ONE SEALING GASKET.

STEP 5

INSTALL THE OIL TEMPERATURE BULB (5) IN THE ENGINE PORT ADJACENT TO THE OIL PRESSURE TRANSMITTER, WITH ONE EACH ADAPTER (FURNISHED WITH THE ENGINE) AND GASKET.

STEP 6

INSTALL A UNION (6) WITH A GASKET IN BOTH THE VENT PORT OF THE OIL PRESSURE TRANSMITTER (4) AND IN THE ACCESSORY GEARBOX OIL PRESSURE TRANSMITTER VENT PORT WHICH FACES AFT BELOW THE OIL PRESSURE TRANSMITTER.

STEP 7

CONNECT THE OIL PRESSURE TRANSMITTER VENT HOSE ASSEMBLY (7) BETWEEN THE UNIONS INSTALLED IN STEP 6. SUPPORT THE HOSE ASSEMBLY WITH ONE CLAMP ATTACHED TO THE NEAREST SCREW IN THE FIREWALL LOWER SHELL.

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Figure 71-5. Engine Buildup-Instrumentation and Air Inlet Duct Installation (Sheet 1 of 2)

STEP 8

ATTACH THE AFT INLET DUCT FLANGE (8) TO THE ENGINE COMPRESSOR CASE FORWARD FLANGE WITH 31 BOLTS, 62 WASHERS, AND 31 NUTS. NOTE THAT THE FUEL HEATER EXHAUST DUCT SUPPORTS (10 AND 11, FIGURE 71-4) SHOULD BE PICKED UP AT TWO PLACES ON THE LOWER PORTION OF THE COMPRESSOR INLET CASE FORWARD FLANGE. TWO CLAMPS (17, FIGURE 71-4) SUPPORTING THE OIL TANK BREATHER TUBE ASSEMBLY (15, FIGURE 71-4) SHOULD SIMILARLY BE PICKED UP ON THE LEFT SIDE OF THE COMPRESSOR INLET CASE FORWARD FLANGE.

STEP 9

ATTACH THE AFT INLET DUCT (9) TO THE DUCT FLANGE (8) WITH ONE CLAMP (10). SNUG UP CLAMP BUT DO NOT TIGHTEN SO THAT DUCT POSITION CAN BE ADJUSTED ON INSTALLATION IN THE NACELLE.

STEP 10

LOOSELY ASSEMBLE A SIMILAR CLAMP TO THE FORWARD FLANGE OF THE AFT INLET DUCT (9). THIS CLAMP WILL BE USED TO CONNECT THE AFT AND FORWARD INLET DUCTS ON ENGINE INSTALLATION.

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Figure 71-5. Engine Buildup-Instrumentation and Air Inlet Duct Installation (Sheet 2 of 2)

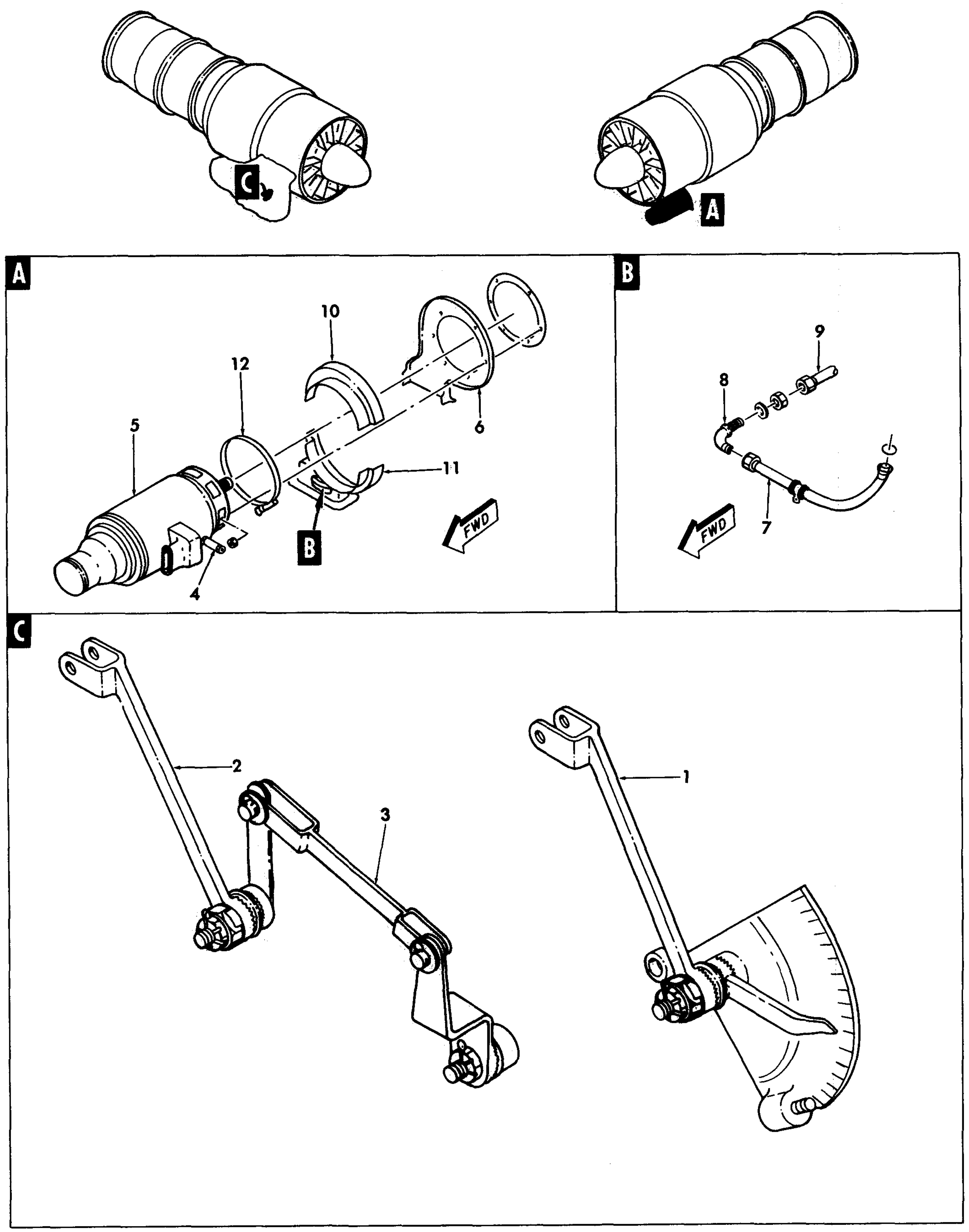

Figure 71-6. Engine Buildup—Engine Controls and Electrical Installation

ENGINE CONTROLS AND ELECTRICAL INSTALLATION

STEP 1

ON ENGINES WITH HOLLEY FUEL CONTROLS ONLY, INSTALL THE BELLCRANK (1) ON THE FUEL CONTROL SHAFT WITH ONE EACH WASHER, NUT, AND COTTER PIN.

NOTE

DIFFERENT BELLCRANKS ARE USED WITH HOLLEY & HAM STANDARD FUEL CONTROLS. REFER TO THE ILLUSTRATED PARTS BREAKDOWN.

STEP 2

ON ENGINES WITH HAMILTON STANDARD FUEL CONTROLS ONLY, INSTALL THE BELLCRANK (2) ON THE FUEL CONTROL IDLER SHAFT WITH ONE EACH TAB WASHER AND 1/2-INCH NUT. TIGHTEN THE NUT TO A MAXIMUM TORQUE OF 50 INCH-POUNDS, AND BEND THE TAB WASHER TO SECURE IT. INSTALL ONE EACH 1/4-INCH SLOTTED NUT AND 1/4-INCH THIN CORROSION RESISTANT STEEL WASHER ON THE END OF THE IDLER SHAFT, PERMITTING 0.0015 TO 0.010 INCH END PLAY OF THE BELLCRANKS, AND SECURE WITH A COTTER PIN. ATTACH THE LINK (3) TO THE ARM WITH ONE EACH PIN, WASHER, AND COTTER PIN.

STEP 3

REMOVE THE OVERHEAT DETECTOR (4) FROM THE STARTER GENERATOR (5). IF JUMPER AND TERMINAL BLOCK IS PROVIDED FOR THE OVERHEAT DETECTOR ON THE GENERATOR, THE WIRE SHOULD BE DISCONNECTED AT THE TERMINAL BLOCK RATHER THAN THE HEAT DETECTOR.

STEP 4

POSITION A GASKET ON THE STARTER GENERATOR PAD, AND POSITION A SHROUD PLATE (6) ON THE GASKET. INSTALL THE STARTER GENERATOR (5) WITH SIX NUTS, TIGHTENING TO A TORQUE VALUE BETWEEN 225 AND 300 INCH-POUNDS.

NOTE

COAT THE STARTER GENERATOR DRIVE SPLINE WITH MIL-G-21164 OR MIL-G-81322 GREASE PRIOR TO INSTALLATION.

STEP 5

REINSTALL THE OVERHEAT DETECTOR (4) IN THE STARTER GENERATOR, TIGHTENING TO A TORQUE VALUE BETWEEN 50 AND 30 INCH-POUNDS.

CAUTION

EXCESSIVE TORQUE ON THE TERMINAL STUD NUT OF THE GENERATOR OVERHEAT DETECTOR CAN CAUSE SERIOUS DAMAGE TO THE DETECTOR. THE TORQUE USED ON THIS NUT SHOULD BE FROM 25 TO 30 INCH-POUNDS WITH SPECIAL CARE NEVER TO EXCEED THE MAXIMUM.

NOTE

STEPS 6 AND 7 ARE APPLICABLE FOR ENGINES HAVING LEAR SIEGLER GENERATORS INSTALLED.

STEP 6

CEMENT AN O-RING BETWEEN THE WELDED, BELLMOUTH SLEEVE ON THE DRAIN TUBE ASSEMBLY (7) AND THE TUBE WALL, USING GENERAL-PURPOSE CHLOROPRENE CEMENT, LOCKHEED NO. 30-783, OR APPROVED EQUIVALENT. FIRST CLEAN THE METAL WITH ALIPHATIC NAPHTHA, AND ROUGHEN THE SURFACE OF THE O-RING. APPLY CEMENT TO BOTH PARTS, WAIT UNTIL IT BECOMES TACKY, AND THEN POSITION THE O-RING IN THE RECESS BETWEEN THE SLEEVE AND THE TUBE.

STEP 7

CLEAN THE AREA AROUND THE DRAIN HOLE IN THE BOTTOM OF THE STARTER GENERATOR CASING WITH ALIPHATIC NAPHTHA. APPLY GENERAL-PURPOSE CHLOROPRENE CEMENT, LOCKHEED NO. 30-783, OR APPROVED EQUIVALENT, TO THIS AREA AND TO THE O-RING PREVIOUSLY ATTACHED TO THE TUBE ASSEMBLY (7).

STEP 8

INSTALL THE ELBOW (8) IN THE HOLE IN THE SHROUD PLATE (6) WITH ONE EACH WASHER AND NUT.

STEP 9

ON ENGINES HAVING LEAR SIEGLER GENERATORS INSTALLED, CONNECT THE DRAIN TUBE ASSEMBLY (7) BETWEEN THE HOLE IN THE GENERATOR CASING AND ELBOW (8), MAKING SURE THAT THE O-RING IS IN GOOD ADHESIVE CONTACT WITH THE GENERATOR CASING. IF A BENDIX GENERATOR IS USED, INSTALL A CAP ON ELBOW AND DELETE TUBE ASSEMBLY (7).

STEP 10

ON ENGINES HAVING LEAR SIEGLER GENERATORS INSTALLED, SECURE THE TUBE ASSEMBLY (7) TO THE SHROUD PLATE (6) WITH ONE EACH CLAMP, SCREW AND WASHER. ON ENGINES HAVING BENDIX GENERATORS INSTALLED CLOSE UP HOLE WITH SCREW AND WASHER.

STEP 11

CONNECT THE TUBE ASSEMBLY (9) BETWEEN THE ELBOW (8) AND THE WELDED UNION ON THE DRAIN OUTLET ASSEMBLY TUBE (7, FIG 71-2).

STEP 12

POSITION THE SHROUD UPPER HALF (10) AND THE SHROUD LOWER HALF (11) ON THE STARTER GENERATOR. ASSEMBLE THE SHROUD HALVES TOGETHER WITH ONE CLAMP (12). INSTALL FIVE EACH SCREWS AND WASHERS, AND TWO NUTS.

STEP 13

SEAL AROUND THE SHROUD ASSEMBLY WITH PRO-SEAL NO. 700 SEALANT COMPOUND (COAST PAINT AND CHEMICAL COMPANY), OR AN APPROVED EQUIVALENT, TO CLOSE ALL GAPS BETWEEN MATING PARTS. APPLY THE SEALANT WITH AN APPROXIMATELY 1/8-INCH BEAD.

NOTE

CURE TIME FOR THE SEALANT IS APPROXIMATELY 24 HOURS.

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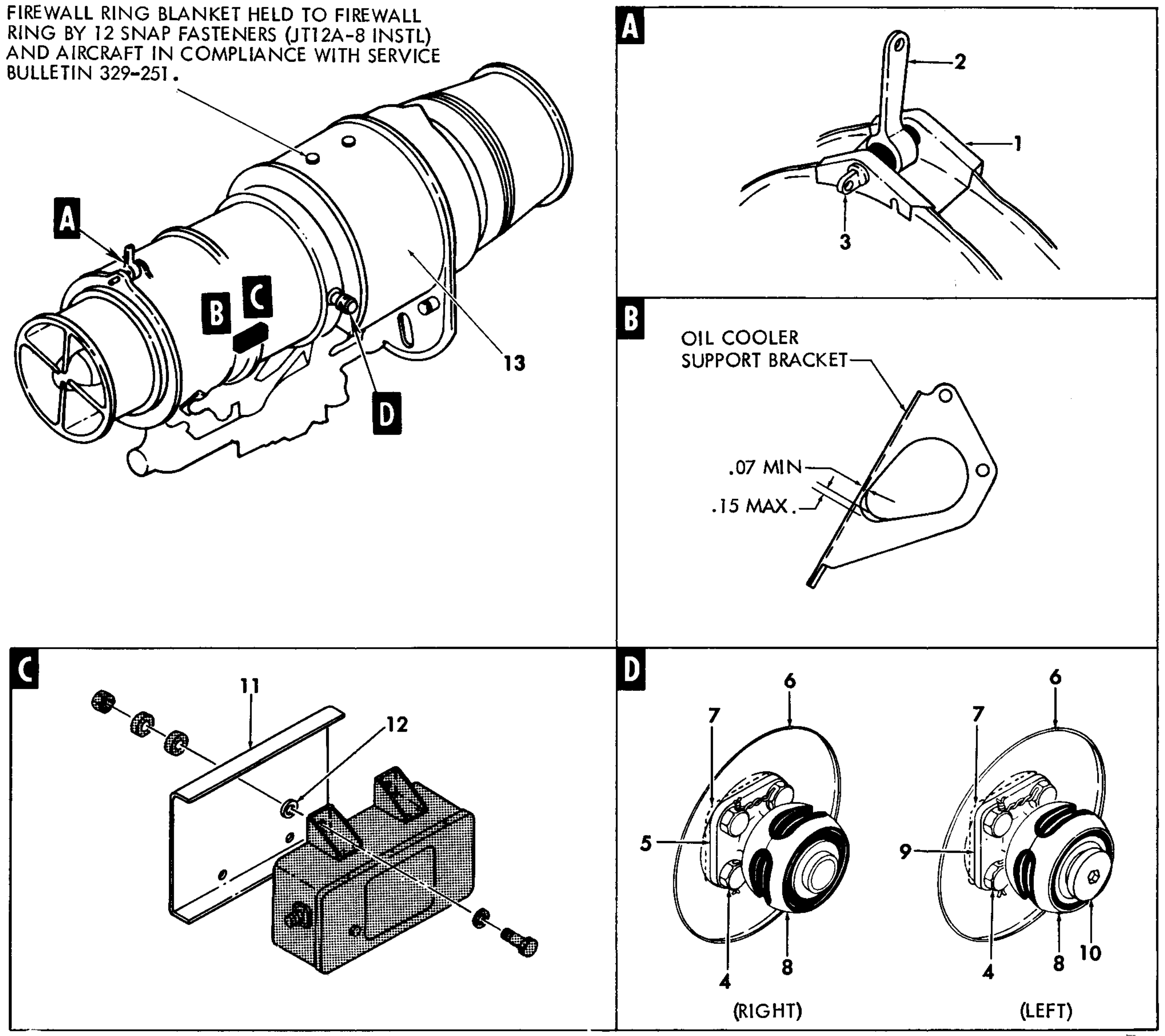

FIREWALL RING BLANKET HELD TO FIREWALL RING BY 12 SNAP FASTENERS (JT12A-8 INSTL) AND AIRCRAFT IN COMPLIANCE WITH SERVICE BULLETIN 329-251.
A
B
C
D
13
A
2
1
3
B
OIL COOLER SUPPORT BRACKET
.07 MIN
.15 MAX
C
11
12
D
6
7
6
7
5
9
4
8
4
8
10
(RIGHT)
(LEFT)

Figure 71-7. Engine Buildup - Engine Mounts and Ignition Exciter Shields Installations

ENGINE MOUNTS AND IGNITION EXCITER HEAT SHIELDS INSTALLATION

STEP 1

POSITION THE COVER ASSEMBLY (1) ON THE FORWARD ENGINE MOUNT LUGS AT UPPER CENTER OF THE COMPRESSOR INLET CASING.

NOTE

ENGINE MOUNTS ARE PROVIDED WITH BOTH ALL-METAL AND BONDED RUBBER CONSTRUCTION. IT IS NOT RECOMMENDED THAT THESE TWO TYPES BE MIXED ON ANY ONE ENGINE INSTALLATION.

STEP 2

POSITION THE FORWARD ENGINE MOUNT (2) BETWEEN THE ENGINE MOUNT LUGS AND IN THE WELL FORMED BY THE COVER (1). PLACE A WASHER UNDER THE HEAD OF THE EYEBOLT (3), AND PASS THE EYEBOLT, WITH ITS HEAD FORWARD, THROUGH THE ENGINE MOUNT LUGS, THE COVER, AND THE FORWARD ENGINE MOUNT. SECURE THE EYEBOLT WITH ONE EACH WASHER, KNURLED NUT, AND COTTER PIN. TIGHTEN THE KNURLED NUT FINGER-TIGHT BEFORE INSTALLING THE COTTER PIN.

NOTE

STEPS (3) THROUGH (12) SHOULD BE ACCOMPLISHED WHILE THE ENGINE IS SUPPORTED BY THE ENGINE HOISTING SLING, JUST BEFORE INSTALLATION IN THE AIRPLANE. THE ENGINE MAINTENANCE AND TRANSPORTATION DOLLY UTILIZES THE ENGINE DIFFUSER CASE MOUNT PADS TO SUPPORT THE ENGINE; THEREFORE THE MAIN ENGINE MOUNTS CANNOT BE INSTALLED WITH THE ENGINE IN THE DOLLY.

STEP 3

INSERT FOUR SPECIAL BOLTS (4) IN THE HOLES IN THE TRUNNION FLANGE OF THE SLIDING ENGINE MOUNT (5). POSITION THE HEAT SHIELD (6) AND THE SHIM (7) ON THE FOUR BOLTS.

NOTE

WHEN THE HEAT SHIELD (6) IS USED, THE SHIM (7) SHOULD BE MADE OF .040 INCH THICK PHENOLIC MATERIAL. THIS SHIELD IS REQUIRED ONLY WHEN ISOLATORS (8) WITH A BLACK PHENOLIC BORE ARE USED. THE HEAT SHIELD IS NOT REQUIRED WITH AN ALL METAL ISOLATOR OR WITH A RUBBER ISOLATOR WITH A METAL BORE. WHEN THE HEAT SHIELD IS NOT USED, A .075 INCH THICK PHENOLIC SHIM (7) SHOULD BE USED. IN CASE AN ALL METAL ISOLATOR IS USED, THIS .075 INCH THICK SHIM MAY BE MADE OF METAL.

NOTE

THE SPECIAL BOLTS (QUANTITY 4) SHOULD HAVE FEL PRO C-5 THREAD COMPOUND APPLIED ON INSTALLATION. (FELT PRODUCTS MFG. CO. SKOKIE, ILLINOIS.)

STEP 4

INSTALL THE SLIDING MOUNT TRUNNION, HEAT SHIELD (IF REQUIRED), AND SHIM SUBASSEMBLY ON THE RIGHT ENGINE MOUNT PAD WITH THE SPECIAL BOLTS (4). TIGHTEN THE BOLTS TO A TORQUE VALUE BETWEEN 100 AND 140 INCH-POUNDS. LOCK THE BOLTS IN PAIRS WITH 0.032 INCH DIAMETER CORROSION RESTRAINTS STEEL LOCKWIRE.

STEP 5

APPLY A LIGHT COATING OF MIL-G-21164 MOLYBDENUM DISULFIDE GREASE BETWEEN THE ENGINE MOUNT TRUNIONS AND THE MATING OR SLIDING SURFACES OF THE ENGINE MOUNTS.

NOTE

MIL-G-21164 GREASE MANUFACTURER INFORMATION

ROYCO 64	ROYAL LUBRICANTS CO. RIVER ROAD HANOVER, NEW JERSEY
SG 1027	SHELL OIL CO. 50 WEST 50TH ST. NEW YORK, N.Y.

STEP 6

SLIDE THE ISOLATOR (8) ON THE TRUNNION POST.

STEP 7

INSERT THE FOUR SPECIAL BOLTS (4) IN THE FLANGE HOLES OF THE FIXED ENGINE MOUNT TRUNNION (9). POSITION THE HEAT SHIELD (IF REQUIRED) (6) AND THE SHIM (7) ON THE BOLTS. (SEE NOTE FOLLOWING STEP 3.)

STEP 8

INSTALL THE FIXED ENGINE MOUNT TRUNNION (9) ON THE LEFT ENGINE MOUNT PAD WITH THE FOUR SPECIAL BOLTS (4). TIGHTEN THE BOLTS TO A TORQUE VALUE BETWEEN 100 AND 140 INCH-POUNDS. LOCK THE BOLTS IN PAIRS WITH 0.032 INCH DIAMETER CORROSION-RESISTANT STEEL LOCKWIRE.

STEP 9

LUBRICATE TRUNNION AS PER STEP 5.

STEP 10

SLIDE THE ISOLATOR (8) ON THE TRUNNION POST.

STEP 11

INSTALL THE SPECIAL THROUGH-BOLT (10) IN THE TRUNNION POST AND TIGHTEN IT TO A TORQUE VALUE BETWEEN 50 AND 70 INCH-POUNDS.

STEP 12

APPLY A LIGHT COATING OF MIL-L-6086 OIL OR ANY OTHER PETROLEUM - BASE OIL TO THE RUBBER PARTS. NOT REQUIRED FOR ALL - METAL MOUNTS.

NOTE

STEPS 13 AND 14 ARE NOT REQUIRED ON ENGINES THAT ARE EQUIPPED WITH PRATT AND WHITNEY FURNISHED IGNITION EXCITER HEAT SHIELDS. (THIS CHANGE WAS ACCOMPLISHED IN PRODUCTION ON AIRCRAFT 5084 AND UP.) REF. PRATT AND WHITNEY SERVICE BULLETIN 1248.

STEP 13

TEMPORARILY REMOVE BOLT IGNITION EXCITER BOXES BY DISASSEMBLING THE ATTACHING HARDWARE, FURNISHED WITH THE ENGINE.

STEP 14

POSITION A HEAT SHIELD (11) ON EACH EXCITER BOX MOUNTING, AND REINSTALL THE BOXES WITH AN ASBESTOS WASHER (12) UNDER EACH BOX MOUNTING FOOT, USING THE HARDWARE PREVIOUSLY REMOVED.

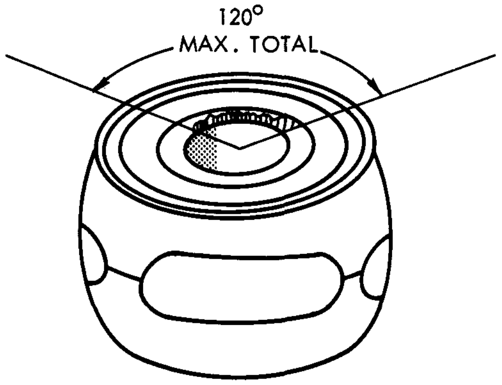
NOTE

ADDITION OF THE HEAT SHIELD AND THE WASHERS UNDER THE LEFT EXCITER BOX MOUNTING FEET MAY CAUSE INTERFERENCE BETWEEN THE IGNITION HARNESS AND THE CUTOUT IN THE OIL COOLER SUPPORT BRACKET THROUGH WHICH THE HARNESS IS ROUTED. IF THIS OCCURS, REWORK THE CUTOUT AS SHOWN IN DETAIL "B", AND ADD ONE 1/4-INCH CORROSION-RESISTANT STEEL WASHER BETWEEN THE OIL LINE FITTING AND THE BRACKET.

STEP 15

ON THE JT12A-8 ENGINE OR AIRCRAFT IN COMPLIANCE WITH SERVICE BULLETIN 329-251, INSTALL ENGINE FIREWALL RING INSULATION BLANKET (13). HOLES MUST BE PIERCED IN THE BLANKET FOR THE BREATHER TUBE CLAMP (FIG 71-4), THE HYDRAULIC BLANKET (FIG 71-3) AND OIL PRESSURE TRANSMITTER VENT HOSE CLAMP (FIG 71-5).

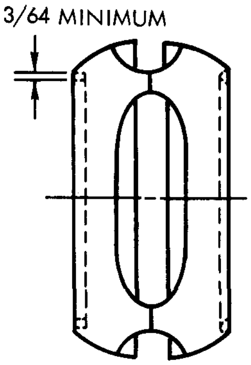
RUBBER MAIN MOUNT ISOLATOR



DO NOT REPLACE THE LM-405 ISOLATOR FOR SMALL CRACKS OR CHIPS (UP TO 1/3 OF TOTAL END AREA) IN THE BLACK PHENOLIC. BORE. UNBROKEN SUPPORT FOR LM-405-6 BOLT HEAD IS REQUIRED FOR 2/3 OF PERIPHERY. REPLACE MOUNTING IF PHENOLIC IS LOOSE IN THE BORE.

NOTE

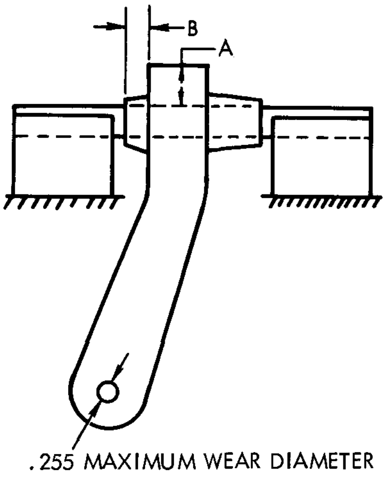
LATER RUBBER ISOLATORS HAVE THE BLACK PHENOLIC BORE REPLACED WITH A METAL BORE.



MUST ACCEPT 3/64 DIAMETER ROD AT ANY POINT EITHER END WHEN ISOLATOR IS NOT SUPPORTING ENGINE WEIGHT.

MAXIMUM WEAR DIAMETER ON METAL BORE OF LM-405-52 ISOLATOR IS .753 INCH. MINIMUM WEAR DIAMETER ON TRUNNION IS .747 INCH.

RUBBER FORWARD MOUNT ISOLATOR

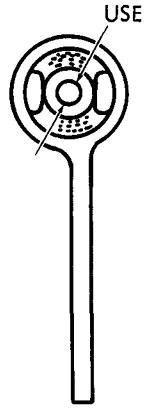


.255 MAXIMUM WEAR DIAMETER

CHECK DIMENSION AT CENTER OF BONDED BRACKET WHEN ISOLATOR IS NOT SUPPORTING ENGINE WEIGHT.

DIMENSION A		DIMENSION B	
MAX.	MIN.	MAX.	MIN.
0.61	0.38	0.38	0.18

.346 MAXIMUM WEAR DIAMETER. MINIMUM WEAR DIA OF EYE BOLT USED HERE IS .335 INCH.



NOTE

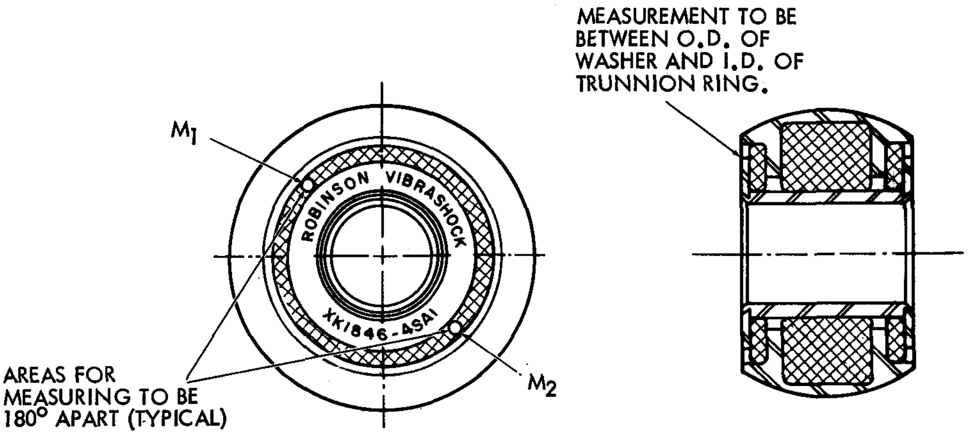
GENERAL NOTES APPLICABLE TO ALL BONDED RUBBER MOUNT PARTS:

1. SURFACE CRACKS OR BOND SEPARATION OF THE RUBBER ARE NOT CAUSE FOR REPLACEMENT UNLESS .18 INCH OR MORE DEEP.
2. REPLACE PARTS WORN IN EXCESS OF MAXIMUM OR MINIMUM WEAR DIAMETERS.
3. SOME EXCESSIVELY WORN METAL PARTS MAY BE SALVAGED BY REPLATING.

JS-1-2-X0/8-7108

Figure 71-8. Rubber Engine Mounts Inspection Data

METAL MAIN MOUNT ISOLATOR

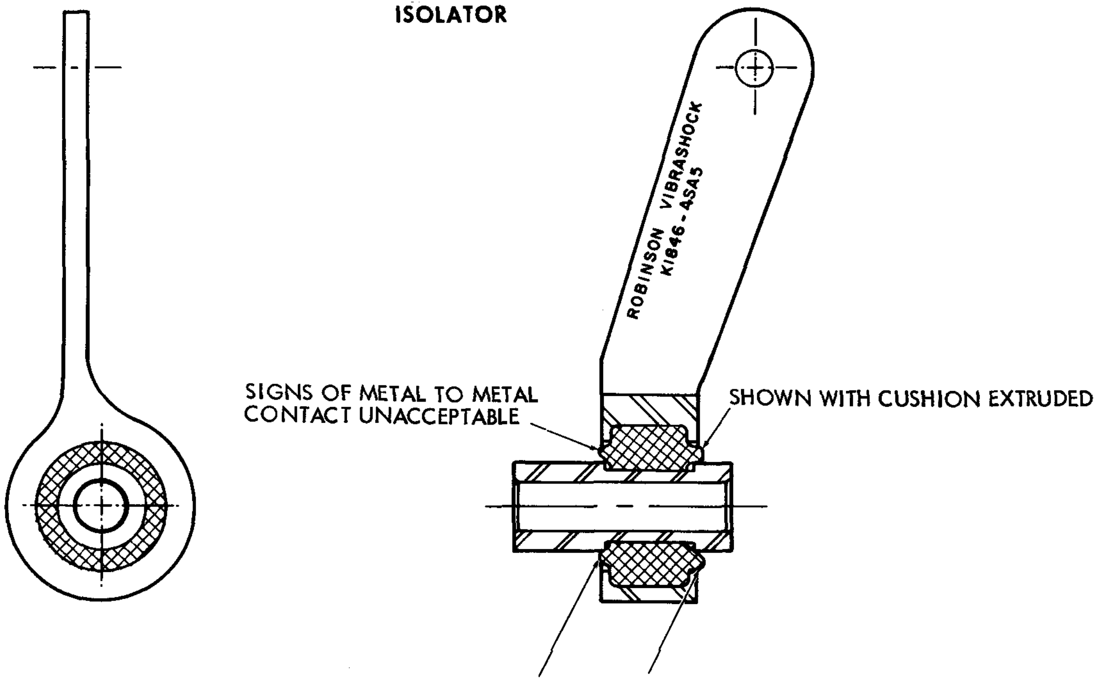


MEASUREMENT TO BE BETWEEN O.D. OF WASHER AND I.D. OF TRUNNION RING.

AREAS FOR MEASURING TO BE 180° APART (TYPICAL)

$M_2 - M_1$ MUST NOT BE MORE THAN 0.055

METAL FORWARD MOUNT ISOLATOR



SIGNS OF METAL TO METAL CONTACT UNACCEPTABLE

SHOWN WITH CUSHION EXTRUDED

INSPECT LOWER SIDE BETWEEN O.D. OF INNER HOUSING AND I.D. OF RING TRUNNION FOR CUSHION EXTRUSION OR SNUBBED CONDITION

Figure 71-9. Metal Engine Mount Inspection Data

CHAPTER 72
ENGINE

Contents Page

DESCRIPTION 72-2
General 72-2
Compressor Inlet Case Section 72-2
Compressor Section 72-2
Diffuser Case Section 72-2
Combustion Section 72-2
Turbine Section 72-2
Exhaust Case Section 72-2
Accessory Section 72-2
Engine Oil System 72-3
Engine Fuel System 72-3
Engine Starting System 72-3
Engine Instrumentation 72-3
Engine-Driven Hydraulic Pump 72-3
Engine Controls 72-3

OPERATION 72-3
Before Starting 72-3
Safety Check 72-3
Exterior Inspection 72-3
Interior Inspection 72-4
Starting Engines 72-4
Starting Engines - External Power 72-4
Starting Engines - Battery Power 72-4A
Failure to Start 72-5
Discontinued - Start Procedure 72-6
Engine Run 72-6
Stopping Engines 72-6
Ground Engine Fire Extinguishing Procedure 72-7

OPERATIONAL CHECKOUT 72-7

TROUBLESHOOTING 72-8

REMOVAL AND REPLACEMENT 72-10
Removal of Engine 72-10
Installation of Engine 72-11

ADJUSTMENT 72-11
Engine Trim - General 72-11
Engine Trim - Procedure 72-12

INSPECTION 72-14A
Engine Hot Section Inspection 72-15

Contents (Cont). Page

Engine Turbine Inspection 72-15
Engine Overspeed Inspection 72-15
Engine Major Inspection 72-15
Engine Vibration Check 72-15

FIGURES

72-1 Engine and Accessories 72-17
72-2 Engine Oil System Schematic Diagram 72-18
72-3 Rated Performance (JT12A-6 and JT12A-6A) 72-18A
72-3A Rated Performance (JT12A-8) 72-19
72-4 Engine Connections 72-20
72-5 Removal of Engine 72-21
72-6 Engine Trim Worksheet 72-22
72-7 Part Power Thrust Setting Curve 72-23
72-8 Takeoff Thrust Setting Curve 72-25
72-9 Fuel Control Adjustment 72-27
72-10 RPM Correction Curve 72-28
72-11 Maximum Acceptable EGT Curve 72-29

72.1 DESCRIPTION.

72.1.1 GENERAL. The JT12 commercial turbojet aircraft engine is a single rotor, axial-flow, gas turbine. It employs a compressor of moderate pressure ratio, can-annular burners, and a reaction turbine. The JT12A-6 engine is provided for aircraft 5001 through 5045. On aircraft 5046 through 5092 and 5094 through 5096, the JT12A-6A engine is provided. The JT12A-6A engine may be used on airplanes 5001 through 5045 if operated at the reduced JT12A-6 ratings. (See figure 72-3.) The JT12A-8 engine is used on 5093, 5097, and up.

72.1.2 COMPRESSOR INLET CASE SECTION. The forward part of the engine is the compressor inlet case, to which the airframe inlet duct is attached. It contains the compressor front bearing support and the inlet guide vanes which direct air to the compressor.

72.1.3 COMPRESSOR SECTION. The compressor contains nine rotor stages and eight stator stages. The air path within the compressor has an increasing inside diameter and a moderately decreasing outside diameter. Immediately downstream from the fourth rotor stage, compressor air is bled off by an automatic arrangement to prevent compressor stall.

72.1.4 DIFFUSER CASE SECTION. Just aft of the compressor is the diffuser case, which reduces the air velocity for entry into the combustion chambers. High-pressure air from this section is bled for engine functions, such as air inlet anti-icing, and for airframe functions, such as pressurization.

72.1.5 COMBUSTION SECTION. The combustion section houses eight burner cans (combustion chambers), each of which has a fuel nozzle. The cans are arranged annularly, hence the name, can-annular or cannular burner, and are interconnected by crossover tubes. Spark igniters, for use in starting only, are installed in burners 3 and 6. The numbering sequence is assumed from a position aft of the engine, looking forward. Then, starting with the top right burner as No. 1, count in a clockwise direction.

72.1.6 TURBINE SECTION. Aft of the combustion chambers is the two-stage turbine.

72.1.7 EXHAUST CASE SECTION. The exhaust case follows the turbine section. This case supports the engine tail pipe. Insulation blankets control heat radiation from these parts.

72.1.8 ACCESSORY SECTION. The accessory section is at the bottom of the engine, with the drives geared to the engine rotor by means of a shaft which passes through a strut in the engine diffuser case.

Pages 72-2A/72-2B Deleted.

72.1.8 (Cont).

72.1.9 ENGINE OIL SYSTEM. (See figure 72-2.) Each engine is provided with an integral oil system. This system is operated from a four-element pump located in the accessory section. One element of this pump supplies oil at a regulated pressure to the three main engine bearings and to the accessory section. The other three elements operate as scavenge pumps to return the oil to the engine mounted oil tank. The fuel-oil cooler is a heat exchanger which uses the engine fuel flow for cooling the oil. The oil tank, accessory section, and all main bearing compartments are interconnected by a breather system. This is in turn vented overboard through a centrifugal air-oil separator located in the accessory section.

72.1.10 ENGINE FUEL SYSTEM. Refer to Chapter 73 for information on the engine fuel system.

72.1.11 ENGINE STARTING SYSTEM. Refer to Chapter 80 for information on the engine starting system.

72.1.12 ENGINE INSTRUMENTATION. Refer to Chapter 77 for information on engine instrumentation.

72.1.13 ENGINE DRIVEN HYDRAULIC PUMP. Refer to Chapter 29 for information on the hydraulic pump.

72.1.14 ENGINE CONTROLS. Refer to Chapter 76 for information on engine controls.

72.2 OPERATION.

72.2.1 BEFORE STARTING. Before starting and running engines, the following safety check, exterior inspection, and interior inspection should be made:

72.2.1.1 SAFETY CHECK. Inspect the following items to assure that the airplane is safe before making the other checks:

Throttles	- "OFF"
Landing gear control handle	- "LG DOWN"
Flap control handle	- Check to ensure that the flap control handle is positioned tc agree with the known flap position prior to connecting external power to the airplane or turning on the battery switch.
Auxiliary hydraulic system pump	- Off
Fuel boost pumps	- Off
BATTERY switch	- "OFF"
ENG SEL switch	- "OFF"
Speed brake	- Extended; switch in neutral
Stall warning system	- Off

WARNING

The vane heaters are energized when the stall warning switches are on, and serious burns to personnel may result if the vanes are touched.

HF radio	- Off

WARNING

The HF system must be turned off during the external inspection. If power is applied to the airplane when the HF system is on, arcing can occur at most points on the airplane, particularly at the HF antenna, causing penetrating burns to personnel and fire or explosion in ground equipment.

72.2.1.2 EXTERIOR INSPECTION.

Nose landing gear ground safety pin	- Installed
Nose landing gear scissors pin	- Installed
Brake accumulator	- Charged
Main landing gear ground locks	- Installed
Wheel chocks	- In place
Engine inlet and exhaust plugs	- Removed
Engine inlet and exhaust areas for evidence of foreign object damage and freedom of intake and tailpipes from tools and debris	- Checked
Thrust-reverser ground safety locks	- Removed
Engine oil system	- Serviced

Aft compartment:

WARNING

Make sure that the speed brake ground control handle is pulled to "SAFE" before entering the aft compartment. Push in to "NORMAL" after leaving the compartment.

Fire extinguisher bottles	- Charged

72.2.1.2 (Cont).

Refrigeration units oil quantity - Checked for normal quantity of oil

Hydraulic reservoirs - Checked for normal quantity of fluid

Hydraulic accumulators - Checked for normal pressure

72.2.1.3 INTERIOR INSPECTION. Depress all push-to-test warning lights to assure proper operation as the following items in the pilots' compartment are checked:

CO_2 fire extinguishers - Charged/Stowed

Emergency generator selector - Normal

Overhead audio control panel - Set

Bleed air control switches - Set

Air conditioning panel - As desired

BATTERY switch - "OFF" if external power is available; "ON" if not available.

DC control panel - Generators off

AC control panel - AC system normal; inverters on.

Interior and exterior lights -- As desired

Pilot's circuit breaker panel - Checked

Pilot's interphone audio selector panel - Set

Emergency landing gear release handle - In

LG CONT switch - "NORM"

CAUTION

If this switch is left in the "EMERG" position, there will be no nose wheel steering nor will brakes be available on hydraulic system No. 1 after accumulator is depleted.

Engine anti-icing switches - Off

DRAG CHUTE handle (If Installed) - In

FIRE EXT AGENT switch - Off (centered) with guard down

Fire emergency handles - In

Fire detection warning lights - Tested

Annunciator warning lights - Tested

Brake and flap system selector switch - "NORM" - 1"

FUEL JETTISON switches - Safetied

FUEL TK ISOLATION switches - Safetied

TANK INTERCONNECT switches - Safetied

Windshield anti-icing - Off unless necessary to remove ice

Engine fuel heaters - Off

Pitot heaters - Off

ENG OIL PRESS LOW ANNUNCIATOR LT CUTOFF switches - "ON"

De-icing control panel - Switches off

Copilot's circuit breaker panels - Checked

Fuel management panel - Set

Rudder gust lock - Unlocked

72.2.2 STARTING ENGINES.

72.2.2.1 STARTING ENGINES - EXTERNAL POWER.

CAUTION

For external cart starting, use only constant potential sources with a rating in the range of 1,000 to 1,500 amperes or constant current sources with current limiting set not higher than 1,200 amperes. Higher settings will damage or shorten appreciably the life of the starter-generator.

Fire protection - Check that adequate fire protection is available, and that all personnel and equipment are clear for the start cycle.

Auxiliary hydraulic pump - On

72.2.2.1 (Cont)

Brakes - Set; brake low pressure light off.

Auxiliary hydraulic pump - Off

POWER SEL switch - "GPU START"

AIR START switch - "NORMAL"

FUEL AIR BLEED LINE switch - "CLOSED"

ENG SEL switch - Set to engine to be started. If all four engines will be run, start in 3, 4, 2, 1, order.

Voltmeter selector switch - Set for engine to be started.

Main fuel tank boost pumps -All on

No. 1 MAIN INV control switch -"TRANSFER"

No. 2 MAIN INV control switch -"ON"

Note

Aircraft 5078, 5092, and up are provided with a cutoff throttle detent. On earlier aircraft the throttle may have to be held against the cutoff stop to engage starter.

START switch - "START" and hold

CAUTION

The starter duty cycle is 30 seconds on, 2 minutes off, 30 seconds on, and then 15 minutes off. Do not exceed this duty cycle.

Throttle - Advance to "IDLE" when RPM reaches 10 to 12 percent

Note

On aircraft 5078, 5092, and up, it is necessary to lift the throttle out of the cutoff detent to advance to "Idle."

Oil pressure - Up

CAUTION

Discontinue the start if a rise in oil pressure is not observed at idle RPM or if the EGT exceeds 525°C.

START switch - Released at 23 to 30 percent RPM.

STARTER ENGAGED light - Out

Hydraulic pressure - Checked for 3,000 PSI on the No. 1 system when the No. 2 engine is started, or on the No. 2 system when the No. 3 engine is started.

Generator voltage - Checked for 28 (±1) volts

CAUTION

If voltmeter reading exceeds 30 volts, shut engine down at once, and check voltage regulator lockout relay contacts for welding.

(Repeat the above procedure for other engines to be started.)

ENG SEL switch - "OFF"

POWER SEL switch - Off

AIR START switch - "OFF"

Generator control switches for operating engines - "ON"

Generator loadmeters - Checked for normal loading

BATTERY switch - "ON"

All inverters - Checked for 26 and 115 volts and 400 cycles

72.2.2.2 STARTING ENGINES - BATTERY POWER.

Fire protection - Check that adequate fire protection is available, and that personnel and equipment are clear for the start cycle.

Battery switch - "ON"

Auxiliary hydraulic pump - On

Brakes - Set; brake low-pressure light off.

Auxiliary hydraulic pump - Off

Windshield inverter - Off

POWER SEL switch - "BAT START"

AIR START switch - "NORMAL"

72.2.2.2 (Cont)

FUEL AIR BLEED LINE switch - "CLOSED"

ENG SEL switch - Set to engine to be started. If all four engines will be run, start in 3, 4, 2, 1 order.

Voltmeter selector switch - Set for engine to be started.

Fuel tank boost pump corresponding to engine to be started - On

No. 1 MAIN INV control switch - "TRANSFER"

No. 2 MAIN INV control switch - "ON"

Note

Aircraft 5078, 5092, and up are provided with a cutoff throttle detent. On earlier aircraft the throttle may have to be held against the cutoff stop to engage starter.

START switch - "START" and hold.

CAUTION

The starter duty cycle is 30 seconds on, 2 minutes off, 30 seconds on, and then 15 minutes off. Do not exceed this duty cycle.

Note

To prolong battery life and to prevent overheat conditions, observe the following battery starting duty cycle:

Phase I Limit starting cycle to six attempts to start (successful or unsuccessful) followed by a charge at 28 volts for 20 to 30 minutes maximum if Phase II is required.

Phase II Limit starting cycle to four attempts to start (successful or unsuccessful) followed by a charge at 28 volts for 20 to 30 minutes maximum if Phase III is required.

Phase III Limit starting cycle to two attempts to start (successful or unsuccessful) followed by a 4-hour cooling period, followed by a charge at 28 volts for 20 to 30 minutes and repeat Phases I, II, and III if necessary.

In cases where the battery starting duty cycle and the starter duty cycle conflict, the more stringent requirement must be followed. If during the starting cycle, the battery fails to crank the engine to light-off speed, it may be due to a reduced state of charge or to low battery temperature. To recharge, apply 28 volts for 30 minutes. To warm the battery, apply external heat and then recharge it for 30 minutes.

THROTTLE - Advance to "IDLE" when RPM reaches 10 to 12 percent.

Note

On aircraft 5078, 5092, and up, it is necessary to lift the throttle out of the cutoff detent to advance to "Idle."

Oil pressure - Up

CAUTION

Discontinue the start if a rise in oil pressure is not observed at idle RPM or if the EGT exceeds 525°C.

Note

The voltmeter will "peg" after the initial starting current drops and until the start cycle is completed. This is a normal condition and will not damage the voltmeter.

START switch - Released at 23 to 30 percent RPM

STARTER ENGAGED light - Out

Generator control switch for operating engine - "ON"

BATTERY switch - "OFF"

No. 1 MAIN INV control switch - "ON"

Loadmeter for generator of operating engine - Checked

Hydraulic pressure - Checked

72.2.2.2 (Cont).

Generator voltage (28 ±1) - Checked

CAUTION

If voltmeter reading exceeds 30 volts, shut engine down at once, and check voltage regulator lockout relay contacts for welding.

(Repeat the above procedure for engine No. 4)

BATTERY switch - "ON"

(Repeat the above procedure for engines 2 and 1 with the battery switch on.)

Note

This procedure allows starting two engines on battery and using generator-assist to start the remaining two engines.

ENG SEL switch - "OFF"

POWER SEL switch - Off

AIR START switch - "OFF"

All generator loadmeters - Checked

Windshield inverter - On and Checked

AC voltage and frequency - Checked for 26 and 115 volts and 400 cycles.

72.2.2.3 FAILURE TO START. If for any reason a starting attempt is discontinued, the entire starting sequence must be repeated from the beginning. The starting attempt should be discontinued immediately if any of the following conditions exist:

a. The engine does not reach 10 percent RPM within 10 seconds after engaging the start switch.

b. The engine fails to light off within 20 seconds after the throttle is advanced.

c. The fuel is shut off inadvertently.

d. If idle RPM (44 to 46 percent) is not reached within 1 minute after combustion.

e. If the engine lights off but the exhaust gas temperature exceeds the established limits, refer to Chapter 5 for required action.

f. If the start is not obtained, observe a 3-minute delay before attempting another start.

72.2.2.4 DISCONTINUED-START PROCEDURE.

a. Throttle - off

b. START switch - Released

72.2.3 ENGINE RUN. Monitor all engine instruments and warning lights during the engine run to make sure that engine limits are not exceeded. All four engines may be run up to takeoff power simultaneously, if the wheels are chocked and the brakes are set and in good working order. Do not run the engines at a higher power setting than necessary or any longer than necessary since this reduces engine life.

CAUTION

Prolonged operation of an engine with a broken compressor bleed strap is harmful and should be avoided. Operating the engine with this condition can induce adverse stresses in the compressor rotor in the area of the bleed valve. A broken bleed strap is indicated, during normal power settings by low engine pressure ratio (EPR) or by increased RPM to maintain the EPR setting. These, in turn, cause increased fuel flow and exhaust gas temperature. If any of these changes are noted, inspect for a broken bleed strap and replace, if required, before further engine operation.

CAUTION

The engine-driven fuel pumps are lubricated by the fuel they are pumping. In case of an engine fire warning, always retard the throttle to the "OFF" position before pulling the fire emergency handle, if time permits. This will prevent the pumps from running dry. If the engine is shut down with the fire emergency handle, inspect all engine fuel pump and engine fuel control filters for evidence of fuel pump damage.

CAUTION

Prolonged engine operation at idle speed during severe icing conditions should be avoided. Even with the anti-icing system ON some ice will accumulate and can result in damage to the engine.

72.2.4 STOPPING ENGINES.

Parking brakes - Set

Throttles - "OFF"

Note

Prior to all engine shut-downs, the engines should be operated for 2 to 5 minutes at idle, depending on the intensity of prior running, then accelerated to 75 percent RPM for 15 to 30 seconds. Shut-down should be made directly from 75 percent RPM. This procedure ensures adequate internal cooling and

72.2.4 (Cont).

proper scavenging of residual oil. Check that the engines decelerate freely, and listen for any unusual noises during shut-down. Engine-driven pump noise is a normal condition when the pump is functioning properly. Oil level should be checked within one hour after shut-down prior to any motoring of the engine. Motoring-over the engine tends to load the engine bearing compartments with oil. If the engine is motored-over at any time prior to servicing the oil tank, the engine must be started and the shut-down procedure outlined above followed before servicing the tank. Once the engine is serviced following shut-down, it will not be necessary to recheck the oil level until termination of the next flight provided the oil strainer has not been removed.

CAUTION

An excessive quantity of oil in the engine oil system can result in possible erratic oil pressure indication or in damaging internal tank pressures with possible rupture.

Wheel chocks	- In place
All switches and systems except battery	- Off
Battery voltages	- Checked for 24 volts
BATTERY switch	- "OFF"
Rudder gust lock	- Set

72.2.5 GROUND ENGINE FIRE EXTINGUISHING PROCEDURE. Use the following procedure if there is evidence of a fire within the engine or extremely high exhaust gas temperature indication.

a. Place the throttle for the affected engine in the "OFF" position.

b. As soon as the tachometer indicates deceleration to 40 percent RPM or less, position the ENG SEL switch to the affected engine.

c. Place the POWER SEL switch in either "GPU" or "BAT," depending on the source of power available.

d. Place the AIR START switch in the "OFF" position.

e. Place the START switch in the "START" position, and maintain starter operation for 20 seconds.

f. Repeat starter operation as necessary.

CAUTION

Do not spray ground fire extinguisher agent into the engine air intake or exhaust tailpipe except in case of a fire which obviously could endanger the airplane.

72.3 OPERATIONAL CHECKOUT. Use the following procedure to ground-check engine operation. Always limit ground operating time to conserve the engine for flight. Steps (c), (d), and (e) check only the thrust-reverser system and may be omitted if that system is known to be operating satisfactorily. Steps (f), (g), and (h) check only the engine anti-icing system and may be omitted if that system is known to be operating satisfactorily.

a. Start the engine in accordance with the instructions in paragraph 72.2.2, then let it run at idle RPM.
RESULT: Start should be normal. Engine operation should be within the limits of figure 72-3. Engine speed should be between 44 and 46 percent RPM. The low-fuel-pressure light should be off, and hydraulic pressure should be 3,000 (±150) PSI.

b. With the engine speed stabilized, observe engine instruments for fluctuation.
RESULT: Readings should be within the following limits: engine pressure ratio (EPR) 0.10 (double amplitude); RPM 0.4 percent (double amplitude); and fuel flow 50 pounds per hour (PPH) (double amplitude).

c. With the engine running at idle, place the thrust-reverser lever for operating engine in the reverse thrust position.
RESULT: (1) The thrust-reverser doors should move to the reverse-thrust position within 2 seconds.
(2) The thrust-reverser unlocked light should go on followed by the reverser extended light within 2 seconds.

d. Return the thrust-reverser lever to the forward-thrust position.
RESULT: (1) The thrust-reverser doors should move to the forward-thrust position within 2 seconds.
(2) The reverser extended light should go out followed by the thrust-reverser unlocked light within 2 seconds.

e. Repeat steps (c) and (d) for the other engines being checked.

f. Place the No. 1 engine anti-icing switch in the "ON" position.
RESULT: (1) The No. 1 engine anti-icing valve open light should go on.

■ 72.3 (Cont).

(2) There should be airflow into the inlet duct at the No. 1 engine nacelle.

CAUTION

Operate the engine anti-icing system only long enough to determine proper operation. Never operate the system more than 5 minutes on the ground.

g. Return the engine anti-icing switch to the "OFF" position.
RESULT: (1) The No. 1 engine anti-icing valve open light should go off.
(2) Airflow into the nacelle should stop.

h. Repeat steps (f) and (g) for the other engines being checked.

i. Momentarily increase engine speed to full throttle or takeoff power:
RESULT: (1) The EPR should increase to the value shown on the takeoff thrust setting curve. (See figure 72-8.)
(2) The EGT should not increase above the value shown on the maximum acceptable EGT curve. (See figure 72-11.)
(3) Engine acceleration time from idle to takeoff EPR should not exceed 6.5 seconds.

CAUTION

Advance the throttle only enough to obtain the required EPR. Take extreme care to avoid exceeding maximum RPM or EPR. Hold takeoff power only long enough to make the necessary checks; then reduce power to idle.

Note

Failure to meet the required acceleration at any particular checkout should not take an aircraft out of commission. Correction of this condition by replacement of the engine fuel control or by other means should, however, be accomplished at the first convenient opportunity.

j. Repeat step (i) for the other engines being tested. If any engine does not meet the specified EPR and EGT requirements, trim the engine.

k. Shut down the engines, and restore the airplane to normal.

Note

Check operation of the fuel pressurization and dump valves during shutdown by observing that a quantity of fuel (approximately 1/2 pint) ejects from the engine drain mast. Normal seepage through the pressurization and dump valve is 300 CC per hour when the engine is running.

Note

If the airplane is to be left inactive with power applied for extended periods, it is recommended that the power be removed from the anti-icing valves and the fuel flow transmitters by pulling the ANTI-ICE PWR and the FUEL FLOW IND circuit breakers.

■ 72.4 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
FLUCTUATION OF RPM AND TURBINE EXHAUST TEMPERATURE.		
Defective engine fuel system.	Observe fuel flow indicator for fluctuations.	Troubleshoot the engine fuel system in accordance with the procedure in Chapter 73.
ENGINE NOT DEVELOPING FULL THRUST AT HIGH POWER SETTINGS AND/OR STALLS DURING ACCELERATION OR DECELERATION.		
Bleed valve not closing.	Check linkage between fuel control and bleed strap. Check bleed strap.	Lubricate, adjust or replace parts as required.

72.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INCORRECT OR FLUCTUATING OIL PRESSURE.		
Oil level in tank too high or insufficient oil supply.	Shut down engine as indicated in paragraph 72.2.4 and check oil level.	Service oil system in accordance with Chapter 12.
Clogged or dirty oil strainer.	Remove and inspect the oil strainer.	Clean the strainer.
Clogged restrictor on oil pressure transmitter.	Inspect restrictor.	Clean or replace the restrictor.
Defective transmitter vent line	Check condition of vent line between transmitter and engine	Replace vent line (use one of all metal construction if available).
Defective oil pressure transmitter and/or indicator.	Attach a direct-reading gage to the tap adjacent to the oil pressure transmitter housing and check pressure.	Replace the transmitter and/or indicator.
Misadjusted relief valve.	Check the pressure indicating system and make an engine operational check. If no malfunction is indicated, the relief valve is probably out of adjustment.	Adjust relief valve.
ENGINE WILL NOT ROTATE DURING START ATTEMPT.		
Engine is "frozen."	Attempt to rotate the engine by hand. If the engine will not rotate, it is "frozen."	Replace engine.
Starter drive shaft is sheared.	If starter can be heard turning but engine does not rotate, a starter drive shaft is sheared. Remove starter-generator, and check whether its shaft is sheared or whether the shaft in the accessory gearbox is sheared.	If shaft of starter-generator is sheared, replace starter-generator. If shaft in accessory gearbox is sheared, replace engine.
INSUFFICIENT RPM DURING ATTEMPTED START.		
Excessive turbine and compressor blade rubbing.	Listen for loud scraping noise while engine is rotating.	Replace engine.
ENGINE FAILS TO LIGHT-OFF WHEN THROTTLE IS ADVANCED TO IDLE.		
No fuel to engine.	Check that crossfeed valves are set for tank-to-engine flow and that appropriate low fuel pressure indicator light is off.	If fuel pressure is not available, refer to Chapter 28 for airplane fuel system troubleshooting procedures. If fuel pressure is available, refer to Chapter 73 for engine fuel system troubleshooting procedures.

72.4 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Defective throttle ignition switches.	Perform continuity check of throttle switches.	Adjust or replace ignition switches.
Defective engine fuel system.	Check fuel flow indicator for minimum starting fuel flow.	Troubleshoot the engine fuel system in accordance with the procedures in Chapter 73.
ENGINE LIGHTS OFF BUT DOES NOT ACCELERATE TO IDLE.		
Engine is "frozen."	Attempt to rotate engine by hand. If unusual resistance is observed, engine is "frozen."	Replace engine.
ENGINE RUNS ROUGH.		
Internal interference between rotor and stators.	Check for scraping noise as engine slows during shutdown. Check oil strainer for metal particles.	Replace engine.
Defective accessories.	Check for unusual noises in accessory section and for malfunction of an accessory-driven aircraft system.	Replace defective accessory.
EXCESSIVE OIL CONSUMPTION.		
Cracked carbon seals at bearing locations.	Check inlet and exhaust areas for collections of oil.	Replace the engine.
Defective oil pressure and scavenge lines or connections.	Visually inspect all oil lines and connections for evidence of leaks.	Tighten or replace as necessary.
Defective oil pump.	Inspect for oil mist spraying from the compressor bleed valve, and for low oil level in tank.	Replace engine.

Note

The engine should be replaced in cases where the amount of discharge from the overboard breather shows a definite increase with time, or where the oil consumption shows a rapid increase and/or the maximum allowable oil consumption is 0.13 gallons per hour, measured over a ten hour period. Isolate the engine section responsible for the breather discharge as an aid in determining the corrective action required.

72.5 REMOVAL AND REPLACEMENT.

72.5.1 REMOVAL OF ENGINE (See figure 72-5).

Tools Required

Nomenclature	Description	Use and Application
Engine Hoist	Lockheed Part Number 11015-MEH	To lower and raise engine in nacelle.
Door Unlock Tool	Lockheed Part Number 11098-MEU-1	To unlock thrust-reverser doors during engine replacement.
Torque Wrench	50 to 70 inch-pounds	To torque engine mount bolts.

72.5.1 (Cont).

a. Remove the three access doors from the top of the nacelle, and the bottom aft access door. Unlatch the center door, and support it in the open position.

b. Install the engine hoist assembly (Lockheed Part No. 11015-MEH) on the nacelle.

c. Remove the forward clamp from the engine inlet duct, loosen the aft clamp and slide inlet duct as far aft as possible.

d. Disconnect the generator cooling duct clamp, and remove the duct and support bracket.

e. Connect the engine hoist cables to the forward and aft engine hoist points through the upper access openings. Take the weight of the engine on the hoist.

f. Disconnect the items listed in Figure 72-4.

g. Remove the exhaust ejector assembly.

h. Using the thrust-reverser unlocking tool (Lockheed Part Number 11098-MEU-1), push the thrust-reverser guide forward past the locked position.

i. *Hold the doors by the forward edge, and pull the* doors to the reverse-thrust position. Install the thrust-reverser ground lock.

j. Remove the frame section extending across the bottom aft nacelle.

k. Remove the bolt from the forward engine mount. This bolt is to be tightened to 50 to 70 inch-pounds when installing the engine.

l. Loosen the nuts on each main engine mount, and swing the mounts open. These nuts are to be tightened to 50 to 70 inch-pounds when installing the engine.

m. Lower the engine from the nacelle.

72.5.2 INSTALLATION OF ENGINE.

a. Apply a light coating of Specification MIL-L-6086 oil (See figure 71-7), or any other petroleum base oil, to the rubber parts of the engine mounts.

Note

The main engine mounts should be treated as assemblies. If the spherical rubber mount requires replacement, the trunnion post should also be replaced.

b. Apply a light coating of molybdenum disulfide grease, Specification MIL-G-21164 (See figure 71-7), between the engine mount trunnions and the mating or sliding surfaces of the engine mounts.

c. Install the engine by reversing the removal procedure.

CAUTION

Special care is required in tightening up the clamps on the engine inlet duct. The duct should first be centered fore and aft so that the gap at both ends is approximately the same. The splitters should also be aligned with the splitters in the nacelle inlet duct as closely as possible. Both clamps should be tightened evenly to a torque of 15 to 25 inch-pounds.

d. Install, tighten, and safety-wire the engine connections specified in figure 72-4.

CAUTION

Each time an engine is replaced, check that the starter-generator wiring to the generator studs has not been reversed before starting the engine. If the wiring connection is reversed, damage to the starter-generator and to the generator control panel can occur within approximately 1 minute after engine start. A reverse connection is shown by the fact that the generator does not come on the line when the engine is started. (Refer to chapter 24 for the proper generator control panel checkout procedure if it is suspected that the generator has been misconnected and then operated.)

CAUTION

After engine installation, motor the engine without ignition to prime the fuel flow transmitter.

e. Rig the engine throttle system. (See Chapter 76.)

72.6 ADJUSTMENT.

72.6.1 ENGINE TRIM - GENERAL.

a. Each engine produces its rated thrust (or more) at an engine pressure ratio (EPR) shown on the thrust setting curve. However, the engine fuel control is trimmed at a part-power condition which is at a thrust level low enough to allow trimming over a complete range of ambient temperatures. Otherwise, trimming at rated power on a cold day could result in thrust below rated on a hot day.

b. The properly trimmed control provides sufficient margin at the part-power stop trim level to allow the engine to make rated thrust. Although the engine thrust level and the fuel control trim are determined by EPR, the fuel control itself senses engine RPM. The relationship of RPM to thrust can

72.6.1 (Cont).

Equipment Required

Nomenclature	Description	Use and Application
Absolute Manifold Pressure Gage	0 to 100 inches Hg. accurate to ±0.10 inch Hg.	To measure engine pressure.
Mercury Thermometer	Calibrated in degrees centigrade.	To measure ambient temperature (TAM)
Flexible Hose and Clamp		To connect the pressure gage to the engine.
Small Screwdriver and 5/32-inch Allen Wrench		To adjust Holley fuel control.
5/64-inch Allen Wrench		To adjust Hamilton Standard fuel control.

change as service life increases (EPR in relation to thrust remains constant) so that retrimming may be necessary.

c. The part-power trim speed at which each engine produced part power thrust at the time of manufacture is stamped on a data plate on the bottom of the accessory gearbox. When the engine is prevented from reaching rated EPR by maximum throttle position, a trim check may be made. When a new fuel control is installed, a trim check must be performed.

CAUTION

To ensure that positive control will be available during trimming, always rig the engine throttle system to remove as much tolerance and play in the system as possible before trimming the engine (see Chapter 76).

CAUTION

Portions of the trimming procedure specify adjustment of the idle and takeoff setscrews on the fuel control. These are the only two fuel control adjustments that should be changed. Do not adjust other setscrews on the fuel control since these are for bench calibration only. Particularly avoid any attempt to adjust the throttle valve differential pressure adjustment setscrew. Adjustment of this screw can result in a change in throttle valve differential pressure and cause severe shifts in fuel flow.

72.6.2 ENGINE TRIM - PROCEDURE.

a. Assemble the equipment listed in the table above.

Note

The first three items of equipment may be replaced with a Howell Engine Trim Checker H218-1, or equal. When this instrument is used, a calibration card that will permit pressure readings within ±0.10 inch Hg. accuracy should be available. The trim checker may also be used for instrument calibration, as indicated in Chapter 77.

Note

An operational checkout of EPR, tachometer and EGT indicating systems per chapter 77 at each engine hot section inspection, engine change or EGT indicator change is recommended.

b. Head the airplane into the wind, and check that the wheel chocks are securely in place. If possible engine trimming should be performed with the aircraft headed directly into the wind with wind velocities less than 5 MPH. Since these conditions do not always exist, the following limitations are permissible:

A maximum of 15 knots wind within ±45 degrees of the engine inlet axis.

A maximum of 5 knots wind within the segment between ±45 degrees and ±90 degrees of the engine inlet axis.

No trimming allowed with wind up the tailpipe or within ±90 degrees of the tailpipe axis.

If wind conditions are greater than allowed in the first two items, and if necessity requires that the engine be trimmed regardless of unfavorable conditions, then the engine must be retrimmed under proper wind conditions at the first opportunity.

72.6.2 (Cont)

Note

The term "Pt5" denotes the total pressure at the exhaust pressure probes. Similarly, "Tt5" denotes the total temperature at the exhaust temperature probes. "Pam" denotes ambient or barometric pressure, and "Tam" denotes ambient temperature.

c. Connect the manifold pressure gage to the engine. The gage connects to the engine Pt5 connection at the turbine exhaust case to measure engine exhaust pressure.

d. Install the part-power stop on the fuel control. This part is normally stowed on the fuel control on all engines and should be replaced in the stowed position after each use. On earlier Hamilton Standard fuel controls, a retaining screw holds the stop securely in place when tightened. On later Hamilton Standard units, the stop can move radially when it is in either the part power or the stowed position, even with the screw securely tightened. Note which type unit is installed, and avoid overtightening of the screw on later controls. Secure the stop in the stowed position on these controls by means of lockwire through the hole provided.

e. Obtain the barometric pressure (Pam).

CAUTION

Barometric pressure is generally corrected to sea level. Insure that the barometric pressure reading is the actual field barometric pressure.

f. Obtain the outside air temperature (OAT or Tam) at the airplane by placing the mercury thermometer in the shade of the airplane.

g. Record the barometric pressure and OAT on line 4 of the engine trim worksheet. (see figure 72-6.)

Note

The maximum time between obtaining the OAT and barometric pressure and the actual engine run should not exceed 15 minutes.

h. Obtain the minimum desired Pt5 by drawing a vertical line from the appropriate OAT value on the part-power thrust setting curve (figure 72-7) to the minimum fuel control tolerance line. Then draw a horizontal line across to intersect the correct field barometric pressure line. From this point, draw a vertical line down to the turbine discharge pressure scale (Pt5). Record this value on line 5 of the worksheet.

i. Obtain the maximum desired part-power Pt5 by adding 0.5 inch Hg to the minmum part power Pt5, and record this value.

j. Obtain the desired part-power minimum engine pressure ratio (EPR) by drawing a vertical line from the appropriate OAT value on the part-power thrust setting curve to the minimum fuel control tolerance line. Then draw a horizontal line to the EPR scale.

k. Obtain the minmum desired takeoff power Pt5 by drawing a vertical line from the appropriate OAT value on the takeoff thrust setting curve (figure 72-8) to the minimum fuel control tolerance line, or to the barometric pressure line, whichever is lowest. Then draw a horizontal line across to intersect the correct field barometric pressure line. From this point, draw a vertical line down to the turbine discharge pressure scale (Pt5). Record this value on lines 6 and 11 of the worksheet.

l. Obtain the desired takeoff EPR by drawing a vertical line from the appropriate OAT value on the takeoff thrust setting curve (figure 72-8) to the minimum fuel control tolerance line, or to the barometric pressure line, whichever is lowest. Then draw a horizontal line across to the EPR. Record this value on line 6 of the worksheet.

m. Enter the data plate speed on line 13 of the worksheet. This speed is stamped on the data plate on the bottom of the accessory gearbox, or is recorded in the engine historical record.

n. Start the engine which is to be trimmed according to the engine ground operation procedures.

o. Close the bleed air shutoff valve, and place the GEN CONT switch in the "OFF" position.

p. Position the throttle lever to "IDLE," and allow the engine speed to stabilize.

q. The engine idle speed should be between 44 and 45% RPM. If the speed is outside these limits, adjust the idle speed at the screwdriver adjustment (Holley) or the Allen key adjustment (Hamilton Standard) on the fuel control. (See figure 72-9.)

Note

Whenever possible make adjustments to the high side of the above tolerance. This will provide better engine performance.

r. Advance the throttle to the part-power stop. Allow the engine speed to stabilize for 5 minutes, then check the pressure gages attached to the

72.6.2 (Cont).

engine and the engine instruments inside the airplane. Record the information on line 7 of the worksheet. Tt5 is the exhaust gas temperature.

s. Determine if Pt5 is within the limits of line 5 on the worksheet.

t. If Pt5 is not within limits, adjust the maximum setting (5/32-inch Allen wrench adjustment for Holley; 5/64-inch Allen wrench adjustment for Hamilton Standard) on the fuel control. (See figure 72-9.)

u. If an adjustment is made, repeat steps (r) through (t) until the desired Pt5 is obtained.

v. After the desired Pt5 is obtained, repeat step (r). Record the information on lines 9 and 10 of the worksheets.

w. Retard the throttle to "IDLE", and recheck the idle setting. Readjust the setting if the RPM is not between 44 and 45 percent. Adjusting the maximum trim adjustment can affect the idle adjustment slightly. If the idle trim is adjusted on a Hamilton Standard fuel control, re-check the part-power setting and adjust as required.

x. Remove the part-power stop, and advance the throttle to the minimum takeoff Pt5 recorded on line 6 of the worksheet. If this Pt5 cannot be obtained with full throttle travel, check throttle rigging per Chapter 76.

y. Check the indicated EGT at takeoff power (static) against figure 72-11, entering the curve at the existing ambient temperature. Make sure the bleed air shutoff valve switches are "OFF". The EGT should not exceed the maximum value for the existing ambient temperature shown in figure 72-11. This will assure that the maximum permissible EGT will not be exceeded at a takeoff power at any other ambient temperature, or with cabin pressurization in operation. If the EGT exceeds the value shown in figure 72-11, shut down the engine, and investigate operation of the fourth stage bleed air valve linkage to assure normal operation. If the linkage appears normal, and the bleed valve strap is intact, return the engine to overhaul for corrective action.

z. Allow the engine to stabilize, and then record the takeoff data on lines 11 and 12 of the worksheet. Do not operate the engine at takeoff power longer than necessary.

aa. Retard the throttle to "IDLE", and recheck the idle speed.

ab. Advance the throttle from idle to takeoff, observing the time required for the engine to accelerate. Engine speed should accelerate from idle to takeoff EPR in 6.5 seconds or less.

Note

Failure to meet the required acceleration at any particular preflight check should not be sufficient reason to take the airplane out of service. Correction of this condition by replacement of the engine fuel control or by other means should, however, be accomplished at the first opportunity.

ac. Shut down engine in accordance with paragraph 72.2.4.

ad. Enter the percent RPM from line 9 of the worksheet on line 14, and convert the percent RPM to actual RPM by multiplying the percent RPM by 159.09 (15,909 RPM is equivalent to 100 percent tachometer reading).

ae. Obtain the speed bias value from the RPM correction curve shown on figure 72-10. Enter on line 15.

af. Multiply the speed bias value by the data plate speed, and enter the product on line 16 of the worksheet. This is the adjusted data plate speed.

ag. Obtain the difference between the adjusted data plate speed and the observed RPM, and record the difference on line 17 of the worksheet.

ah. If the observed RPM exceeds the adjusted data plate speed by 300 RPM or more, and investigation discloses no Pt5 indicating system leakage or instrument errors, return the engine to overhaul for cleaning.

ai. Remove all the instruments from the engine.

72.7 INSPECTION. Engine inspections are required as indicated in Chapter 5 and in the following tables (Applicable to JT12A-6, -6A and -8 except where indicated for the -8 only.):

a.

CONDITION	INSPECTION REQUIRED
Operating EGT in excess of:	
800°C	Complete Engine Overhaul
730°C 770°C*	Turbine Inspection (Para. 72.7.2)
720°C 760°C* for 5 sec. or more	Turbine Inspection (Para. 72.7.2)
720°C 760°C*	Hot Section Inspection (Para. 72.7.1)
690°C 730°C* for 2 min. or more	Turbine Inspection (Para. 72.7.2)
690°C 730°C* for 5 sec. or more	Hot Section Inspection (Para. 72.7.1)
677°C 720°C*	Shut down and determine cause
Starting EGT in excess of:	
800°C	Complete Engine Overhaul
700°C	Turbine Inspection (Para. 72.7.2)
630°C for 5 sec. or more	Turbine Inspection (Para. 72.7.2)
630°C	Hot Section Inspection (Para. 72.7.1)
595°C for 5 sec. or more	Hot Section Inspection (Para. 72.7.1)
525°C	Shut down and determine cause

b.

CONDITION	INSPECTION REQUIRED
Engine Speed in excess of:	
106.2% RPM	Complete Engine Overhaul
105.0% RPM	Overspeed Inspection (Para. 72.7.3)

* Applicable only to the JT12A-8 Engine.

72.7.1 ENGINE HOT SECTION INSPECTION. The following will be inspected to see that they are within the limits prescribed in the P & W JT12A Maintenance Manual:

a. External area of combustion chamber outer case for hot spots, distortion and cracks; combustion chamber fuel drain valve for cleanliness, damage, and security.

b. Combustion chambers, retaining clamps and retaining rings for cracks, distortion, burning and erosion; flame tubes for coking, cracks, and burning.

c. Fuel manifolds (right and left) for cracks and security; fuel nozzles for dents and burning on face; air swirl guides for security, damage, and cracks.

d. Combustion chamber inner case for cracks, distortion, and hot spots.

e. Compressor exit vanes, ninth-stage stator vanes, and compressor blades for nicks, cracks, evidence of foreign object damage, and missing blades or vanes.

f. Turbine blades and vanes for burned spots, cracks, nicks, dents and distortion; exhaust duct for foreign objects.

g. Combustion chamber outlet duct for hot spots, cracks, and distortion.

h. Second stage turbine for cracked, bent, broken or missing blades.

i. Exhaust cone and strut assembly for distortion, cracks, and damage.

72.7.2 ENGINE TURBINE INSPECTION. The following will be inspected to see that they are within the limits prescribed in the P&W JT12A Maintenance Manual:

a. All items listed under 72.7.1 Engine Hot Section Inspection.

b. Turbine vanes for bow; fluorescent penetrant inspection of all turbine blades.

c. Turbine blades and disk for growth and hardness (first and second stage turbine disk hardness must be at least 66 on Rockwell A scale).

72.7.3 ENGINE OVERSPEED INSPECTION. The following will be inspected to see that they are within the limits of the P&W JT12A Maintenance Manual:

a. Air inlet duct for foreign material.

b. Inlet guide vanes for damage.

c. Visible stages of compressor rotor and compressor blades and stator vanes for cracks and bent, broken or missing blades.

d. Exhaust cone and strut assembly for distortion, cracks, and damage.

e. Second stage turbine for cracked, bent, broken or missing blades.

f. Tail pipe for foreign material.

g. Visible portion of combustion chamber outer case for hot spots and cracks.

h. Compressor and turbine for free rotation and no audible evidence of rubbing.

72.7.4 ENGINE MAJOR INSPECTION.

a. Accomplish the Major Periodic Inspection as described in the P & W JT12A Maintenance Manual. (Hot Section Inspection)

b. Remove the following items from the engine:

Engine fuel pump
Engine hydraulic pump (if provided)
Engine starter-generator
Engine tachometer generator

c. Clean the drive spline, apply a coat of MIL-G-21164 or MIL-G-81322 grease to the drive spline, and replace.

72.7.5 ENGINE VIBRATION CHECK.

a. Obtain a Vibration Analyzer Kit, Lockheed, part No. 11109TK.

Note

Before conducting a vibration-level inspection, check the accuracy of the vibration-measuring equipment with suitable calibration equipment, according to the instructions furnished in the kit.

b. Remove the nacelle upper forward access cover from the engine to be checked. Install the transducer on the engine by mounting it on the Pratt & Whitney bracket and attaching the bracket to the

72.7.5 (Cont).

compressor front flange with bolts through the second and third holes to the right of the "12 o'clock" position. Make sure the transducer bracket is positioned on the flange so the centerline of the transducer is perpendicular to the engine axis.

c. Connect the cable to the transducer connector and to the vibration meter, and place the meter in a suitable location in the fuselage. Route the cable clear of the engine intake and exhaust areas, below the wing and forward to the fuselage entrance door, and secure it as necessary with tape or other suitable means.

d. Install the 1-003-110 plug in the No. 1 socket of the input network section of the vibration meter. (The socket is inside the meter.) With a source of 115-volt, 60- or 400-cycle AC power provided, position the selector on the meter as follows:

Input Network	1
Channel	as desired
Range	5
Operation	DX1.0
Power Switch	On*

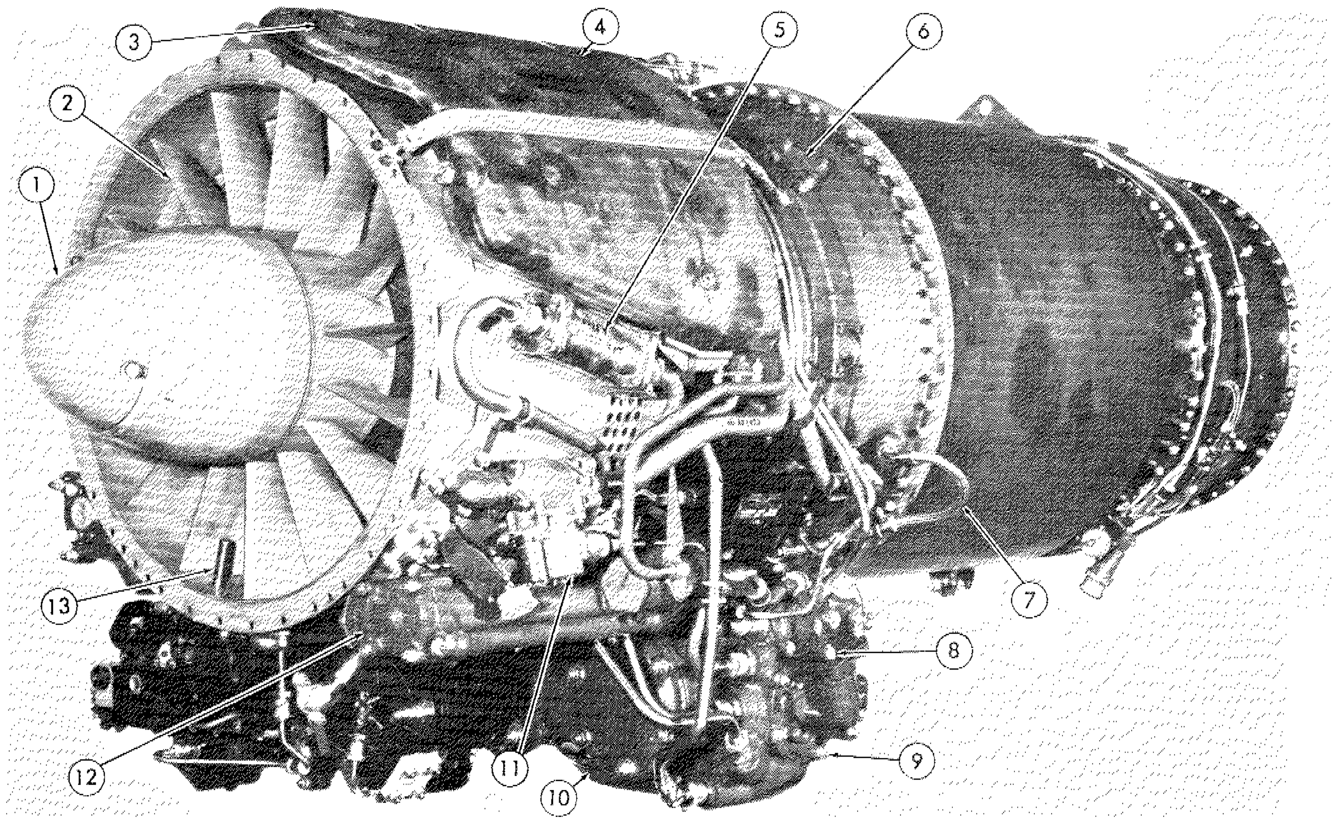
*Allow meter to warm up for at least 20 minutes.

e. Start the engine.

f. Record the vibration levels at each 5 percent RPM from "IDLE" to "TAKEOFF." If the vibration level exceeds 2.5 mils (single amplitude) at or below 60 percent RPM, or if it exceeds 2.0 mils (single amplitude) above 60 percent RPM, the condition is unsatisfactory. Replace the engine.

g. Keep the vibration level record with the engine log book or other engine record forms, regardless of the outcome of the check.

h. Remove the equipment from the airplane, and replace the access panel.



1. NOSE CONE
2. COMPRESSOR INLET GUIDE VANES
3. OIL TANK FILLER
4. OIL TANK
5. SHIPPING ADAPTER FOR FUEL FLOWMETER
6. BLEED AIR MANIFOLD PORT
7. SPARKIGNITER LEAD
8. TACHOMETER DRIVE
9. OIL PUMP
10. ACCESSORY DRIVE GEARBOX
11. ENGINE ANTI-ICING VALVE
12. FUEL-OIL COOLER
13. INLET PRESSURE PROBE
14. TURBINE EXHAUST CASE
15. ENGINE HOIST MOUNT
16. DIFFUSER CASE
17. COMPRESSOR 4TH- STAGE BLEED OUTLETS
18. COMPRESSOR CASE
19. FORWARD ENGINE MOUNT LUGS
20. OIL BREATHER LINE
21. FUEL HEAT VALVE
22. FUEL HEATER
23. FUEL CONTROL
24. FUEL PUMP
25. FUEL PRESSURIZING AND DUMP VALVE
26. SPARKIGNITER
27. MAIN ENGINE MOUNT PAD
28. IGNITION EXCITER
29. COMBUSTION CASE
30. FUEL DRAIN VALVE
31. THERMOCOUPLE HARNESS
32. TURBINE DISCHARGE PRESSURE PROBE

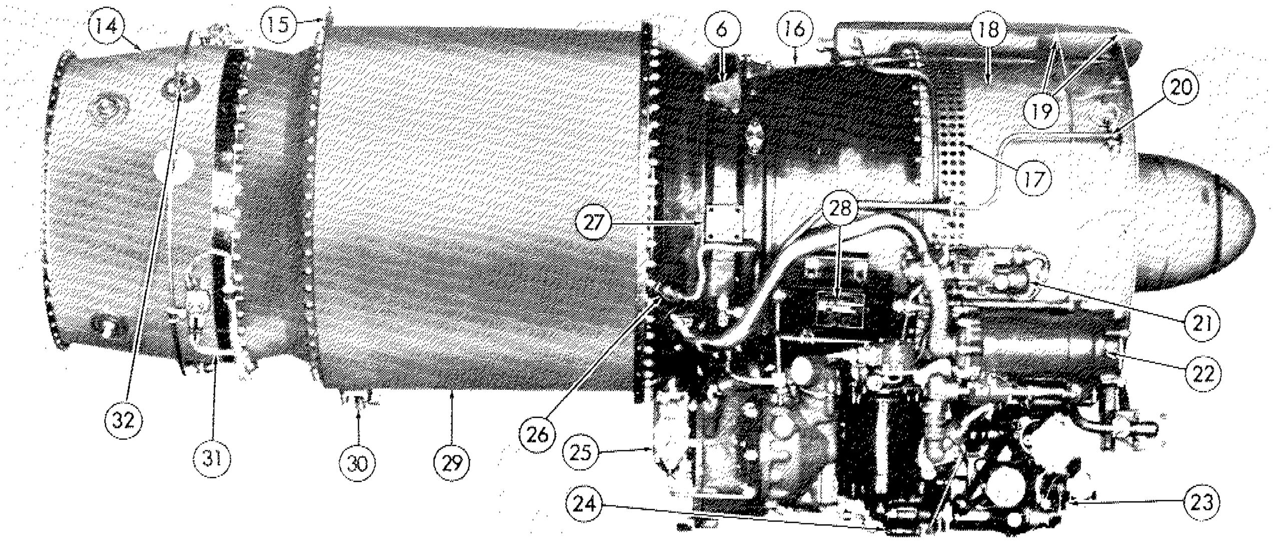


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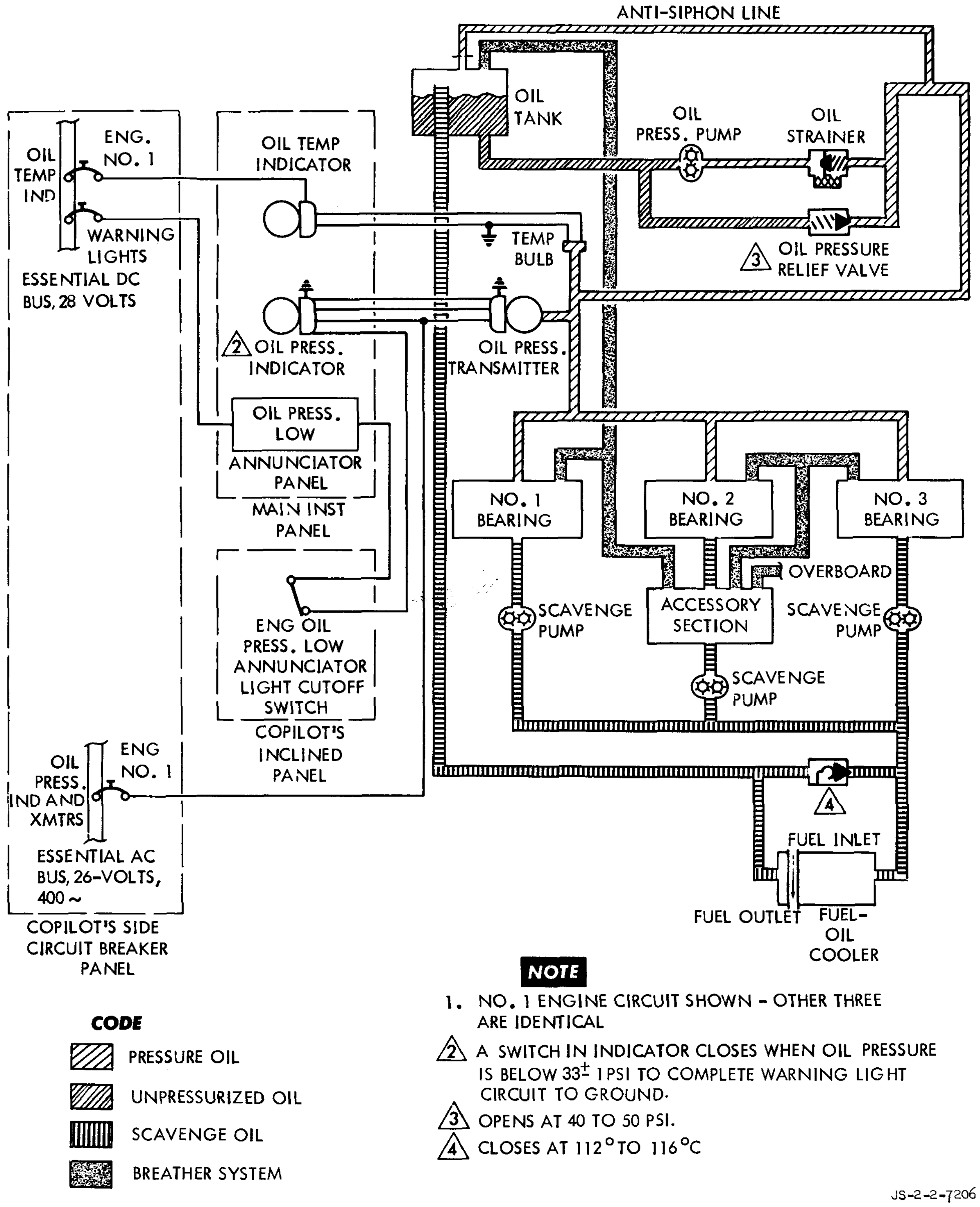
Figure 72-1. Engine and Accessories



NOTE

1. NO. 1 ENGINE CIRCUIT SHOWN – OTHER THREE ARE IDENTICAL

△2 A SWITCH IN INDICATOR CLOSES WHEN OIL PRESSURE IS BELOW 33 ± 1 PSI TO COMPLETE WARNING LIGHT CIRCUIT TO GROUND.

△3 OPENS AT 40 TO 50 PSI.

△4 CLOSES AT 112 °TO 116 °C

Figure 72-2. Engine Oil System Schematic Diagram

JT12A-6 AND JT12A-6A ENGINE PERFORMANCE DATA

(For Reference Only - See Airplane Flight Manual for FAA-Approved Operating Limits.)

OPERATING CONDITION		RATED ENGINE THRUST (SEA LEVEL - STANDARD DAY - STATIC THRUST) (4)		OPERATING LIMITS (2)		
THRUST SETTING	TIME LIMIT (MINUTES)	JT12A-6	JT12A-6A	MAXIMUM OBSERVED EXHAUST GAS TEMP (°C)	OIL PRESSURE (PSIG) NORMAL	OIL TEMP RANGE (°C)
Takeoff (1)	5	3000 lbs	3000 lbs	677	40-50	40-120
Max. Continuous (6)	Contin.	2400 lbs	2570 lbs	577	40-50	40-120
Max. Cruise	Contin.	2250 lbs	2570 lbs	577 (7)	40-50	40-120
Idle	Contin.	-	-	515 (5)	35-50	40-120
Starting	---	-	-	525 (3)	---	---
Acceleration	---	-	-	677 (3)	40-50	40-120

1 The specified takeoff rating is the maximum thrust allowed at or below standard temperature. This rating is to be used for takeoff only.

2 The maximum engine rotor operating speed is 105% (16,700) RPM.

3 These temperatures are time-limited to momentary for starting and except during take-off, to 2 minutes for engine (not airplane) acceleration. Momentary is considered a period not to exceed 5 seconds.
If the starting temperature limit is exceeded, the engine should be shut down and inspected in accordance with the requirements of Chapter 5.

4 Refer to the Airplane Flight Manual (SMR 150) for EPR limits.

5 This temperature is not a limit but rather a guideline value which, if exceeded, may indicate an abnormal condition.

6 This is the maximum thrust that may be used continuously and, is intended only for emergency use of the pilot's discretion.

7 Under the maximum cruise operating condition, a properly functioning JT12A-6 will not normally exceed an EGT value of 530°C. Similarly, a JT12A-6A will not normally exceed an EGT value of 555°C. Whenever these cruise EGT values are observed or exceeded, a flight log entry should be made for maintenance information.

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Figure 72-3. JT12A-6 and JT12A-6A Rated Performance

JT12A-8 ENGINE PERFORMANCE DATA

(For Reference Only - See Airplane Flight Manual for FAA-Approved Operating Limits.)

OPERATING CONDITION		RATED ENGINE THRUST (SEA LEVEL - STANDARD DAY - STATIC THRUST) (4)	OPERATING LIMITS (2)		
THRUST SETTING	TIME LIMIT (MINUTES)		MAXIMUM OBSERVED EXHAUST GAS TEMP (C°)	OIL PRESSURE (PSIG) NORMAL	OIL TEMP RANGE (C°)
Takeoff	5	3300 lbs	718	40-50	40-121
Max. Continuous (6)	Contin.	3000 lbs	655	40-50	40-121
Max. Cruise	Contin.	2800 lbs	655 (7)	40-50	40-121
Idle	Contin.		515 (5)	35-50	40-121
Starting	---		525 (3)	---	---
Acceleration (1)	---		718 (3)	40-50	40-121

1. During an engine acceleration; the EGT should stabilize at or below the noted limit within two minutes of advancing the throttle.
2. The maximum engine rotor operating speed is 105% (16,700) RPM.
3. These temperatures are time-limited to momentary for starting. Momentary is considered a period not to exceed 5 seconds. If the starting temperature limit is exceeded, the engine should be shut down and inspected in accordance with the requirements of Chapter 5.
4. Refer to the Airplane Flight Manual (SMR 150) for EPR limits.
5. This temperature is not a limit but rather a guideline value which, if exceeded may indicate an abnormal condition.
6. This is the maximum thrust that may be used continuously and, is intended only for emergency use at the pilot's discretion.
7. Under the maximum cruise operating condition a properly functioning engine will not normally exceed an EGT value of 620°C. Whenever 620°C or greater is observed, a flight log entry should be made for maintenance information.

JS-1-2-X0/2-7203A

Figure 72-3A. JT12A-8 Rated Performance

ITEM	LOCATION	INSTALLATION TORQUE (INCH-POUNDS)
Fuel flowmeter electrical connection	LH	Safety Wire
Generator ram air tube clamp	LH	30 to 40
Generator overheat electrical connection	LH	25 to 30
Generator power and control leads	LH	3/8-inch nuts 50 to 70 1032 nuts 12 to 15
Tachometer generator electrical connection	LH	Safety Wire
Engine electrical disconnect	LH	Safety Wire
Fuel air bleed valve electrical connection	Bottom	Safety Wire
Fuel inlet hose	Bottom	1/4-inch AN818-4 nut to 40 to 65, 1-inch hose end fitting to 500 to 700, 1/4-28 stud nuts to 65-85
Oil temperature bulb electrical connection	LH	Safety Wire
Clamps from bleed air bellows	Top	30 to 40
Oil pressure transmitter electrical connection	LH	Safety Wire
Fuel pressure switch electrical connection	Bottom	Safety Wire
Fuel differential pressure switch electrical connector	Bottom	Safety Wire
Tachometer generator ground terminal	LH	65 to 85
Hydraulic couplings (Two each on no. 2 and 3 engines only)	Bottom	Tighten nut on end of hose until it bottoms. A sharp rise in torque will be noted at the bottom point. Then apply from 1/6 to 1/3 turn (one to two flats on hex nut).
Exhaust gas temperature electrical connection	LH	Safety Wire
3/8-inch hydraulic pump case drainline	Bottom	Torque until fitting is tight
Pressure ratio hose	Bottom	Torque until fitting is tight
Throttle control telescopic arm (Disconnect from bellcrank)	RH	Standard
Clamps at engine inlet duct (See note following Para 72.5.2c)	Between engine and nacelle	15 - 25

NOTE

1. Torque values for all standard fasteners except those attached directly to the engine are contained in Chapter 51. Fasteners attaching directly to the engine are to be torqued while wet with engine oil, unless otherwise specified, to the following values:

Thread Size	Torque Inch-Pounds (1)	Torque Inch-Pounds (2)
5/16 - 24	125 to 170	160 to 170
1/4 - 28	65 to 85	80 to 85
3/8 - 24	225 to 300	275 to 300

(1) If cotter pin hole alignment is required.
(2) If hole alignment is not required.

2. This figure makes no provisions for non-standard equipment used by some JetStar operator. For example: vibration pickup(s) and oil pressure warning switch.

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Figure 72-4. Engine Connections

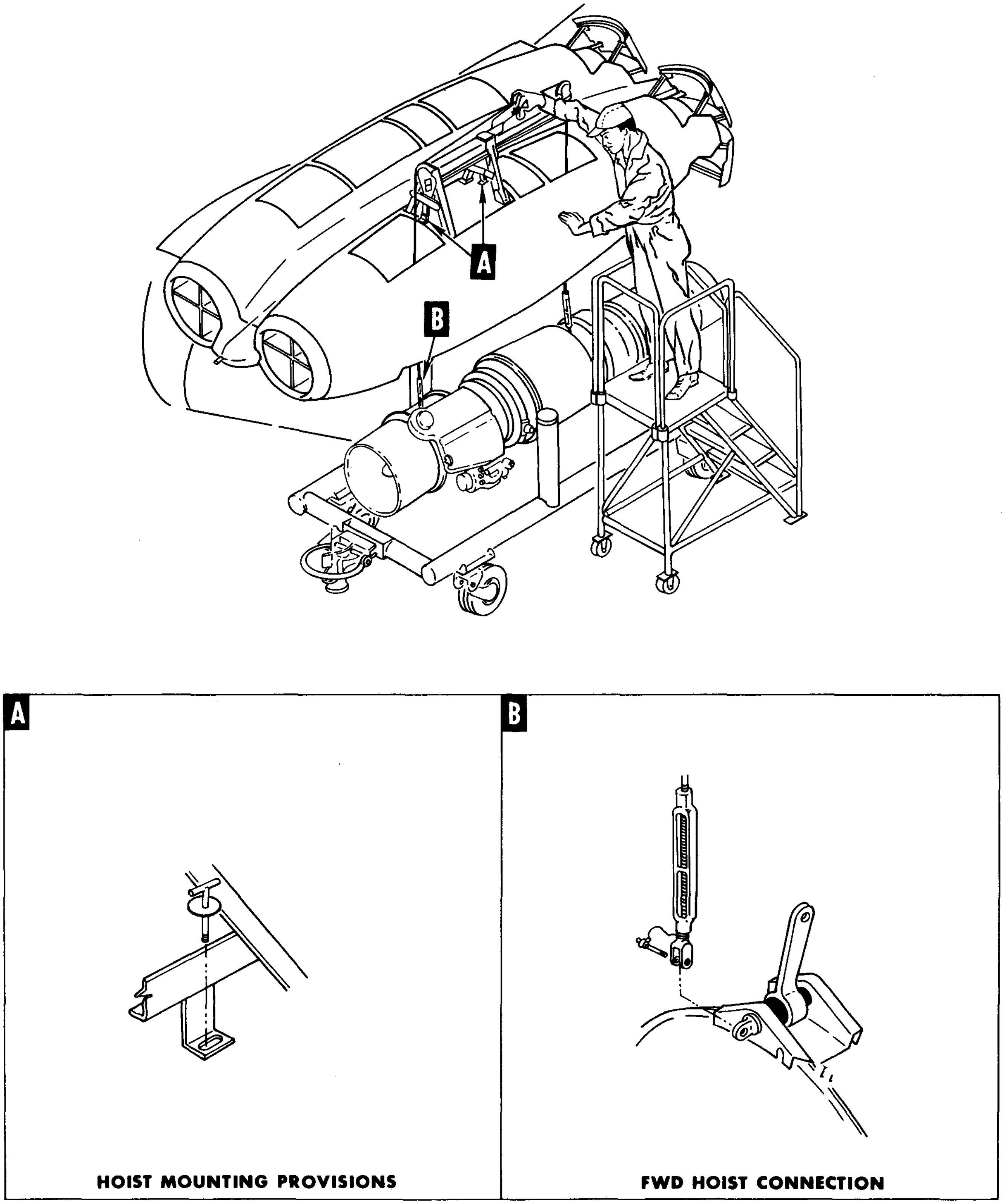

Figure 72-5. Removal of Engine

1. ENGINE S/N __________ POS. # __________ AIRCRAFT # __________ DATE __________

2. ENGINE TOTAL FLIGHT TIME __________ AIRCRAFT TOTAL FLIGHT TIME __________

3. FUEL CONTROL P/N __________ S/N __________

4. OAT __________ °C FIELD P_{am} __________ "hg ENGINE TYPE __________

DESIRED PT5 AND EPR

5. PART POWER P_{t5} MIN. __________ "hg MAX. __________ "hg EPR __________

6. TAKE OFF P_{t5} MIN. __________ "hg EPR __________ "hg

PRIOR TO ADJUSTMENT - PART POWER

7. Pt5 __________ "hg RPM __________ % Tt5 __________ °C

8. EPR __________ FUEL FLOW __________ PPH OIL PRESS __________ PSI OIL TEMP __________ °C

AFTER ADJUSTMENT - PART POWER

9. Pt5 __________ "hg RPM __________ % Tt5 __________ °C

10. EPR __________ FUEL FLOW __________ PPH OIL PRESS. __________ PSI OIL TEMP __________ °C

TAKE-OFF POWER CHECK

11. Pt5 __________ "hg RPM __________ % Tt5 __________ °C

12. EPR __________ FUEL FLOW __________ PPH OIL PRESS __________ PSI OIL TEMP __________ °C

PART POWER SPEED ACCEPTABILITY

13. DATA PLATE SPEED __________ RPM __________ %RPM

14. OBSERVED __________ RPM __________ %RPM

15. SPEED BIAS FACTOR __________

16. CORRECTED __________ RPM

17. __________ RPM (ABOVE, BELOW) DATA PLATE SPEED

18. REMARKS:

Figure 72-6. Engine Trim Worksheet

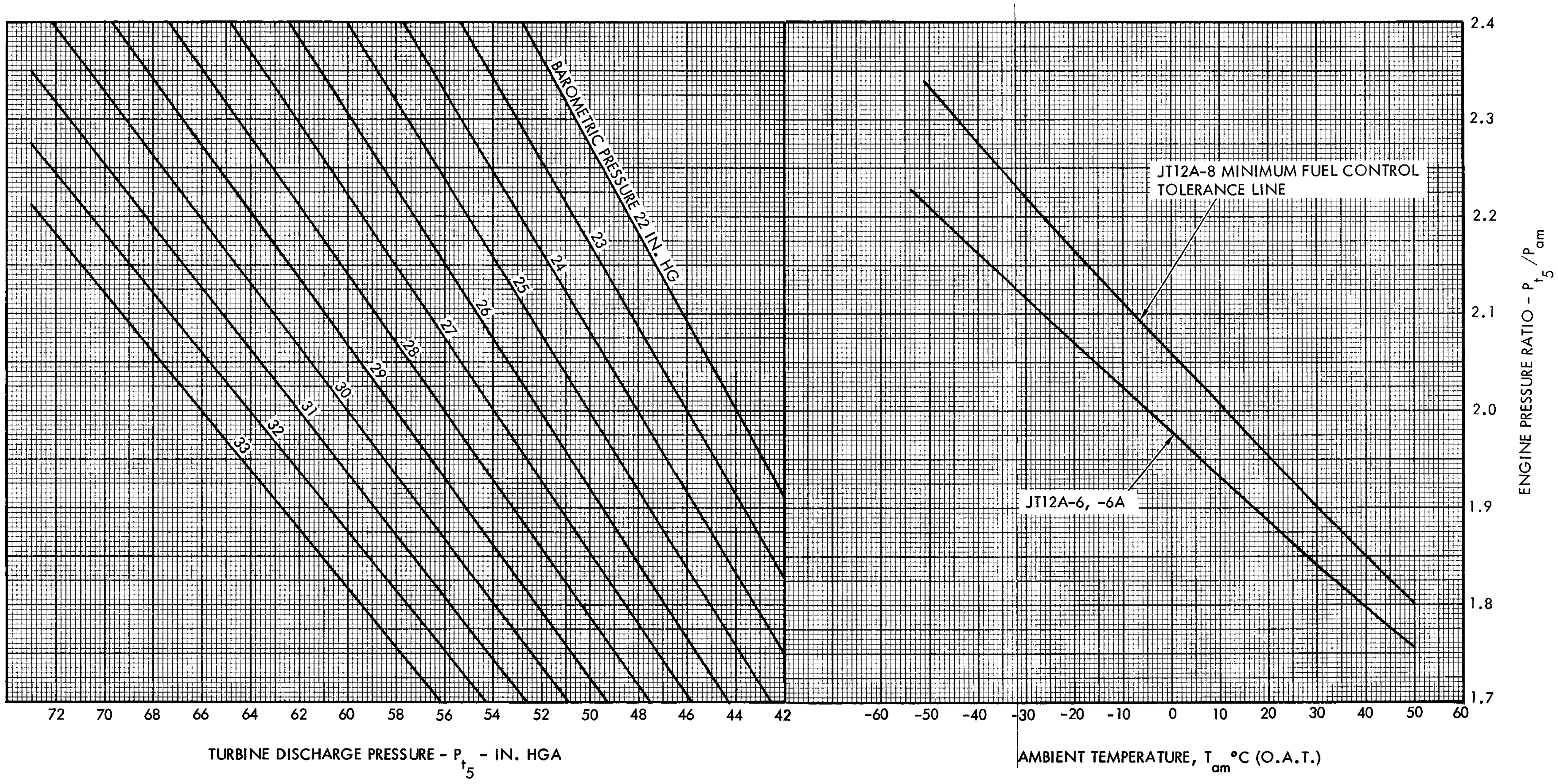

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Figure 72-7. Part Power Thrust Setting Curve

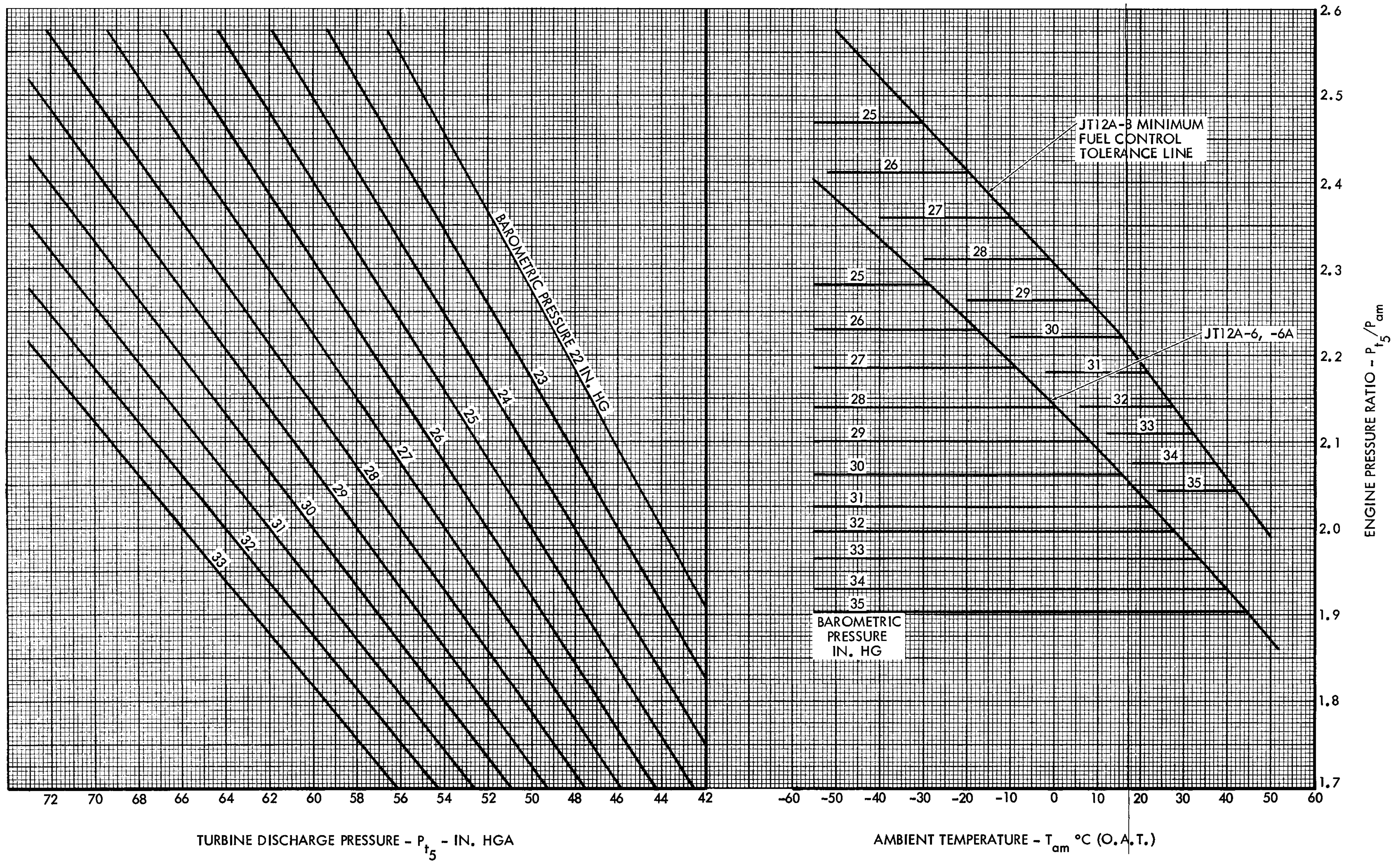
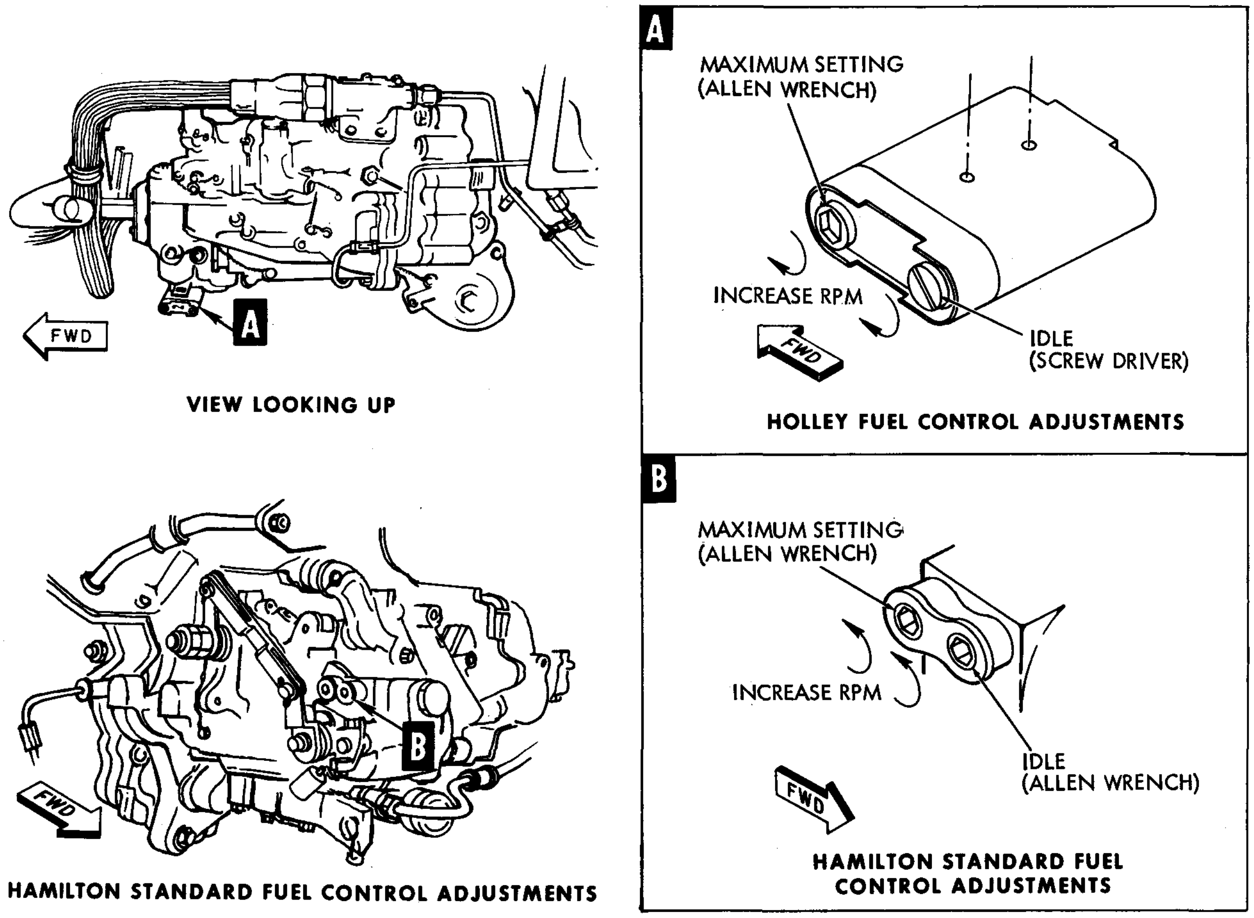


Figure 72-8. Takeoff Thrust Setting Curve



VIEW LOOKING UP

HOLLEY FUEL CONTROL ADJUSTMENTS

HAMILTON STANDARD FUEL CONTROL ADJUSTMENTS

HAMILTON STANDARD FUEL CONTROL ADJUSTMENTS

Figure 72-9. Fuel Control Adjustment

TEMPERATURE - SPEED BIAS CURVE
PART POWER RATED ENGINES.
FOR ALL JT12A AND J60 MODELS

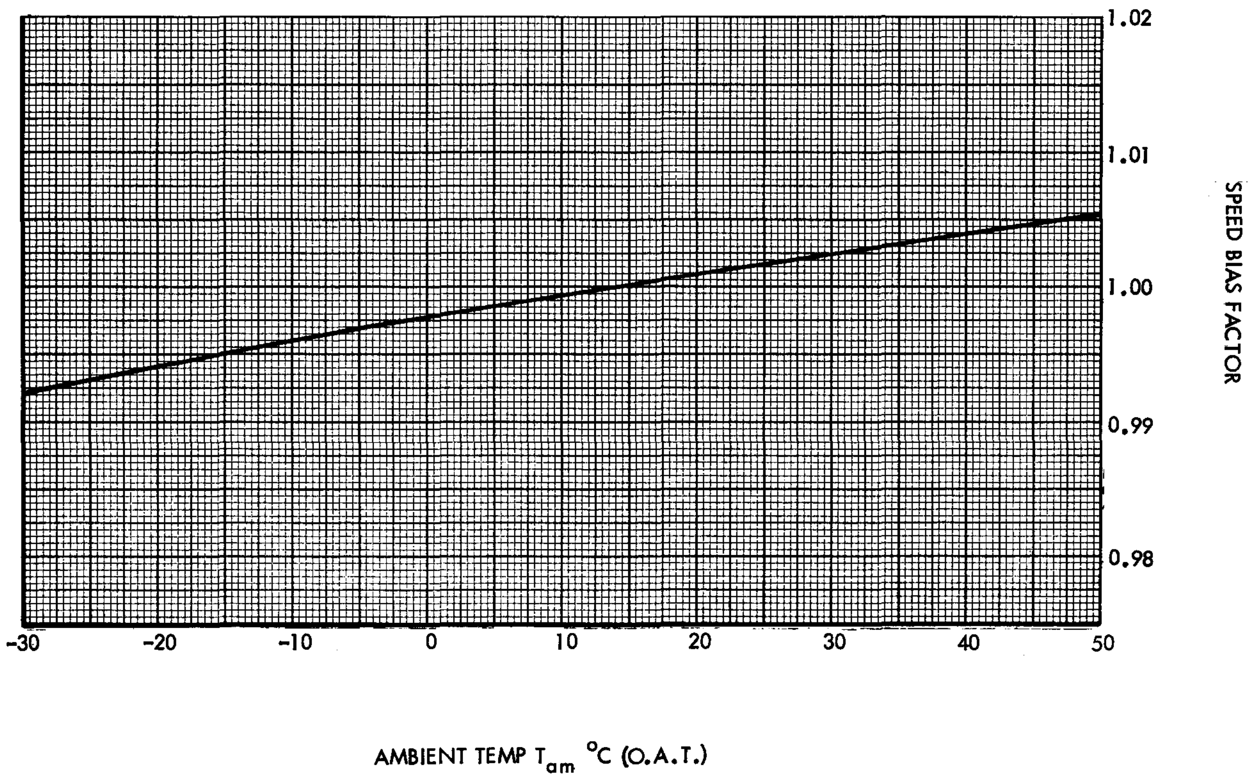


Figure 72-10. RPM Correction Curve
Changed 1 June 1967

MAXIMUM ACCEPTABLE EGT
VS. OUTSIDE AIR TEMPERATURE

STATIC TAKEOFF THRUST
NO AIR BLEED

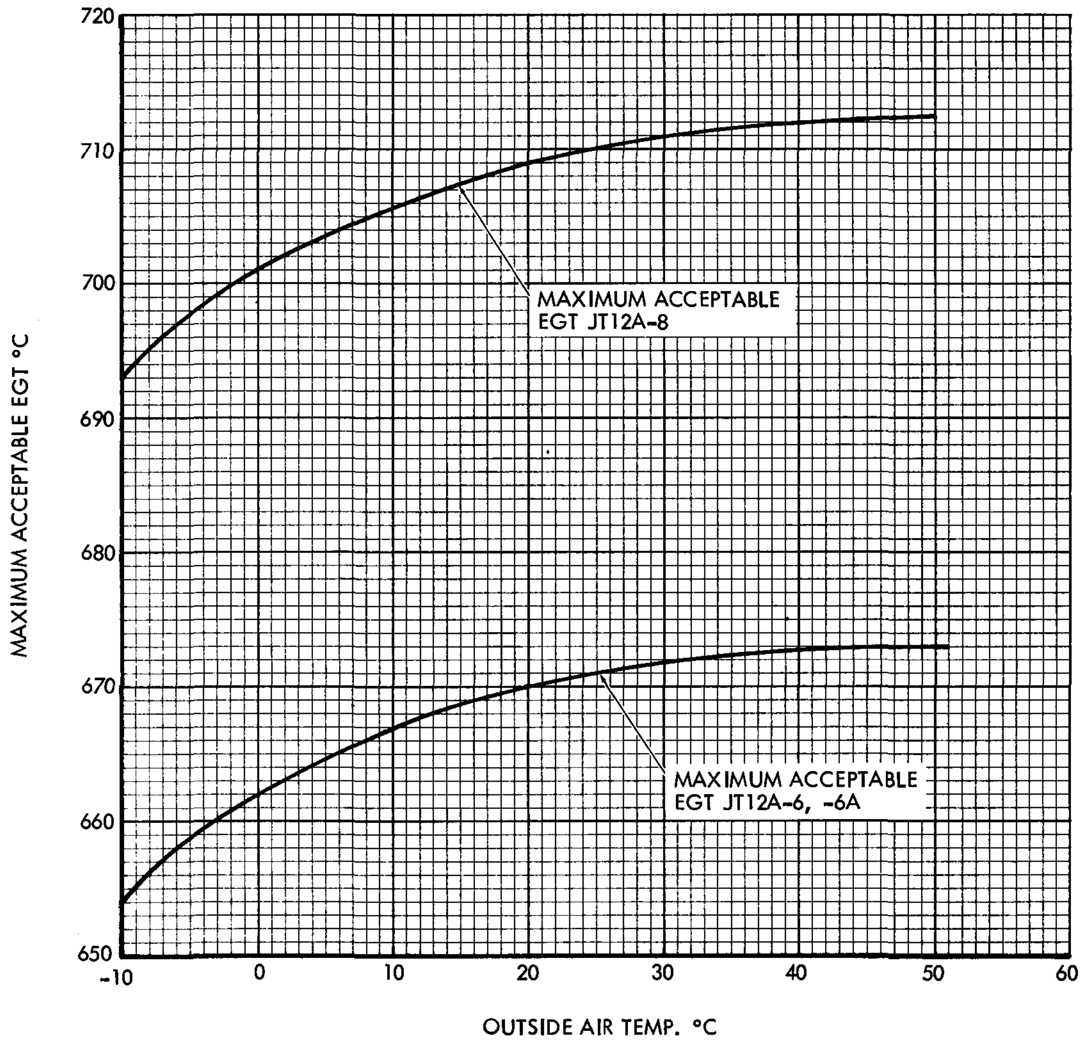


Figure 72-11. Maximum Acceptable EGT Curve

CHAPTER 73
ENGINE FUEL AND CONTROL

Contents Page

DESCRIPTION AND OPERATION 73-1
  Description 73-1
  Engine Fuel Pump 73-1
  Fuel Heater 73-1
  Fuel Heat Valve 73-2
  Fuel Low-Pressure Switches 73-2
  Fuel Filter Clogging Indicator Switch 73-2
  Fuel Control 73-2
  Fuel Flowmeter 73-2
  Fuel-Oil Cooler 73-2
  Pressurizing and Dump Valve 73-2
  Fuel Nozzles 73-2
  Operation 73-2

TROUBLESHOOTING 73-2

REMOVAL AND REPLACEMENT 73-4A

RIGGING AND ADJUSTMENT 73-4A

FIGURES
  73-1 Engine Fuel Flow Schematic Diagram 73-5

73.1 DESCRIPTION AND OPERATION.

73.1.1 DESCRIPTION. (See figure 73-1.) Each engine is provided with an individual fuel system. The four engines are all supplied from the aircraft fuel system which is described in Section 28.0. The principle components of the system on each engine are an engine-driven fuel pump which incorporates an internal fuel filter, a fuel heater, a fuel control, a flowmeter, a fuel oil cooler, a pressurizing and dump valve, and eight fuel nozzles.

73.1.2 ENGINE FUEL PUMP. The fuel pump assembly is mounted on a drive pad on the accessory drive gearbox, and contains the following components: (1) A centrifugal stage for boosting supply pressure about 18 PSI, (2) a 40-micron fuel filter, (3) a fuel filter bypass valve set to open with approximately 10 PSI differential pressure across the filter, (4) a gear stage to boost pressure to about 800 PSI for introduction into the fuel control, and (5) a relief valve across the gear stage set to crack at 825 PSI with full flow at 900 PSI.

73.1.3 FUEL HEATER. A fuel-air heat exchanger is provided to heat the fuel in case of fuel icing. Normally fuel passes through the heater after leaving the centrifugal boost stage of the engine fuel pump. When heat is required, a shut off valve in the air line is opened to admit ninth-stage compressor bleed air. The heater should be cycled on one minute and off 30 minutes when flying in sub-freezing ambient air temperatures or when the FUEL FILTER CLOGGING warning light illuminates.

CAUTION

If fuel heater operation does not extinguish the FUEL FILTER CLOGGING light, fuel contamination should be suspected.

73.1.3 (Cont).

A relief valve, set at about 18 PSI, is provided to bypass the fuel heater in case the fuel flow becomes blocked or restricted within the heater.

73.1.4 FUEL HEAT VALVE. The fuel heat valves are controlled by two switches on the copilot's instrument panel. One switch controls the valves for engines No. 1 and 4, and the other controls the valves for engines No. 2 and 3. When the switches are held in the "ON" position, they energize the open windings of the fuel heat valve motors. Opening the valves permits hot engine bleed air to enter the fuel heater, and energizes a FUEL HEAT ON indicator light for the respective engines, on the copilot's inclined panel. When the control switch is released, the valves return to the closed position to terminate the fuel heating operation.

73.1.5 FUEL LOW-PRESSURE SWITCHES. Two pressure actuated switches, mounted on a bracket aft of the accessory section on the bottom of the engine, are connected to operate a FUEL PRESS LOW light on the annunciator panel and one of four lights on the fuel management panel. One switch senses the pressure of the fuel just before it enters the engine fuel pump, and the other senses the differential pressure across the centrifugal boost stage of the engine-driven pump. When either switch is actuated, either due to low fuel pressure (less than 8 ±1 PSI) at the engine inlet, or due to low differential pressure (less than 8 ±1 PSI) across the centrifugal pump stage, the indicator light will operate. For identification, the indicator light flashes when operated by the differential pressure switch. This same indicator light may also be illuminated by low pressure at a main tank boost pump. This signal is wired through the boost pump control switch, and is inoperative when the boost pump is turned off. For a diagram of these circuits, see figure 28-2.

73.1.6 FUEL FILTER CLOGGING INDICATOR SWITCH. A differential pressure switch is connected to sensing ports on the fuel pump to detect abnormal pressure across the internal fuel filter. If the pressure drop exceeds 7 to 8 PSI, the switch completes a circuit to a FUEL FILTER CLOGGING indicator light on the copilot's inclined panel. This indication is useful to show filter restrictions due to contamination or to ice. It is used in flight as an indicator of the need for fuel heat. Momentary illumination of this light in flight after a boost pump is turned on when the engine is not running sometimes occurs. Such indications characteristically, are not uniform and corrective action should be withheld until some indication is observed when the engine is operating.

73.1.7 FUEL CONTROL. The engine fuel control is hydromechanical. Mechanical movement of the fuel control bellcrank by the throttle linkage causes corresponding changes in fuel output. It is mounted on a drive pad on the engine fuel pump, and receives power through the pump. The fuel control is divided into two major sections: the metering section and the computing section. The basic function of the metering section is to meter the fuel to allow the correct amount to flow to the combustion section according to the amount of thrust selected on the throttle. The computing section contains servo systems to move the fuel metering valve as necessary to control the engine within operating limits. Fuel controls manufactured by the Holley Carburetor Company and the Hamilton-Standard Division of United Aircraft are both used on the JT12 engine. These controls are functionally interchangeable but require different throttle rigging procedures. See Chapter 76 for complete throttle rigging instructions.

73.1.8 FUEL FLOWMETER. A mass fuel flow transmitter is provided downstream of the engine fuel control. See Chapter 77 for information on this transmitter.

73.1.9 FUEL-OIL COOLER. A fuel-oil heat exchanger is provided to permit rejection of engine heat from the oil to the fuel entering the engine. A thermally regulated bypass is provided which is closed when the oil temperature is above 118° (±3°)C (245° (±5°)F).

73.1.10 PRESSURIZING AND DUMP VALVE. In the pressurizing and dump valve, the metered fuel flow from the fuel control is divided into a primary and secondary flow for the eight fuel nozzles. Initially the valve remains closed, sending all fuel to the primary nozzle manifold. At a predetermined pressure, the valve is opened and a portion of the flow is admitted to the secondary nozzle manifold. This valve also provides a means for dumping the fuel from the nozzle manifolds after engine shut down.

73.1.11 FUEL NOZZLES. Eight fuel nozzles are provided to spray fuel into the engine combustion section. To obtain the optimum spray pattern, there is a primary and a secondary fuel flow path through each nozzle. These flow paths are fed by the primary and secondary nozzle manifolds which are connected to the pressurizing and dump valve.

73.2 OPERATION. Refer to Chapter 72 for information on engine operation, and to Chapter 80 for information on the engine starting system.

73.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE FAILS TO LIGHT-OFF WHEN THROTTLE IS ADVANCED TO IDLE.		
Misadjusted throttle linkage.	Check actual throttle linkage movement limits with those presented in the rigging instructions.	Re-rig the engine throttle system. (See Chapter 76.)

73.3 (Cont).

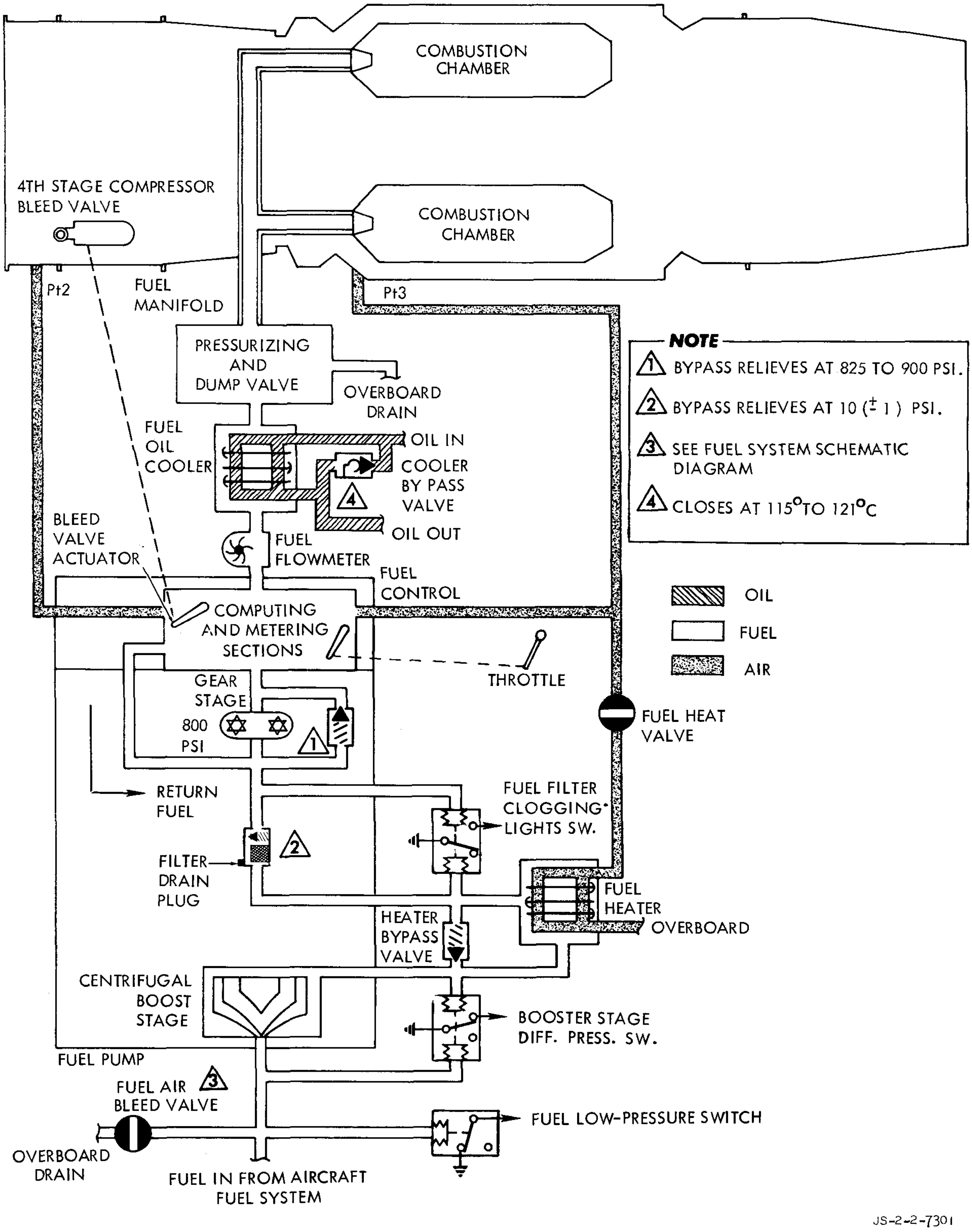
PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE FAILS TO LIGHT-OFF WHEN THROTTLE IS ADVANCED TO IDLE. (Cont)		
Faulty cutoff valve in fuel control.	If there is fuel from the airplane fuel system and the throttle linkage is rigged correctly, the cutoff valve in the fuel control unit is not operating.	Replace the fuel control.
Defective engine-driven fuel pump.	If the fuel flowmeter indicate some fuel flow, but not enough for normal start, the pump may be defective.	Replace the engine fuel pump.
ENGINE LIGHTS-OFF, BUT FAILS TO ACCELERATE TO IDLE.		
Faulty accelerating system in the fuel control unit.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
Clogged fuel control inlet filter screen.	Check the fuel flowmeter. Minimum fuel flow during start should be indicated.	Clean the filter.
ENGINE MAKES A HOT START.		
Minimum flow adjustment too high.	Check fuel flowmeter to determine if a normal starting fuel flow is indicated.	Replace the fuel control.
IMPROPER FUEL FLOW AT ALTITUDE.		
Defective fuel control.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
LOW MINIMUM FUEL AND/OR LOW-SPEED INSTABILITY.		
Idle speed too low.	Check idle RPM.	Adjust the fuel control.
Minimum fuel flow adjustment too low.	If idle speed is at minimum limit, fuel flow is too low.	Replace the fuel control.
LOW MAXIMUM FUEL FLOW AND INSTABILITY.		
Clogged filter screen in fuel control unit.	Clean filter, and check engine operation. If condition persists, fuel control is defective.	Replace the fuel control.
GROUND IDLE SPEED INCORRECT.		
Fuel control adjustment not correct.	Check tachometer idle RPM.	Adjust the engine idle speed.
ENGINE SURGES DURING ACCELERATION, OR FAILS TO ACCELERATE PROPERLY.		
Defective fuel control.	Visually inspect operation of the bleed valve strap while the engine is operating. If satisfactory, the fuel control is faulty.	Replace fuel control.

73.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE SURGES DURING ACCELERATION, OR FAILS TO ACCELERATE PROPERLY. (Cont).		
Clogged filters in engine-driven fuel pump or fuel control.	Inspect filters.	Replace the fuel pump filter cartridge and/or clean the fuel control filter. Check cause of contamination and take necessary corrective action.
ENGINE FAILS TO DECELERATE WHEN THROTTLE IS RETARDED.		
Misadjusted minimum fuel flow setting.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
ENGINE FLAMES OUT DURING DECELERATION.		
Defective fuel control.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
ENGINE CONTINUES TO RUN A SHORT TIME WITH THROTTLE IN THE "OFF" POSITION.		
Incorrect throttle rigging.	Check rigging.	Re-rig throttle system as required.
Defective cutoff valve in the fuel control.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
ENGINE OPERATES AT AN OVERTEMPERATURE CONDITION.		
Improperly trimmed fuel control.	Check engine trim.	Trim the engine.
Defective exhaust gas temperature indicating system.	Troubleshoot the EGT system in accordance with the procedures in Chapter 77.	Repair, adjust, or replace as necessary.
Defective fuel control.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
Internal engine damage.	Perform a visual inspection of the engine.	Replace the engine.
ENGINE OPERATES AT AN OVERSPEED CONDITION.		
Improperly trimmed fuel control.	Check engine trim.	Trim the engine.
Defective fuel control.	Substitute a fuel control known to be in good operating condition.	Replace the fuel control.
Internal engine damage.	Perform a visual inspection of the engine.	Replace the engine.
ENGINE FALSE OR HUNG START		
Electrical Problems.		Refer to Chapter 80.
Defective Fuel Control.	Substitute a fuel control known to be in good operating condition.	Replace fuel control.
Water in Fuel.	Inspect for presence of water in fuel.	Remove water or replace fuel. Changed.

73.4 REMOVAL AND REPLACEMENT. For removal and replacement of engine fuel system components refer to the P&W JT12 Maintenance Manual. For anything not covered in this manual refer to Chapter 71, Paragraph 71.2.3.

73.5 RIGGING AND ADJUSTMENT. Refer to Chapter 72 for engine rigging and adjustments.



NOTE

1. BYPASS RELIEVES AT 825 TO 900 PSI.
2. BYPASS RELIEVES AT 10 (± 1) PSI.
3. SEE FUEL SYSTEM SCHEMATIC DIAGRAM
4. CLOSES AT 115°TO 121°C

Figure 73-1. Engine Fuel Flow Schematic Diagram

CHAPTER 74
IGNITION

As ignition is an integrated part of the engine starting system, it has been combined with the starting system chapter. (See Chapter 80.)

CHAPTER 75
AIR

Engine Anti-Icing is covered under Ice and Rain Protection. (See Chapter 30.)

CHAPTER 76
ENGINE CONTROLS

Contents Page

DESCRIPTION AND LEVER OPERATION 76-1
  Throttle Quadrant 76-1
  Throttles 76-1
  Thrust-Reverser Lever 76-1
  Thrust-Reverser Arming Switch 76-2
  Thrust-Reverser Switch 76-2
  Minimum Cruise Switch 76-2
  Ignition Switch 76-2
  Throttle and Rudder Gust Lock Interconnection 76-2
  Rudder Gust Lock System 76-2
  Throttle Cables 76-2
  Changeover Boxes 76-2
  Push-Pull Cable System 76-2

TROUBLESHOOTING 76-2
  Throttle Control System 76-2

REMOVAL AND REPLACEMENT 76-2
  Teleflex Push-Pull Control 76-2
  Removal 76-2
  Replacement 76-2

ADJUSTMENT AND RIGGING 76-3
  Rod Adjustment 76-3
  Cable Ball Position 76-4
  Throttle Lever Clearance 76-4

THROTTLE RIGGING 76-4
  Holley Fuel Controls 76-4
  Hamilton Standard Controls 76-4
  Engine Trim 76-4
  Knob Check 76-4
  Thrust Reverser Throttle Adjustment 76-4

THROTTLE QUADRANT ADJUSTMENT 76-5
  Throttle and Thrust-Reverser Rigging Instructions 76-5
  Throttle-Activated Switches 76-5

FIGURES

76-1. Engine Controls Components Locations 76-7
76-2. Throttle Control Adjustment 76-8
76-3. Throttle and Thrust-Reverser Lever Rigging Instructions 76-10
76-4. Throttle Switch Adjustment 76-11
76-5. Throttle Knob Check 76-12

76.1 DESCRIPTION AND OPERATION.

76.1.1 DESCRIPTION. (See figure 76-1.) The engine controls consist of a throttle quadrant, four changeover boxes, cables between the throttle quadrant and the changeover boxes, and push-pull cables from the changeover boxes to the engine fuel control.

76.1.2 THROTTLE QUADRANT. Four throttles, each having a thrust-reverser arming switch, a thrust-reverser switch, a minimum cruise switch, an ignition switch, and a thrust-reverser control lever are mounted in the throttle quadrant.

76.1.3 THROTTLES. Throttle movement provides an "OFF" position at which the engine fuel is shut off, an "IDLE" position which furnishes idle fuel flow, and an increasing power range up to the "TAKE-OFF" position. At the "IDLE" position, when being retarded, the throttle contacts an idle stop and must be raised to retard the throttle to "OFF." On aircraft 5078, 5092, and up, an "Off" detent is provided. On these aircraft it is necessary to raise the throttle to move from "Off" to "Idle." On earlier aircraft some spring back may be expected when the throttle is placed in the "Off" position, and the throttle should be held in the "Off" position when starting engines (See Note in paragraphs 72.2.2.1 and 72.2.2.2).

76.1.4 THRUST-REVERSER LEVER. A thrust-reverser lever is mounted on the forward side of

76.1.4 (Cont).
each throttle. Because of the construction of the reverser lever, the throttle must be in the "IDLE" position before the thrust-reverser lever can be raised to the reverse thrust position. Additional raising of the lever mechanically moves the throttle cables to increase engine power.

76.1.5 THRUST-REVERSER ARMING SWITCH. There is one thrust-reverser arming switch per throttle. This arming switch arms the thrust-reverser circuit when the thrust-reverser lever is actuated with the throttle in "IDLE" position.

76.1.6 THRUST-REVERSER SWITCH. A thrust-reverser switch assembly, containing two thrust-reverser switches, is mounted on each throttle. The switches energize the thrust-reverser circuit when the thrust-reverser lever is pulled past the point of arming switch actuation.

76.1.7 MINIMUM CRUISE SWITCH. A minimum cruise switch assembly, containing two minimum cruise switches, is mounted on each throttle. These switches energize a circuit to provide an audible signal in the interphone system when the throttle is retarded to approximately 78 percent RPM, with the landing gear in the retracted position. The switches operate together and are actuated by a common actuator.

76.1.8 IGNITION SWITCH. A throttle ignition switch on each throttle breaks the ignition circuit when the throttle is in the "OFF" position, and energizes the circuit during a starting cycle when the throttle is advanced approximately 3 degrees.

76.1.9 THROTTLE AND RUDDER GUST LOCK INTERCONNECTION. With the rudder gust lock engaged, the movement of all four throttles is mechanically restricted to a travel of approximately 13 degrees forward of the "IDLE" position. It is not possible to obtain takeoff power with the rudder gust lock engaged. Disengagement of the rudder gust lock permits full throttle travel.

76.1.10 RUDDER GUST LOCK SYSTEM. This is operated by a T-handle on the right side of the center console. The T-handle is connected to a push-pull cable assembly, which rotates a torque tube below the throttle quadrant through a bellcrank. Two stops mounted on the torque tube are rotated upwards when the gust lock handle is pulled. This places the stops on the torque tube in the path of a stop on each throttle drive pulley, preventing throttle movement beyond the 13-degree position. Each torque tube stop limits the movement of two engine throttles. The system is such that the rudder gust lock cannot be engaged while any throttle is positioned forward of the 13-degree position, nor while any thrust reverser lever is positioned to provide a corresponding engine power output. The system is spring-loaded towards the disengaged position by a tension spring acting directly on each torque tube stop.

76.1.11 THROTTLE CABLES. The cable system is of the conventional two-cable tension type, supported by a series of pulleys. These cables are routed from beneath the throttle quadrant to changeover boxes in the aft compartment.

76.1.12 CHANGEOVER BOXES. Changeover boxes in the aft compartment convert the two-cable system of controls into a one-cable push-pull system. The changeover box consists of an input pulley and a drive wheel attached to the pulley. The outer periphery of the drive wheel is serrated to mesh with a helical cable, and provides the driving force for the push-pull cable. Two changeover boxes are installed on each side of the aft compartment. Each box serves one engine.

76.1.13 PUSH-PULL CABLE SYSTEM. A helical cable, a conduit, and a telescopic drive unit compose the push-pull cable system for each engine. The conduit is routed from the changeover box, through the nacelle box beam, to the telescopic drive unit, which is attached to a drive arm on the engine fuel control. The helical cable is attached to the slider of the telescopic drive unit and is passed through the conduit and engaged with the drive wheel of the changeover box.

76.2 TROUBLESHOOTING.

76.2.1 THROTTLE CONTROL SYSTEM. Problems with the throttle control system will probably be in system misadjustment rather than equipment failures. If the throttles become difficult to operate, either because of too little or too much effort required to move them, rerig the system in accordance with the adjustment procedures given in this chapter. Broken or obviously defective components should be replaced with serviceable spares.

76.3 REMOVAL AND REPLACEMENT.

76.3.1 TELEFLEX PUSH-PULL CONTROL.
(See figure 76-1.)

76.3.1.1 REMOVAL. Locate the throttle cable turnbuckles in the aft end of the cabin just forward of the pressure bulkhead. Loosen and remove the two turnbuckles in the cables going to the changeover box (12) affected. This will allow the changeover box to rotate fully and the flex cable may be removed by pulling the telescopic drive unit (14) at the engine and through the conduit (13). Remove the telescopic drive unit from the flex cable.

76.3.1.2 REPLACEMENT.

CAUTION

Damage can be done if the throttles are operated with more than 13 pounds force when the system is not fully rigged.

a. Clean the flex cable with alcohol, or with trichloroethylene vapor.

CAUTION

Do not use other cleaners, such as acid-type agents. They may affect the solid film lubricant.

76.3.1.2 (Cont).

b. Disconnect the conduit (13) at the changeover box (12), and remove the short piece of conduit on the opposite (forward) side of the changeover box.

c. Examine the clean flex cable. It will be noted that one end has about two feet treated with a dry film lubricant. The cable should be installed so that this lubricant will be at the engine end of the cable, and the other end will be at the changeover box. Examine also the condition of the cable ends. They should be rounded and smooth. If not, correct this by grinding.

d. The cable may be fed into the conduit from either end. A light coating of Dow Corning Moly Kote 33, Type II (MIL-G-46886, Type II) grease, should be applied to the untreated portion of the cable as it is fed into the conduit. The force required to move the flex cable through the conduit should not exceed 3.25 pounds.

e. Connect the telescopic drive unit (14) to the engine end of the cable. Tighten the clamping screw to a torque of 40 inch-pounds maximum. The cable should be visible through the lower inspection hole.

Note

On installations which have been in service for some time, the rod end bearing of the telescopic unit may have worn excessively. This can cause throttle alignment problems and should be checked if alignment difficulty has been noted. No definite wear tolerances are established, but any radial clearance greater than 0.002 inch should be investigated. A clearance no greater than 0.020 inch is permissible if there is no difficulty in throttle alignment. If throttle alignment presents a problem, these bearings should be checked and possibly replaced.

f. Connect the telescopic drive unit (14) to the fuel control bellcrank (15). Make sure the bolt is installed with the head away from the engine.

g. Install the part power stop (see Chapter 72) on the fuel control, and position the fuel control bellcrank (Item 10, figure 76-2) so that the 11.34 dimension shown in figure 76-2 is obtained. Replace the part power stop in its stowed position. The 11.34 dimension is for the initial setting only. Small variations from this figure upon completion of rigging are insignificant.

h. Rotate the fuel control bellcrank to the point where the fuel control is against the "OFF" stop. On engine No. 2, this is approximately 30° aft of vertical, and on the other engines it is approximately 30° forward of vertical.

i. With the fuel control held in the "OFF" position, feed the flex cable into the changeover box (12), and engage the conduit nut. When this nut is tight, the position of the cable ball retainer should be on, or slightly aft, of a vertical centerline through the changeover box. This point is shown in Detail C of figure 76-2. It may be necessary to remove the flex cable several times to get the ball retainer in the proper location. The free end of the flex cable should extend 4.63 (±.06) inches forward of the changeover box, as shown in Detail C of figure 76-2. This may be adjusted by loosening the cable retaining bolt (Item 9, figure 76-2) and rotating the flex cable within the conduit. If a new cable is being installed, it may be necessary to cut the cable and grind the end to a rounded and smooth shape. The cable end should be visible, as indicated in paragraph (e), after these adjustments are complete.

j. Replace the short piece of conduit on the forward side of the changeover box.

k. Connect throttle cable turnbuckles in the aft end of the cabin.

Note

Adjusting rods should be set in accordance with paragraph 76.4.1, if required, prior to step (l).

l. With the cable tension in accordance with the chart in figure 76-2, and with the engine control lever against the full power stop, adjust the turnbuckles to obtain 0.050 clearance (as shown in Detail B, figure 76-2) at the pilot's throttle lever. Use feeler gauge to measure clearance.

m. Repeat the above steps, as required, for each throttle system. After completion of step (l) for all four positions, the four throttle knobs should align within 0.06 inch. The force required to operate the throttle in the idle range should not exceed 2.55 pounds, and in the full power range, it should not exceed 5.5 pounds.

Note

Pilot's throttle movement is 17° between "OFF" and "IDLE" and 71° between "IDLE" and "TAKEOFF."

76.4 ADJUSTMENT AND RIGGING. (See figure 76.2.)

Note

The adjustments presented in paragraph 76.4.8.1 are required only when the pilot's throttle quadrant is disassembled. Adjustments in paragraph 76.4.8.2 are required only when quadrant switches need adjustment. If either of these is required, it should be accomplished prior to adjustments in 76.4.1 through 76.4.7.

76.4.1 ROD ADJUSTMENT. With no tension on the throttle cables, install a rig pin (2) through the bracket

76.4.1 (Cont).

and pulleys (detail A). Adjust length of rods (1) so that all throttle levers are in the idle stops. Remove rig pin.

76.4.2 CABLE BALL POSITION. If the cable ball is not in position on or slightly aft of a vertical centerline through the changeover box, as shown in detail C of figure 76-2, the Teleflex flex cable should be removed and the changeover box reclocked by accomplishing paragraphs 76.3.1.2 (i) through (m).

76.4.3 THROTTLE LEVER CLEARANCE. The 0.050 minimum clearance of detail B, figure 76-2, should be readjusted, as required, in accordance with paragraphs 76.3.1.2 (l) and (m).

76.4.4 THROTTLE RIGGING.

76.4.4.1 HOLLEY FUEL CONTROLS. After the above adjustments have been made and checked, the engine power lever should be rigged with the pilot's lever in the idle detent. Use Holley tool Part No. AT1890 to determine the idle position accurately. Adjust the serrated plate under the fuel control bellcrank arm (10, figure 76-2). This should place the fuel control pointer on or slightly above the "IDLE" mark.

76.4.4.2 HAMILTON STANDARD CONTROLS. After the adjustments detailed in paragraphs 76.4 through 76.4.3 have been made and checked, the cockpit lever should be moved against the "OFF" stop while holding a .005 feeler between the fuel control "Off" stop and index pin. The feeler should be locked or tight indicating .005 inch or less clearance. Next, move the cockpit lever forward and back against the "IDLE" stop. In this position the "Go" (12°) end of Pratt & Whitney Tool No. PWA13821 should fit in between the fuel control "Off" stop and index pin. (Adjustment to obtain the two above conditions can be accomplished at the serrated plate as shown in Detail E of figure 76-2.) The Pratt and Whitney tool should then be turned over and the "No Go" (16°) end applied. This end should not fit in this same position.

a. When the above checks are made, the fuel control bellcrank arm (10) should be held with a positive force to remove any throttle system backlash that may be present. This force should be applied in the CW (Increased Power) direction for the "Off" stop check and the "No Go" check, and in the CCW (Decreased Power) direction for the "Go" check. The cockpit lever should be held firmly against the appropriate stop simultaneously.

b. It is expected that something more than zero clearance will usually be required for the "Off" stop check in order to meet the "Go" requirement at "IDLE."

CAUTION

When working on the engine fuel control be sure not to disturb the adjustment of the quadrant or the quadrant pointer. Mislocation of these parts will require recalibration of the fuel control by the manufacturer, or at a qualified repair facility

76.4.5 ENGINE TRIM. At this point, accomplish engine trim in accordance with Chapter 72.

76.4.6 KNOB CHECK. Following engine trim, make the following check. With one or more engines operating at a stabilized idle, move one of the pilot's throttle levers slowly toward cut off and observe the RPM. The RPM should decrease rapidly when the lever is moved as much as one knob diameter below idle, and before the lever reaches a point one knob diameter above cut off. See figure 76-5.

CAUTION

During the above check, the throttle should be quickly moved to cut off after a drop in RPM, or if any rise in EGT is observed. In no case should the throttle be advanced after a movement below the idle stop, as this is likely to cause engine overtemperature.

76.4.7 THRUST REVERSER THROTTLE ADJUSTMENTS.

WARNING

Keep all personnel clear of the thrust-reversers during the following procedure.

a. Remove the thrust-reverser ground locks. Connect external electrical and hydraulic power to the airplane. Engage all REVERSE THRUST circuit breakers.

b. Retard the throttles to "IDLE," and actuate the thrust-reverser levers to their full travel stops. The thrust-reversers should extend, the pointers on the engine fuel controls should move to the part-power position, and the thrust-reverser lever lever knobs should align within 0.12 inch.

76.4.7 (Cont).

c. Move the thrust-reverser levers forward until the minimum detents are contacted. The thrust-reverser should remain extended. Move the thrust-reverser levers fully forward. The thrust-reversers should retract.

d. Move the throttles 10 degrees to 70 degrees forward of "IDLE." Apply 7.5 pounds maximum force to each thrust-reverser lever, in the reverse-thrust direction. The thrust-reversers should not be actuated.

e. Engage the rudder gust lock. Apply 15 pounds maximum force to each throttle in the forward direction, and check that it does not advance more than 13 degrees forward of "IDLE."

f. Repeat steps (a) through (e) for each of the other throttles. Retard all throttles to "OFF."

g. With engines operating after a five-minute stabilization run, check reverse-thrust operation, and adjust the thrust-reverser lever stop to achieve a maximum reverse engine RPM of 91 to 93 percent for the JT12A-6, -6A or 89 to 90 percent for the JT12A-8.

h. Restore the airplane to normal.

76.4.8 THROTTLE QUADRANT ADJUSTMENT.

76.4.8.1 THROTTLE AND THRUST-REVERSER RIGGING INSTRUCTIONS.

(See figure 76-3.)

a. Disconnect external hydraulic power from the airplane, pull out the thrust-reverser circuit breakers, and bleed off thrust-reverser accumulator pressure.

b. Remove the throttle quadrant cover to gain access to the inside of the quadrant.

c. Adjust the link adjusting pin (3) to make an angular distance of 131 (±0.25) degrees between the throttle lever (2) and the secondary throttle lever (4).

d. Move the thrust reverser lever (1) toward the reverse position until the detent spring (7) is disengaged. Attach a spring scale to the knob on the lever and measure the force required to move the lever toward the forward thrust position. The following minimum forces should result:

(1) 0.2 pound to 0.6 pound before the detent spring is contacted.

(2) 3.5 pounds to 5.0 pounds as the detent spring is being deflected.

(3) 0.5 pounds to 1.2 pounds after the detent spring has been deflected and the linkage is approaching the on-center position.

(4) 2.25 pounds to 2.75 pounds as the linkage deflects in passing over center. Loosen the lock nut (5), and adjust the set screw (6) to obtain the required force. With the linkage in the over-center position, check that the throttle (2) slides freely.

Note

A smoother and more positive action between the throttle and thrust-reverser systems can be obtained, if desired. Obtain this by adjusting the set-screw (6) to get a force of 3.50 to 4.00 pounds as the linkage passes over center. If this increased force causes binding of the throttle, reduce the force in 0.50-pound increments until the throttle slides freely. Do not decrease the force below 2.25 pounds.

e. Replace the throttle quadrant cover, and restore all airplane systems to normal.

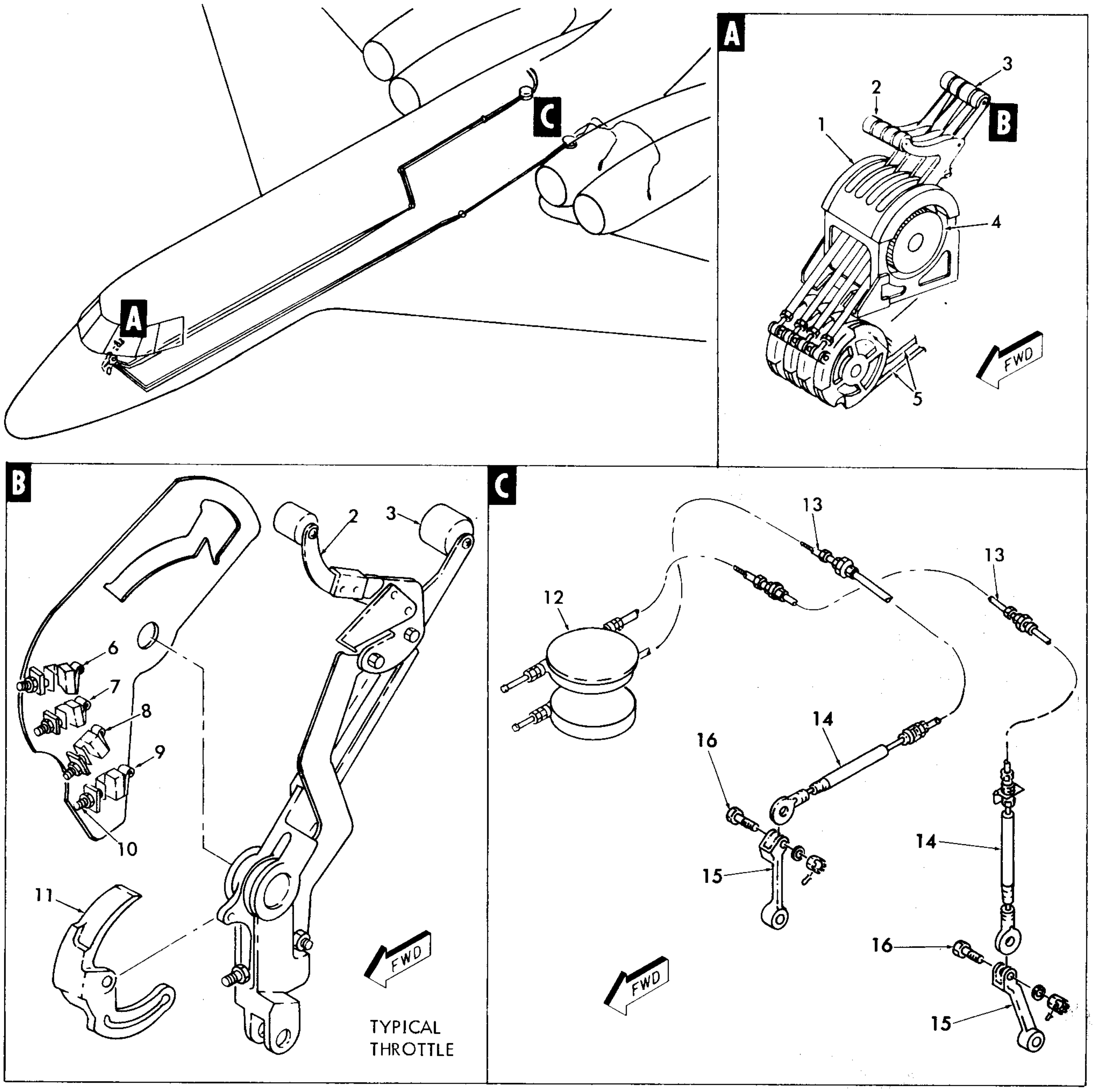
76.4.8.2 THROTTLE-ACTIVATED SWITCHES. (See figure 76-4.) Before attempting to adjust any of the throttle-actuated switches in the throttle control quadrant, make sure the throttle control system and the throttle levers have been adjusted correctly in accordance with the procedures in this chapter. No adjustment is provided to move the switches angularly around the throttle shaft. This is because correct throttle control rigging provides correct switch timing. But roller wear and/or loss of actuator spring tension in the switches themselves may prevent positive opening or closing of the respective circuits at the corresponding throttle lever positions. Inward or outward movement of the switches by means of the switch adjustment provisions compensates for this, without affecting switch timing. Adjust the switches as follows:

a. Loosen the screw and nut which secure the switch to the throttle track assembly.

b. Loosen the locknut on the switch adjusting screw; and rotate the screw by the screwdriver slot in its end, to obtain correct switch positioning to correspond with the existing throttle position. Tighten the locknut and the attaching screw and nut.

c. Connect external power to the airplane and check throttle switch operation.

d. Reinstall the throttle quadrant cover, and disconnect electrical power from the airplane.



1. THROTTLE QUADRANT
2. THRUST REVERSER LEVERS
3. THROTTLES
4. FRICTION WHEEL
5. THROTTLE CABLES
6. THRUST REVERSER SWITCH
7. THRUST REVERSER ARMING SWITCH
8. MINIMUM CRUISE SWITCH
9. IGNITION SWITCH
10. SWITCH ADJUSTMENT SCREW
11. SWITCH ACTUATING CAM
12. CHANGE-OVER BOX
13. TELEFLEX CABLE CONDUIT
14. TELESCOPIC DRIVE UNIT
15. FUEL CONTROL BELL CRANK
16. ⚠1 BOLT, WASHER, NUT, AND COTTER-PIN

NOTE

⚠1 INSTALL BOLT HEAD AWAY FROM ENGINE

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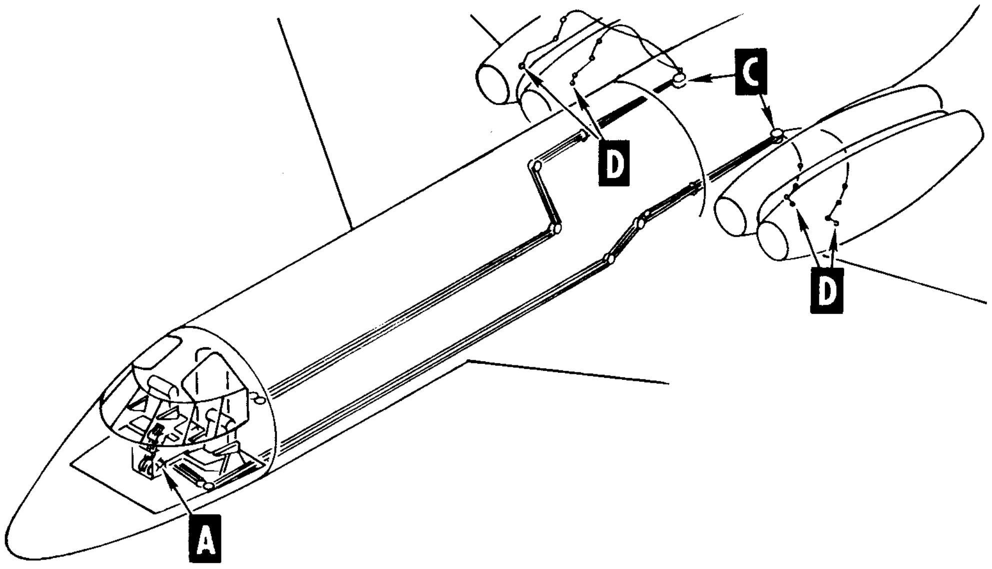
Figure 76-1. Engine Controls Components Locations



1. ADJUSTING ROD
2. RIG PIN
3. STOP BUSHING
4. FULL POWER STOP
5. SAFETY WIRE HOLE (TYPICAL FOR ALL ENGINES)
6. LINE BETWEEN LUG CENTER AND CENTER OF TELEFLEX CONTROL BOX
7. SWAGED BALL ON TENSION CABLE
8. SWIVEL JOINT
9. CABLE RETAINING BOLT
10. FUEL CONTROL BELL CRANK ARM
11. TELESCOPIC DRIVE ARM
12. CONNECTING LINK
13. ENGINE CONTROL LEVER
14. ENGINE IDLER LEVER
15. FUEL CONTROL "OFF" STOP
16. FUEL CONTROL INDEX PIN

NOTE

△1 IF NEW CABLE IS BEING INSTALLED, CUT CABLE TO 4.63 ± .06 AFTER ENGINE CONTROL IS PROPERLY ADJUSTED AND SET IN "OFF" POSITION. ROUND OFF END OF CABLE TO PREVENT DAMAGE TO CONDUIT LINING.

△2 .050 INCH CLEARANCE.

△3 BOLT AT THIS POINT SHOULD BE INSTALLED WITH NUT ON ENGINE SIDE OF BELL CRANK ARM

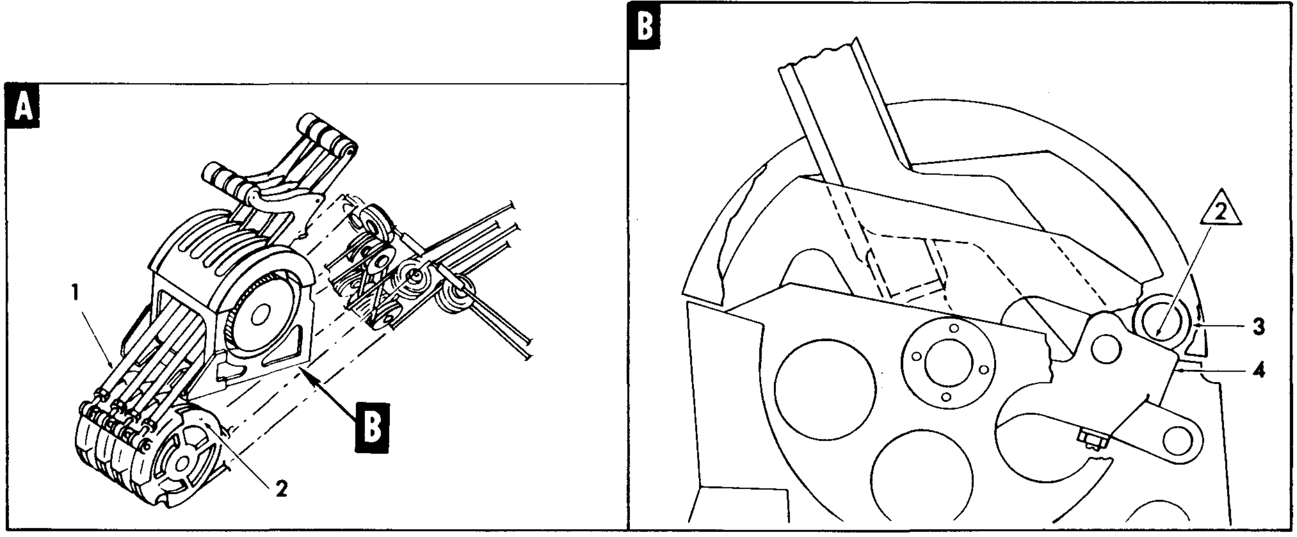


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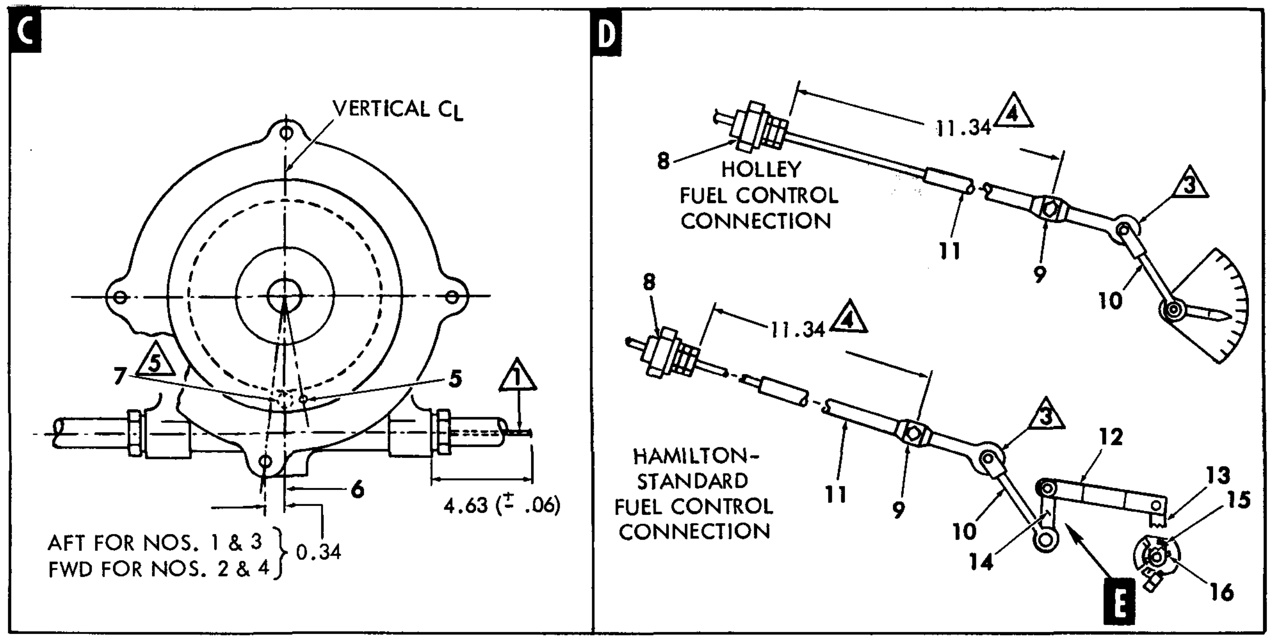
Figure 76-2. Throttle Control Adjustment (Sheet 1 of 2)

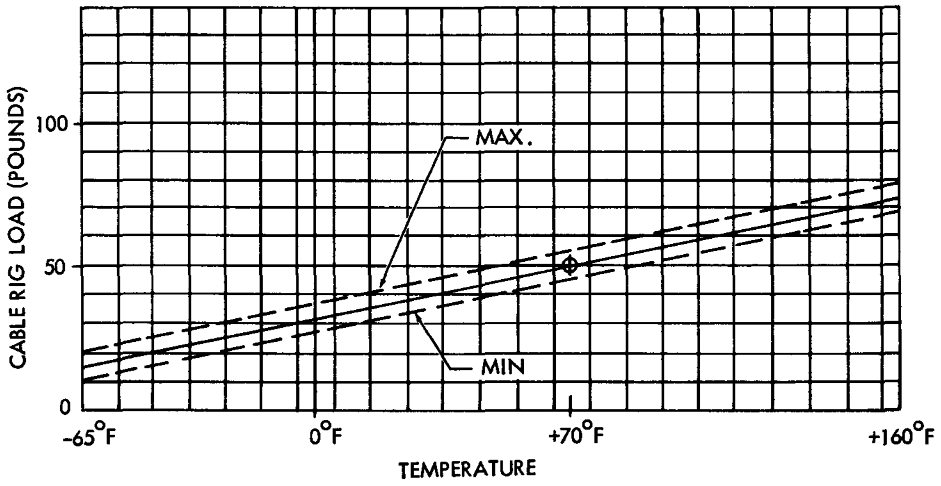


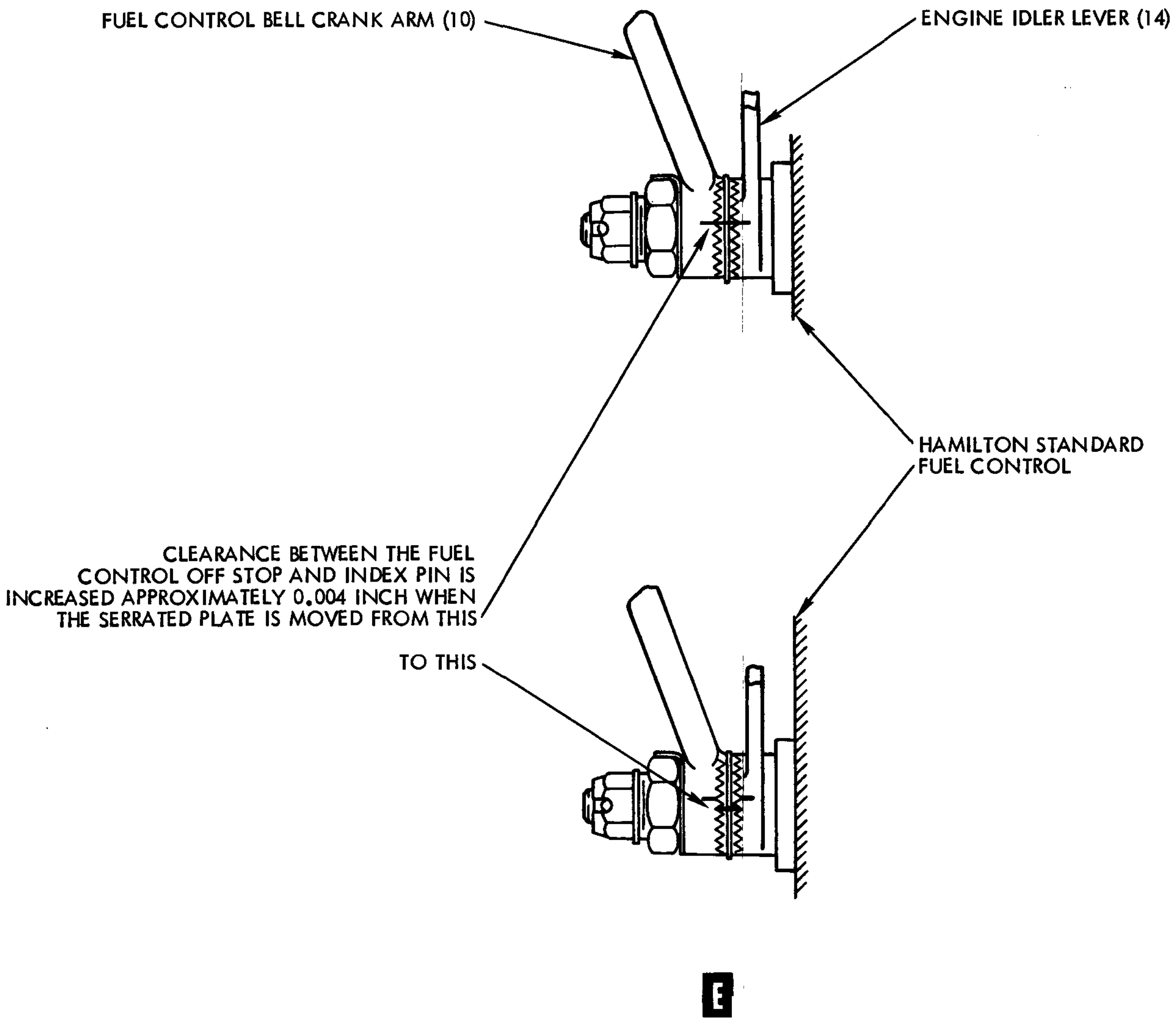
NOTE

3 INSTALL BOLT HEAD AWAY FROM ENGINE

4 IN PART POWER POSITION

5 ITEM 7 SHOULD BE LOCATED ON VERTICAL CENTERLINE OR SLIGHTLY AFT.

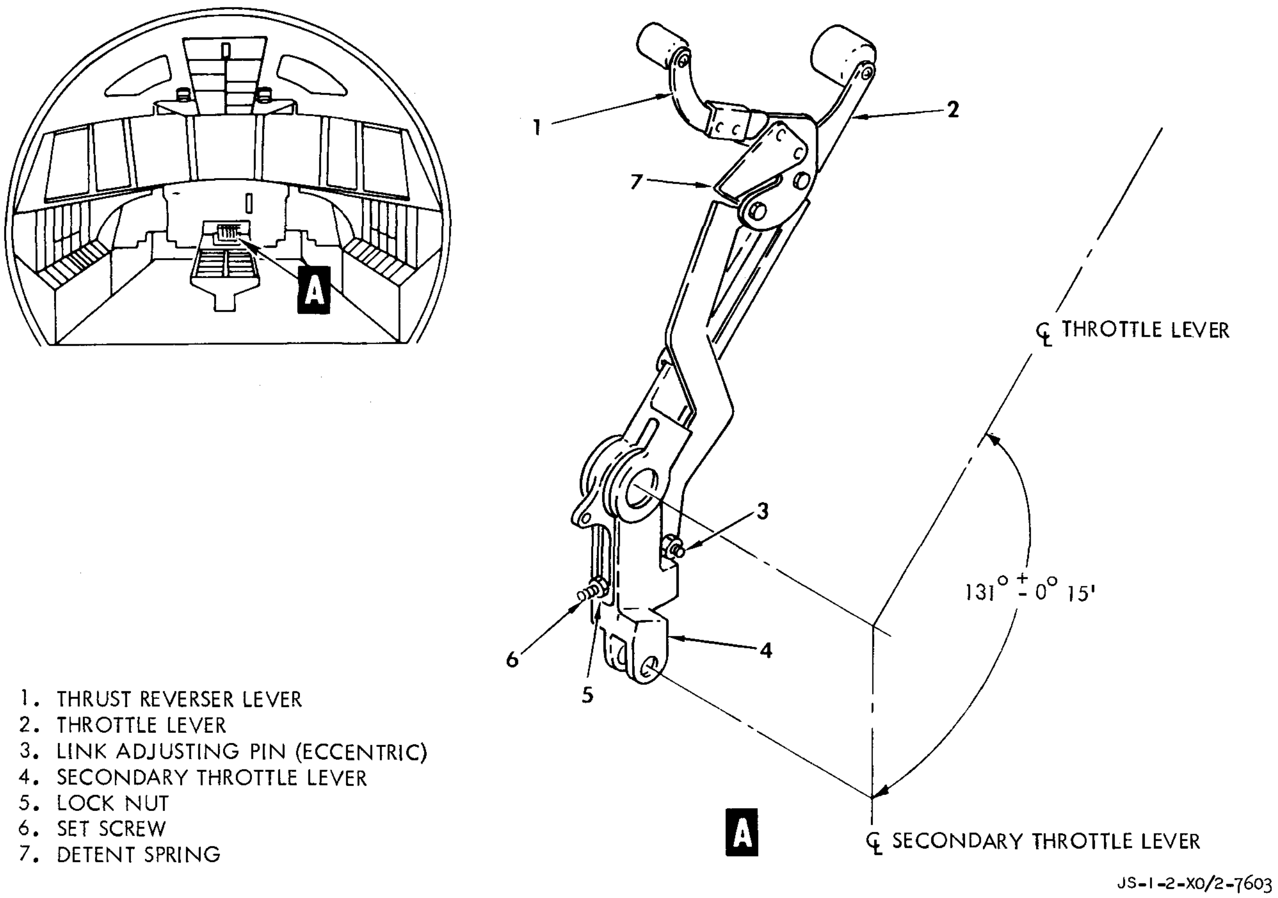




CLEARANCE BETWEEN THE FUEL CONTROL OFF STOP AND INDEX PIN IS INCREASED APPROXIMATELY 0.004 INCH WHEN THE SERRATED PLATE IS MOVED FROM THIS

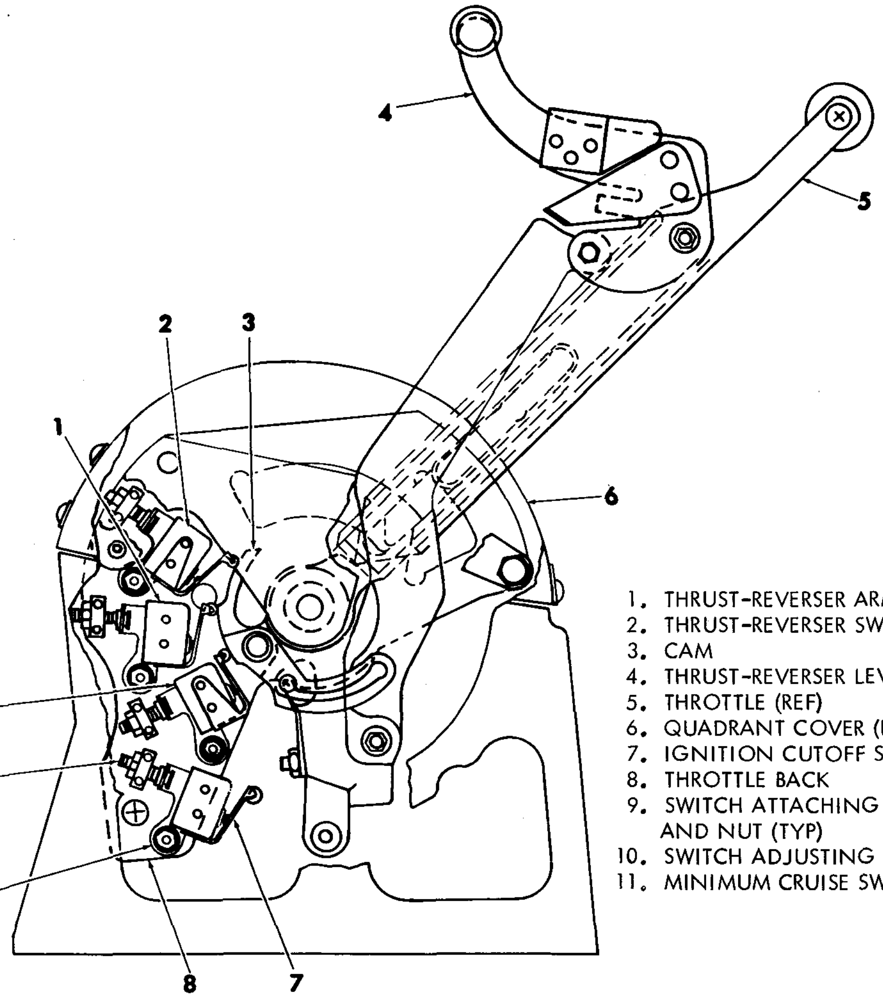
TO THIS

Figure 76-2. Throttle Control Adjustment (Sheet 2 of 2)



1. THRUST REVERSER LEVER
2. THROTTLE LEVER
3. LINK ADJUSTING PIN (ECCENTRIC)
4. SECONDARY THROTTLE LEVER
5. LOCK NUT
6. SET SCREW
7. DETENT SPRING

Figure 76-3. Throttle and Thrust-Reverser Lever Rigging Instructions



1. THRUST-REVERSER ARMING SWITCH
2. THRUST-REVERSER SWITCH
3. CAM
4. THRUST-REVERSER LEVER
5. THROTTLE (REF)
6. QUADRANT COVER (REF)
7. IGNITION CUTOFF SWITCH
8. THROTTLE BACK
9. SWITCH ATTACHING SCREW AND NUT (TYP)
10. SWITCH ADJUSTING SCREW (TYP)
11. MINIMUM CRUISE SWITCH

THROTTLE LEVER (5) SWITCH SETTINGS

IGNITION CUTOUT SWITCH (7)	2.1° MIN. DECREASING 2.8° MAX. INCREASING	ABOVE CUTOFF
MINIMUM CRUISE SWITCH (11)	19.1° MIN. DECREASING 22.5° MAX. INCREASING	ABOVE IDLE

THRUST-REVERSER LEVER (4) SWITCH SETTINGS

THRUST-REVERSER ARMING SWITCH (1)	22.5° MIN. DECREASING 30.5° MAX. INCREASING	ABOVE T/R STOWED POSITION
THRUST-REVERSER SWITCH (2)	30.5° MIN. DECREASING 33.6° MAX INCREASING	

Figure 76-4. Throttle Switch Adjustment

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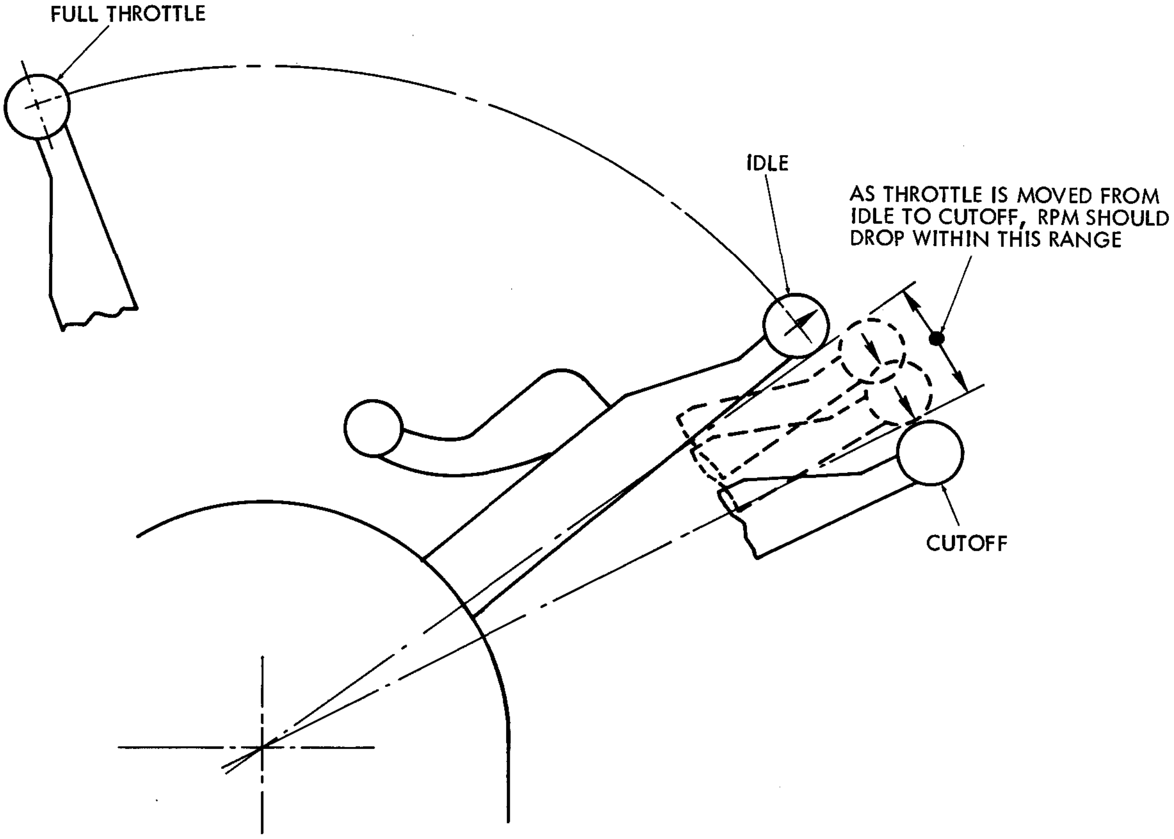

FULL THROTTLE
IDLE
AS THROTTLE IS MOVED FROM IDLE TO CUTOFF, RPM SHOULD DROP WITHIN THIS RANGE
CUTOFF

Figure 76-5. Throttle Knob Check

CHAPTER 77
ENGINE INDICATING

Section Page

77.0 ENGINE PRESSURE RATIO 77-3

77.1 TACHOMETER 77-5

77.2 EXHAUST GAS TEMPERATURE 77-7

77.3 FUEL FLOW 77-9

77.4 OIL PRESSURE 77-11

77.5 OIL TEMPERATURE 77-13

SECTION 77.0
ENGINE PRESSURE RATIO

Contents Page
DESCRIPTION AND OPERATION 77-3
OPERATIONAL CHECKOUT 77-3
TROUBLESHOOTING 77-4
REMOVAL OF PROBE 77-4
REPLACEMENT OF PROBE 77-4

77.0.1 DESCRIPTION AND OPERATION. An engine pressure ratio (EPR) indicating system provides a measure of engine thrust. The Pt_0 signal comes from a probe between the engine inlets on each nacelle. The Pt_5 signal is provided from a connection on each engine. Two pressure ratio transducers are provided in the aft fuselage compartment. One of these receives the Pt_5 signal from engine No. 1, the Pt_5 signal from engine No. 2, and the Pt_0 signal from the left nacelle. The other transducer receives signals from the right engines and nacelle. Each transducer contains two pressure ratio transmitters (one for each engine). Each transmitter is made up of a bellows actuated ratio computer, a cam and gear train, amplifier, two phase motor and velocity generator, and a transmitting synchro. The indicator, mounted in the center section of the main instrument panel, is provided with a synchro repeater which operates the dial pointer to give a visual indication of engine pressure ratio. This system receives power from the aircraft 115 volt A.C. system.

77.0.2 OPERATIONAL CHECKOUT. (Recommended at each engine change or hot section inspection.) If desired, to conserve fuel, operational checkouts of the engine pressure ratio indicating system (this section), tachometer (Section 77.1), and the exhaust gas temperature indicating system (Section 77.2) may be performed concurrently when using the Howell Engine Trim Checker H218-1, or equal.

a. Make sure the pressure source is regulated to zero pressure. Connect the air pressure source to the No. 1 engine Pt_5 system and to the engine trim checker, using pressure hose and a tee, as required. The connections should be made so that pressure can be simultaneously applied to the engine pressure ratio transmitter and to the engine trim checker.

b. Obtain the correct barometric pressure.

c. Close the engine pressure ratio circuit breakers.

d. Increase the pressure with the regulator until the pressure reading on the checker equals 1.5 times the barometric pressure.
RESULT: Aircraft pressure ratio indicator should read 1.50 ±0.09.

e. Increase the pressure with the regulator until pressure reading on the checker equals 2.0 times the barometric pressure.
RESULT: Aircraft pressure ratio indicator should read 2.00 ±0.05.

f. Increase the pressure with the regulator until pressure reading on the checker equals 2.5 times the barometric pressure.
RESULT: Aircraft pressure ratio indicator should read 2.50 ±0.05.

g. Increase the pressure with the regulator until pressure reading on the checker equals 3.0 times the barometric pressure. Hold for 5 minutes.
RESULT: Aircraft pressure ratio indicator should read 3.00 ±0.09, and should remain constant for 5 minutes.

g1. Reduce pressure to zero.

h. In turn, connect pressure hose of checker and pressure source to each Pt_5 exhaust pressure

Test Equipment Required

Nomenclature	Description	Use and Application
Engine Trim Checker	H218-1, Howell Instruments Inc., or equal.	Operational checkouts of the EPR, RPM and EGT.
Air Pressure Supply	0 to 50 PSI with regulator, to supply dry filtered air.	Application of pressure to system.

77.0.2 (Cont).

ratio fitting, and repeat steps d through g for engines No. 2, No. 3, and No. 4.

i. Reduce pressure to zero. Disconnect the checker and the pressure source, and restore systems to normal.

77.0.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR FOR ONE ENGINE FAILS TO PROVIDE AN EPR INDICATION.		
Applicable PRESS RATIO circuit breaker open.	Check applicable circuit breaker.	Close circuit breaker.
Defective indicator.	Substitute an indicator known to be operable.	Replace indicator.
Defective pressure ratio transmitter.	Substitute a transmitter known to be operable.	Replace transmitter.
HIGH EPR READING FOR BOTH ENGINES ON ONE SIDE (in flight).		
Leak in Pt_0 plumbing for affected engines.	Check Pt_0 plumbing and connections for leakage.	Tighten connections or repair as necessary.
LOW EPR READING FOR ONE ENGINE.		
Leak in Pt_5 plumbing for affected engine.	Check Pt_5 plumbing and connections for leakage.	Tighten connections or repair as necessary.

77.0.4 REMOVAL OF PROBE. Remove nacelle inlet lip assembly to obtain access to probe. Uncouple probe.

77.0.4.1 REPLACEMENT OF PROBE. Install probe and perform operational checkout per 30.4.2. Perform necessary leak test on disturbed tubing. Reinstall nacelle inlet lip assembly, and seal around the projecting portion of the probe and hole in assembly with firewall sealing compound (Coast ProSeal and Mfg. Co. PRO-SEAL 700).

SECTION 77.1
TACHOMETER

Contents

DESCRIPTION AND OPERATION 77-5
OPERATIONAL CHECKOUT 77-5
TROUBLESHOOTING 77-5

77.1.1 DESCRIPTION AND OPERATION. The tachometer indicating system provides indications of engine rotor speed (RPM) in percent (100% = 15,909 RPM). Four tachometer indicators, one for each engine, are mounted in the center of the main instrument panel. A tachometer generator, geared to the engine rotor, is mounted on a pad on the engine accessory gear case. When this generator is rotated by the engine, a three phase alternating current is produced with a frequency proportional to engine speed. This current is supplied to a synchronous motor in the tachometer indicator whose speed is directly dependent upon the frequency of the input. Therefore, the generator is locked together electrically with the motor in the indicator and both operate at the same speed. The motor drives a magnetic assembly which actuates the indicator pointer against a calibrated spring to provide a visual indication of engine speed. This system requires no external power and is completely, independent of the aircraft power systems. Colored range markings are applied to the near side face of the tachometer glass. These marks indicate the following: red radial-- 105% of engine r.p.m; green arc covers the 43% to 101.2% range.

77.1.2 OPERATIONAL CHECKOUT. (Recommended at each engine change or hot section inspection.) If desired, to conserve fuel operational checkouts of the engine pressure ratio (Section 77.0), tachometer (this section), and the exhaust gas temperature (Section 77.2) may be performed concurrently when using the Howell Engine Trim Checker H218-1, or equal.

a. Connect RPM check cable to engine trim checker, or equal, and to the aircraft's No. 1 engine tachometer generator through a Y RPM check adapter.
b. Start aircraft engines in accordance with Chapter 72.
c. Advance throttle for No. 1 engine until trim checker RPM indicator reads between 95 and 100%.
RESULT: Aircraft tachometer indicator reading is the same as the trim checker within ±1%.
d. Shut down engine and in turn, connect RPM check cable to the No. 2, No. 3 and No. 4 engines tachometer generators. Repeat step c for each engine.
e. Shut down aircraft engines, disconnect trim checker, and restore systems to normal.

Test Equipment Required

Nomenclature	Description	Use and Application
Engine Trim Checker	H218-1, Howell Instruments Inc., or equal.	Operational checkout of the EPR, RPM, and EGT.

77.1.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO RPM INDICATION.		
Tachometer generator shaft broken.	Substitute a tachometer generator known to be operable.	Replace tachometer generator.
Open phase in indicating circuit. (If one phase opens during flight RPM indication for the affected system will not noticeably decrease. However, upon starting engines after a shutdown RPM indications will not be obtained).	Substitute an indicator and/or generator known to be operable; perform continuity check of interconnecting wiring.	Replace defective unit or repair wiring as necessary.

SECTION 77.2
EXHAUST GAS TEMPERATURE

Contents Page

DESCRIPTION AND OPERATION 77-7

OPERATIONAL CHECKOUT 77-7

CALIBRATION 77-7

TROUBLESHOOTING 77-8

77.2.1 DESCRIPTION AND OPERATION. This system measures the engine exhaust gas temperature (EGT) in the engine tail cone. Three dual element chromel-alumel thermocouple assemblies are installed in each engine. Two connectors on the engine are used to pick up these thermocouples. One connector, which is used only for test work, will provide individual circuits to each of the three radial positions. The other connector, which is used for this indicating system, provides only one circuit which averages the temperature in the three positions. The thermocouples are of the bimetalic type in which a small voltage is generated proportional to the temperature at the junction of the two metals. The indicator, mounted in the center of the main instrument panel, is a milivoltmeter calibrated to read in degrees C. It is designed to operate with a total resistance of the thermocouples and their leads of 22 ohms. To attain this requirement, an adjustable resistor is provided. This system is completely self contained and requires no power from the aircraft power systems. Colored range markings are applied to the near side face of the indicator glass. For the JT12A-6 and 6A, these marks indicate the following: red radial 677°C: yellow arc: 577°C to 677°C; green arc: 400°C to 577°C. For the JT12A-8 engine they are: red radial 718°C: yellow arc 655°C to 718°C: green arc 400°C to 655°C.

77.2.2 OPERATIONAL CHECKOUT. (Recommended at each engine change, hot section inspection or whenever an EGT indicator is changed.) If desired, to conserve fuel operational checkouts of the engine pressure ratio (Section 77.0), tachometer (Section 77.1), and exhaust gas temperature (this section) may be performed concurrently when using the Howell Engine Trim Checker H218-1, or equal.

a. Connect engine trim checker, or equal, to the aircraft engine EGT function connector.

Note

It may be necessary to use the trim checker's cable adapter, or equal, to match the engine EGT connector.

b. Start aircraft engine in accordance with Chapter 72.

c. Monitor aircraft and trim checker EGT readings for each engine. If there is a disagreement of ±10°C at any exhaust gas temperature between 600° and 700°C the EGT thermocouple resistor for the applicable engine should be calibrated in accordance with paragraph 77.2.3.

d. Shut down aircraft engines, disconnect trim checker, and restore systems to normal.

Note

The above checkout limits check the aircraft indicator and wiring only. A check of the engine thermocouple may be accomplished by the use of a thermocouple heater in conjunction with a calibrated thermocouple, as indicated in Howell Instrument Jetcal Analyzer, or equal.

77.2.3 CALIBRATION. If needed, calibrate the EGT thermocouple resistor in accordance with the following steps:

a. *Disconnect leads from the applicable engine thermocouples and determine that their resistance is between 0.6 and 1.0 ohm. Reconnect leads.*

b. *Determine that the engine EGT system is correctly wired.*

c. Disconnect the indicator leads and connect a wheatstone bridge, or equal, across the applicable engine's EGT system leads.

Test Equipment Required

Nomenclature	Description	Use and Application
Engine Trim Checker	H218-1, Howell Instrument Inc., or equal.	Operational checkout of the EPR, RPM, and EGT.
Wheatstone Bridge or equal	Standard.	Calibration of EGT thermocouple resistor.
Thermocouple Tester	Jetcal Analyzer, Howell Instrument Inc.	Check of engine thermocouples.

77.2.3 (Cont).

RESULT: The indicated resistance should be 22.00 ±0.05 ohms.

d. If necessary adjust the thermocouple resistor to obtain 22.00 ±0.05 ohms. The resistor is located at F.S. 589.5 near the centerline of the aircraft. Cut off and remove necessary turns of wire on the spools to reduce the resistance. The resistance is increased by replacing the spools.

e. Restore the applicable EGT system to normal and decal the resistor spool cover.

77.2.4 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
EGT INDICATOR READING TOO HIGH		
Resistance of thermocouple system less than 22.00 ±0.05 ohms.	Check resistance, refer to paragraph 77.2.3.	Calibrate resistor, refer to paragraph 77.2.3.
Resistance of engine harness too low.	Check resistance, refer to paragraph 77.2.3 (step a).	Replace thermocouple if found defective.
EGT INDICATOR READING TOO LOW		
Bad electrical connections	Check connections at indicator, at pressure bulkhead (F.S. 570) and at the engine disconnect.	Tighten, clean, and/or repair connections as necessary.
Resistance of thermocouple system more than 22.00 ±0.05 ohms.	Check resistance, refer to paragraph 77.2.3.	Calibrate resistor, refer to paragraph 77.2.3.
Resistance of engine harness too high.	Check resistance, refer to paragraph 77.2.3 (step a).	Replace thermocouple if found defective.
NO EGT INDICATION		
Defective indicator	Substitute an indicator know to be operable.	Replace indicator.
Open circuit in aircraft or engine harness	Check resistance; refer to paragraph 77.2.3.	Repair or replace wiring

SECTION 77.3
FUEL FLOW

Contents Page

DESCRIPTION AND OPERATION 77-9

TROUBLESHOOTING 77-9

77.3.1 DESCRIPTION AND OPERATION. The fuel flow system provides an indication of the mass fuel flow to each engine measured in pounds per hour. A fuel flow transmitter is mounted in the fuel line leaving the engine fuel control on each engine. This transmitter is made up of a motor driven impeller and a restrained turbine. As the fuel enters the transmitter, it is introduced into the impeller, which is being turned at a constant speed by a synchronous motor. This imparts an angular momentum to the fuel, which is then introduced into the turbine, which is restrained by a calibrated spring. The turbine turns against the spring by an amount directly proportional to the mass of the fuel passing through it. The position of the turbine is converted into an electric signal which is sent to the fuel flow indicators, one for each engine. The indicators are mounted in the center of the main instrument panel. On aircraft 5001 through 5045, this electrical signal is produced by a three wire position synchro. The indicator is provided with a synchro repeater and the system operates directly from the aircraft 115-volt, A.C. power system. On aircraft 5046 and up, a different system is provided. This system operates in a similar manner except that a second harmonic selsyn telemetering system is used for the signal between the transmitter and the indicator. This system operates from the aircraft 115-volt, A.C. power supply through two power supply units. One power supply unit operates the fuel flow system for engines No. 1 and No. 4, while the other operates the system for engines No. 2 and No. 3.

77.3.2 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR FOR ONE ENGINE FAILS TO PROVIDE A FUEL FLOW INDICATION.		
Applicable fuel flow indicator circuit breaker open.	Check circuit breaker.	Close circuit breaker.
Defective indicator.	Substitute an indicator known to be operable.	Replace indicator.
Defective fuel flow transmitter.	Substitute a transmitter known to be operable.	Replace transmitter.
INDICATORS FOR ENGINES 1 AND 4 OR 2 AND 3 FAIL TO PROVIDE A FUEL FLOW INDICATION (AIRCRAFT 5046 AND UP).		
For indicators 1 and 4 - Fuel flow power supply No. 1 (Airplanes 5046 and up).	Substitute a power supply known to be operable.	Replace power supply.
For indicators 2 and 3- Fuel flow power supply No. 2 (Airplanes 5046 and up).	Substitute a power supply known to be operable.	Replace power supply.
FUEL FLOW FLUCTUATION IN EXCESS OF ±25 PPH.		
Defective transmitter.	Substitute a transmitter known to be operable.	Replace transmitter.

SECTION 77.4
OIL PRESSURE

Contents

Contents	Page
DESCRIPTION AND OPERATION	77-11
OPERATIONAL CHECKOUT	77-11
TROUBLESHOOTING	77-12

77.4.1 DESCRIPTION AND OPERATION. The oil pressure indicating system provides an indication of oil pressure for each engine measured in pounds per square inch (PSI). The system is provided with a transmitter on each engine, an OIL PRESS LOW warning light, and two indicators on the center of the main instrument panel. Each of these indicators is a dual indicator. The left indicator shows the oil pressure for engine No. 1 and No. 2, while the right one shows it for engines No. 3 and No. 4. The transmitter is mounted directly on the engine on a port on the engine accessory gear case. This port is connected to the engine oil pump outlet through a restrictor mounted in the pressure port of the transmitter. The transmitter is vented to the engine gear case. Pressure operating on a bellows for both the pressure and vent ports produces movement in a linkage within the transmitter proportional to the differential pressure involved. This movement operates a synchro which produces a signal for the synchro repeater in the indicator. This system requires 26 volts, AC from the aircraft power supply. Each indicator is provided with two switches. one for each engine, which are closed to operate the OIL PRESS LOW annunciator warning light at pressures below 33 ±1 PSI. Colored range markings are applied to the near side face of the indicator glass. These marks indicate the following: red radial: 50 p.s.i. and 35 p.s.i.; yellow arc: 35 p.s.i. to 40 p.s.i.; green arc: 40 p.s.i. to 50 p.s.i.

77.4.2 OPERATIONAL CHECKOUT.

a. Place all ENG OIL PRESS LOW ANNUNCIATOR LT CUT OFF switches to the 'OFF' position.

b. Energize the essential and main 26 volt AC, 400 cycle buses.

c. Close the warning lights circuit breaker on the pilot's side circuit breaker panel.
RESULT: The master caution lights should go on.

d. Reset the master caution lights by depressing face of lights.
RESULT: The lights should go off.

e. Disconnect the pressure port connector on the No. 1 engine oil pressure transmitter.

f. Connect a pneumatic pressure source to the pressure port of the oil pressure transmitter.

g. Close the No. 1, No. 2, No. 3, and No. 4 OIL PRESS IND circuit breakers on the copilot's side circuit breaker panel.

h. Place the No. 1 ENG OIL PRESS LOW ANNUNCIATOR LT CUT OFF switch in the "ON" position.
RESULT: The OIL PRESS LOW warning and master caution lights should go on and the No. 1 engine oil pressure indicator should indicate 0 ±5 PSI.

i. Place the No. 1 ENG OIL PRESS LOW ANNUNCIATOR LT CUTOFF switch in the "OFF" position, and then in the "ON" position.
RESULT: The OIL PRESS LOW warning and master caution lights should first go off and then go on as the switch is returned to the "ON" position.

j. Slowly increase the pressure until the OIL PRESS LOW warning and master caution lights go off. (This should occur at an applied pressure of 33 ±3 PSI.)

Test Equipment Required

Nomenclature	Description	Use and Application
Pneumatic Pressure Source (with a Master Gage)	Capable of supplying controlled pressures from 0 to 50 PSI.	Operational checkout of the engine oil pressure indicators.

77.4.2 (Cont).

RESULT: The No. 1 engine oil pressure indicator should read 33 ±1 PSI at the moment the lights go off.

k. Slowly increase the pressure to 50 PSI.
RESULT: The No. 1 engine oil pressure indicator should indicate 50 ±3 PSI.

l. Slowly decrease the pressure to zero.
RESULT: The OIL PRESS LOW warning and master caution lights should go on when the applied pressure reaches 33 ±3 PSI, and the No. 1 engine oil pressure indicator should read 33 ±1 PSI at the moment the lights go on.

m. Reset the master caution lights by depressing the face of the lights.
RESULT: The master caution lights should go off.

n. Place the No. 1 ENG OIL PRESS LOW ANNUNCIATOR LT CUT OFF switch in the "OFF" position.
RESULT: The OIL PRESS LOW warning light should go off.

o. Repeat steps e, f, and h through n for engines No. 2, No. 3, and No. 4 except use the corresponding engine oil pressure transmitter, indicator, and switch.

p. Disconnect the pressure source, reconnect the transmitter pressure port, de-energize the electrical buses, and restore the systems to normal.

77.4.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
OIL PRESS LOW WARNING AND MASTER CAUTION LIGHTS DO NOT GO OFF OR ON AT CORRECT PSI READING.		
Defective switch in the applicable indicator.	Substitute an indicator known to be operable.	Replace indicator.
INDICATOR FOR ONE ENGINE FAILS TO PROVIDE AN OIL PRESSURE INDICATION.		
Applicable OIL PRESS IND circuit breaker open.	Check circuit breaker.	Close circuit breaker
Defective indicator.	Substitute an indicator known to be operable.	Replace indicator.
Defective transmitter.	Substitute a transmitter known to be operable.	Replace transmitter.
OIL PRESS LOW WARNING AND MASTER CAUTION LIGHTS FAIL TO GO ON (INDICATOR READS CORRECT PSI).		
Warning lights circuit breaker open.	Check circuit breaker.	Close circuit breaker.
Defective switch in the applicable indicator.	Substitute an indicator known to be operable.	Replace indicator.
OIL PRESSURE FLUCTUATION.		
Defective engine oil system.	Refer to Chapter 72.	Refer to Chapter 72.
ONE POINTER ON INDICATOR IS 180 DEGREES OUT OF PHASE WITH OTHER POINTER WITH BOTH ENGINES AT APPROX. SAME SETTING.		
One phase of the circuit open.	Substitute an indicator and/or transmitter known to be operable; perform continuity check of interconnecting wiring	Replace defective unit or repair wiring as necessary.

77.4.3 (Cont).

OIL PRESSURE INDICATION DECREASES AS ALTITUDE INCREASES		
Defective oil pressure transmitter vent line.	Check for restriction in transmitter vent line.	Replace rubber vent line with one of all metal construction. Clean out all metal vent lines.

SECTION 77.5
OIL TEMPERATURE

Contents Page
DESCRIPTION AND OPERATION 77-13
OPERATIONAL CHECKOUT 77-13
TROUBLESHOOTING 77-13

77.5.1 DESCRIPTION AND OPERATION. The engine oil temperature indicating system consists of an oil temperature indicator and a temperature sensitive pickup (bulb) for each engine. The bulb is mounted in a port on the engine adjacent to the oil pressure transmitter. The four indicators are mounted on the right, center main instrument panel above the oil pressure indicators and provide temperature indications in degrees C. The system is electrically operated, and employs the wheatstone bridge principle of measuring a resistance which is varied by temperature changes. The system is powered by 28-volt DC from the aircraft power supply. Colored range markings are applied to the near side face of the indicator glass. The marks indicate the following: red radial: 121°C; and green arc: 40°C to 121°C.

77.5.2 OPERATIONAL CHECKOUT.

a. Energize the aircraft main DC bus.

b. Remove the electrical plug from the No. 1 engine oil temperature bulb.

c. Close engine No. 1, No. 2, No. 3, and No. 4 OIL TEMP IND circuit breakers on the copilot's side circuit breaker panel.

d. Using precision resistors, connect 104.6 (±0.4) ohms across pins A and B of the electrical plug disconnected in step b above.
RESULT: The No. 1 engine oil temperature indicator should read 40 (±5)°C.

e. Increase the resistance across the pins to 120.36 (±0.5) ohms.
RESULT: The No. 1 engine oil temperature indicator should read 80 (±5)°C.

f. Increase the resistance across the pins to 137.78 (±0.6) ohms.
RESULT: The No. 1 engine oil temperature indicator should read 120 (±5)°C.

g. Repeat steps b, and d through f for engines No. 2, No. 3, and No. 4.

h. Disconnect resistors, reconnect all electrical plugs, de-energize the main DC bus and return all system to normal.

Test Equipment Required

Nomenclature	Description	Use and Application
Precision Resistors	Capable of providing resistances of 104.6 ±0.4, 120.36 ±0.5, and 137.78 ±0.6 ohms respectively.	Operational checkout of the engine oil temperature indicators.

77.5.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR FOR ONE ENGINE FAILS TO PROVIDE AN OIL TEMPERATURE INDICATION.		
Applicable OIL TEMP IND circuit breaker open.	Check circuit breaker.	Close circuit breaker.
Defective indicator.	Substitute an indicator known to be operable.	Replace indicator.

77.5.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
INDICATOR FOR ONE ENGINE FAILS TO PROVIDE AN OIL TEMPERATURE INDICATION. (Cont).		
Defective temperature bulb.	Substitute a temperature bulb known to be operable.	Replace bulb.
TEMPERATURE READING TOO HIGH ON ONE OR MORE INDICATORS.		
Loose, dirty or defective electrical connector at the pressure bulkhead (F.S. 570) causing circuit resistance to increase.	Check connector.	Tighten, clean, or repair as necessary.

All data on pages 77-15 through 77-18, including figures 77-1 and 77-2 deleted.

CHAPTER 78
EXHAUST

Contents Page

DESCRIPTION AND OPERATION 78-2
General 78-2
Actuators 78-2
Thrust-Reverser Selector Valves 78-2
Accumulator, Air Filler Valve, and Pressure Gage 78-2
Low Hydraulic Pressure Warning Switch 78-2
Full Reverse Thrust Indicating System 78-2
Full Forward Thrust Indicating System 78-2
Operation 78-2
Restrictors 78-3
Check Valves 78-3

OPERATIONAL CHECKOUT 78-3
Thrust-Reverser System 78-3
Arming Switches 78-4
Actuator Position Switch 78-4

TROUBLESHOOTING 78-5

REMOVAL AND REPLACEMENT 78-7
Removal 78-7
Replacement 78-9

RIGGING AND ADJUSTMENT 78-11

THRUST REVERSER MINOR REPAIR 78-12

THRUST REVERSER MINOR REPAIR 78-12
Door Inner Skin Blind Rivet Repair 78-12
Door End Plate Crack Repair 78-12
Door Aft "Z" Angle Wear Repair 78-13
Door Outer Skin Crack Repair 78-13
Repair of Dry Film Lubricated Parts 78-13

PERIODIC LUBRICATION 78-14

INSPECTION AND OVERHAUL REQUIREMENTS 78-14
600-Hour Inspection 78-14
Cleaning 78-14
Inspection 78-14
Reassembly 78-14

OVERHAUL 78-14
Part Replacement Criteria 78-14

FIGURES
78-1. Thrust-Reverser Door Positions 78-15
78-2. Thrust-Reverser Schematic Diagram 78-16
78-3. Actuator Position Switches 78-17
78-4. Unlocking Doors 78-18
78-5. Doors and Supporting Links 78-19
78-6. Overcenter Links and Sliding Fittings 78-20
78-7. Actuator 78-21
78-8. Longeron Assembly Removal 78-22
78-9. Driver Link and Driven Link Adjustment 78-22
78-10. Overcenter Position Rigging 78-23
78-11. Door Preload Adjustment 78-24
78-12. Guide Spring Adjustment 78-25

Contents Page

78-13. Door Inner Skin Blind Rivet Repair 78-26
78-14. Door Aft "Z" Angle Repair 78-27
78-15. Door End Plate Crack Repair 78-28
78-16. Overcenter Position Rigging - Optional Tool 78-30
78-17. Door Outer Skin Crack Repair 78-31
78-18. Thrust-Reverser Wear Limits 78-32
78-19. Thrust-Reverser Track Wear Inspection 78-33

78.1 DESCRIPTION AND OPERATION.

78.1.1 GENERAL. (See figures 78-1 through 78-3.) Each engine is provided with a nacelle mounted, hydraulically operated, electrically controlled, two position target type thrust reverser for ground operation only. When reverse-thrust levers on the throttles are moved to the reverse-thrust position, pressure from the normal hydraulic system actuates the clamshell doors into the path of the engine exhaust gases. This causes a general forward deflection of the gases which results in reverse thrust on the airplane. No emergency operating system is provided. It should be noted, however, that the normal hydraulic system may be pressurized with the auxiliary hydraulic pump in case of normal pump failure. Airplanes 5001 through 5035 have partial provisions for a nitrogen-actuated emergency thrust-reverser system. These provisions consist of a circuit breaker, control switch, wiring, control valve, and plumbing, all of which are inoperative.

CAUTION

Operating of the thrust reverser system at airplane speeds less than those specified in the Airplane Flight Manual may contribute to damage of the empennage pivot fitting.

78.1.2 ACTUATORS. There is one double-acting actuator for each thrust-reverser. Extending the actuator cylinder retracts the thrust-reverser doors to the forward thrust position. Retracting the actuator cylinder extends the thrust-reverser doors to the reverse-thrust position. Position switches on the actuator are used in the thrust-reverser control and indicating circuits. Ram air cooling for the actuator is provided on airplanes 5046 and up.

78.1.3 THRUST-REVERSER SELECTOR VALVES. Four hydraulic selector valves, one for each thrust-reverser, control pressure to the actuating cylinders in accordance with the position of the thrust-reverser levers. Each valve is a four-way, three-position, double solenoid valve. The selector valves for engines No. 1 and 2 are attached to the left side of the aft compartment. The selector valves for engines No. 3 and 4 are attached to the right side.

Note

The thrust reverser selector valves have a relatively high internal leakage rate between cylinder 1 and return and between cylinder 2 and return. When normal (No. 1) system pressure is reduced, the THRUST REV PRESS LOW warning module may illuminate within three minutes.

78.1.4 ACCUMULATOR, AIR FILLER VALVE, AND PRESSURE GAGE. The thrust-reverser accumulator has a capacity of 150 cubic inches; it is a piston type accumulator attached to the airplane structure forward of the speed brake opening inside the aft compartment. An air filler valve and pressure gage aid accumulator servicing.

Note

A check valve in the thrust-reverser system prevents hydraulic pressure from bleeding off after normal hydraulic system pressure is reduced. To relieve hydraulic pressure in the thrust reverser system accumulators it is necessary to operate the thrust-reversers.

78.1.5 LOW HYDRAULIC PRESSURE WARNING SWITCH. As hydraulic pressure in the thrust-reverser system decreases, the air pressure in the accumulator decreases until the precharge value is reached. A pressure sensing switch tapped into the air side of the accumulator closes at 2,500 (±50) PSI to complete a circuit to the THRUST REV PRESS LOW warning module in the annunciator panel. Electrical power to this module is cut off when reverse-thrust is selected on any one thrust-reverser.

78.1.6 FULL REVERSE-THRUST INDICATING SYSTEM. Normal actuation of the thrust-reverser system is indicated by a blue light for each engine. The lights are labeled THRUST REV EXTENDED and are on the instrument panel glare shield. Each light is turned on when the corresponding position-indicating switch closes as the thrust-reverser actuator moves to the full reverse-thrust position. Failure of any one thrust-reverser to move to the full reverse-thrust position results in the indicator light remaining out after reverse thrust has been selected. This warns the pilot of an asymmetrical thrust condition.

78.1.7 FULL FORWARD-THRUST INDICATING SYSTEM. When a reverser is not in the full forward-thrust position, an amber THRUST-REVERSER UNLOCKED light on the center console comes on. One light is provided for each of the four reversers and is controlled by the position switch on the reverser actuator.

78.1.8 OPERATION.
(See figure 78-2.)

a. To actuate any thrust-reverser to the reverse-thrust position. the throttle for that engine must be in "IDLE". Construction of the throttle prevents the reverser lever from being moved into reverse unless the throttle is at "IDLE."

b. Pulling the reverse thrust lever up and back toward the throttle knob first closes the reverse-thrust arming switch, and then closes the reverse-thrust switch. This switch completes an electrical circuit from the 28-volt, DC main bus to the reverser selector valve. The energized selector valve moves to a position to route hydraulic pressure to retract the reverser actuator. The reverser actuator positions the reverser doors in the exhaust stream. and reverse thrust is obtained. Additional movement of the reverse-thrust lever increases engine power.

c. Each actuator is provided with switches that are actuated at each end of the piston stroke. When the actuator piston reaches the forward-thrust locked position, two switches are actuated. One switch cuts off thrust-reverser hydraulic pressure, and the other switch breaks the circuit to the thrust-reverser unlocked light, thus indicating that the thrust-reverser doors are retracted and locked. When the actuator piston reaches the reverse position, a switch is actuated and provides electrical power to a thrust-reverser extended light, indicating that the thrust-reverser doors are fully extended to the reverse-thrust position. A circuit to an extend-sensing relay is also completed when the actuator moves to the reverse-thrust position which disables the REV PRESS LOW light on the annunciator panel as long as any reverser is in the reverse-thrust position.

d. To retract the reversers, the thrust-reverse lever is moved down to the forward-thrust position. This closes the reverse-thrust switch and routes power from the 28-volt DC main bus, through one of the actuator position

78.1.8 (Cont).

switches, to the retract solenoid of the reverser selector valve. This directs hydraulic pressure to the reverser actuator to extend the actuator and move the doors to the forward-thrust position. As the doors reach the full forward-thrust position, position switches on the actuator are operated to energize indication circuits. These switches also break the circuit to the selector valve and the valve returns to the neutral position, which connects both the retraction and the extension ports to the normal hydraulic system return.

78.1.9 RESTRICTORS. Several restrictors are installed in the thrust reverser door operating systems to control or meter hydraulic flow.

a. A restrictor is installed in the pressure supply line for the thrust reverse system. Action of this restrictor permits continued operation of the normal (No. 1) hydraulic system, following thrust reverse operation, by controlling the rate at which the thrust reverse accumulator is replenished.

b. A two-way restrictor is installed in each thrust reverse door hydraulic actuating circuit to control the operating speed of the doors.

78.1.10 CHECK VALVES. Check valve assemblies are installed in the thrust reverse hydraulic plumbing, positioned to prevent undesirable fluid flow.

a. A check valve is installed in the pressure supply line for the thrust reverse system. This check valve prevents immediate loss of thrust reverse accumulator pressure in the event that the normal (No. 1) hydraulic system is depressurized.

b. Airplanes 5046 and up, and airplanes having Service Bulletin 329-214 incorporated, have a check valve installed in the return line downstream of each thrust reverse selector valve. The primary purpose of these check valves is to prevent unintentional unlocking of the thrust reverser assemblies due to transient pressure surges in the normal (No. 1) hydraulic system return line.

78.2 OPERATIONAL CHECKOUT.

Test Equipment Required

Nomenclature	Description	Use and Application
Ohmmeter or suitable continuity checker		To check continuity through actuator switches.
Hydraulic Handpump with Gage.	0- to 500-PSI capacity	To pressurize the thrust-reverser actuator.
Door Unlock Tool	Lockheed Part No. 11098-MEU-1	To unlock the thrust-reverser doors during checkout.

78.2.1 OPERATIONAL CHECKOUT OF THRUST-REVERSER SYSTEM.

WARNING

Prevent personnel from entering the immediate area of the thrust-reversers. The thrust-reversers operate with a rapid action and with 3,000-PSI hydraulic pressure.

a. Advance all four throttles until they are above the "IDLE" position.

b. Energize the main and essential DC buses.

c. Close the No. 1, No. 2 and No. 3, No. 4 engines thrust-reverser circuit breakers on the pilot's wainscot circuit breaker panel.

d. Close the warning lights circuit breaker on the pilot's side circuit breaker panel.

78.2.1 (Cont).

RESULT: The THRUST REV LOW PRESS warning light on the annunciator panel may go on. Several other warning lights may go on when the warning lights circuit breaker is closed.

e. Charge the thrust-reverser accumulator and main hydraulic system accumulator in accordance with instructions in Chapter 12.

f. Connect an external source of 3,000-PSI hydraulic pressure capable of delivering 3 gallons per minute to the airplane hydraulic system in accordance with the instructions in Chapter 12.

g. Apply hydraulic pressure to the airplane hydraulic system.
RESULT: The THRUST REV LOW PRESS warning light should go out when the pressure in the airplane hydraulic system increases to 2,800 to 3,000 PSI.

78.2.1 (Cont).

h. With the No. 1 engine throttle above the "IDLE" position, attempt to actuate the No. 1 engine thrust-reverser lever.
RESULT: The thrust reverser lever should not move.

i. Place the No. 1 engine throttle in the "IDLE" position and pull the No. 1 engine thrust-reverser lever up.
RESULT: (1) The No. 1 engine thrust-reverser should extend within 2 seconds.
(2) The amber No. 1 THRUST-REVERSER UNLOCKED light should come on.
(3) The blue No. 1 THRUST-REVERSER EXTENDED light should come on.

j. With the No. 1 engine thrust-reverser extended, attempt to move the No. 1 engine throttle above the "IDLE" position.
RESULT: The throttle should not move.

k. Return the No. 1 engine thrust-reverser lever to its normal position.
RESULT: (1) The No. 1 engine thrust-reverser should retract within 2 seconds.
(2) The blue indicator light should go out.
(3) The amber indicator light should go out.

l. Slowly move the No. 1 engine thrust-reverser lever to the reverse-thrust position, and note the position of the lever when the thrust-reverser actuates.
RESULT: The movement of the lever should not not have exceeded 2.7 inches as measured at the knob.

m. Return the No. 1 engine thrust-reverser lever to its normal position.

n. Repeat steps (h) through (m) for the other three engines.

o. Disconnect all test equipment, and restore the airplane to normal.

78.2.2 THRUST-REVERSER ARMING SWITCHES CHECKOUT. The following procedure should be used after engine change, or at any time the throttle control system has been adjusted, or after any work has been performed on the airplane which has involved disturbing the thrust-reverser electrical wiring between the center console and the aft pressure bulkhead.

a. With no external electrical power connected to the airplane, and with all four throttles in the "IDLE" position, check that each thrust-reverser lever is in forward-thrust position, and that all REVERSE THRUST circuit breakers are disengaged.

b. Obtain access to the fuselage aft compartment, and disconnect electrical cable connectors J-1 and J-14 at the aft pressure bulkhead.

c. Connect the electrical continuity checking equipment between pin "h" in the bulkhead connector J-1 and a suitable ground point, and check for electrical continuity.
RESULT: Continuity should be indicated.

d. Place No. 4 thrust-reverser lever in the reverse-thrust position.
RESULT: An open circuit should be indicated.

e. Depress the No. 1 THRUST-REVERSER UNLOCKED light.
RESULT: Continuity should be indicated.

f. Return No. 4 thrust-reverser lever to the forward thrust position, disconnect the test lead from pin "h", and connect it to pin "k."
RESULT: Continuity should again be indicated.

g. Place No. 3 thrust-reverser lever in the reverse thrust position.
RESULT: An open circuit should be indicated.

h. Depress the No. 3 THRUST-REVERSER UNLOCKED LIGHT.
RESULT: Continuity should be indicated.

i. Return No. 3 thrust-reverser lever to the forward-thrust position.
RESULT: An open circuit should be indicated.

j. Repeat steps (c) through (i), using pins "n" and "p" in bulkhead connector J-14 and thrust-reverser levers No. 1 and No. 2 respectively, for the same results.

k. Disconnect test equipment and restore aircraft to normal.

78.2.3 THRUST-REVERSER ACTUATOR POSITION SWITCH CHECKOUT.
(See figure 78-3.)

a. Check that all four throttles are in the "OFF" position with the thrust reverser levers in the forward thrust position. Disengage the REVERSE THRUST circuit breakers.

b. Obtain access to the desired thrust-reverser actuator by removing the engine tail pipe, ejector, tunnel drain, and tunnel covers (if installed). Disconnect the actuator electrical connector at the nacelle firewall.

c. Disconnect both hydraulic lines from the actuator ports. Provide a suitable container for spilled fluid and position it as necessary. Move the thrust-reverser doors to the forward-thrust position.

d. Connect a hydraulic handpump to the extension port of the actuator, and check for continuity between pins "A" and "B" of the electrical connector while applying 150 PSI to fully extend the actuator.
RESULT: There should be no continuity between pins "A" and "B."

78.2.3 (Cont).

e. Release hydraulic pressure and again check the continuity between pins "A" and "B."
RESULT: There should be no continuity between pins "A" and "B."

f. Using door unlock tool, unlock doors and check continuity between pins "A" and "B."
RESULT: There should be continuity between pins "A" and "B."

g. Repeat steps d and e checking continuity between pins "D" and "E."
RESULT: There should be no continuity between pins "D" and "E."

g1. Repeat step f checking continuity between pins "D" and "E."
RESULT: There should be continuity between pins "D" and "E."

h. Disconnect the pump from the extension port of the actuator and connect it to the retraction port of the actuator.

i. Apply 150 PSI hydraulic pressure to the retraction port of the actuator, and check for continuity between pins "C" and "D."
RESULT: There should be continuity between pins "C" and "D."

j. Release hydraulic pressure and check for continuity between pins "C" and "D."
RESULT: There should be no continuity between pins "C" and "D."

k. Disconnect the hydraulic handpump and connect the hydraulic hose from the airplane hydraulic system.

l. Install the upper tunnel cover. Seal with firewall sealant (see table of Consumable Materials, preceding paragraph 78.4.2).

m. Pass the electrical lead through the hole in tne upper tunnel cover and install it in the receptacle in the frame adjacent to the tunnel.

n. Install the electrical fairleads around the electrical lead where it passes through the upper tunnel cover. Seal with firewall sealant.

WARNING

Before operating the thrust-reversers, make sure all personnel are clear of the doors. Install the ground lock as soon as possible after completing the check.

o. Apply hydraulic system pressure, and engage the applicable thrust-reverser system circuit breaker. Move the throttle to the "IDLE" position, and cycle the thrust-reverser several times to bleed the hydraulic system, finishing in the forward thrust position. Check for hydraulic leaks during cycling. Return the throttle to the "OFF" position, and disengage the thrust-reverser circuit breakers.

p. Install the side tunnel cover and the lower cover. Seal with firewall sealant.

q. Install the tunnel drain.

r. Install the ejector assembly.

s. Install engine tailpipe in accordance with instructions in Chapter 71.

78.3 TROUBLESHOOTING.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
THRUST-REVERSER WILL NOT MOVE TO POSITION FOR REVERSE THRUST.		
No hydraulic power.	Check hydraulic pressure gage.	If pressure is low or no pressure is indicated, troubleshoot the hydraulic system.
Thrust-reverser arming switch or thrust-reverser switch in throttle quadrant not actuating properly.	Check throttle switches for continuity.	Replace or adjust as necessary.
Thrust-reverser selector valve defective.	Check for 28-volts, DC between pins A and C of selector valve solenoid.	If correct voltage is present, replace valve.

78.3 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
THRUST-REVERSER WILL NOT MOVE TO POSITION FOR FORWARD THRUST.		
No hydraulic power.	Check hydraulic pressure gage.	If pressure is low or no pressure is indicated troubleshoot the hydraulic system.
Thrust-reverser switch in throttle quadrant not actuating properly.	Check switch for continuity.	Replace or adjust as necessary.
Forward-thrust position switch on reverser actuator defective.	Check for continuity through the actuator switches.	Replace actuator.
Forward-thrust position switch plunger or plunger spring defective.	Check for plunger movement as actuator is cycled.	Replace actuator.
THRUST-REVERSER POSITION LIGHT DOES NOT GO ON WHEN DOORS ARE IN REVERSE-THRUST POSITION.		
Position switch on reverser actuator defective.	Check for continuity through the actuator switches.	Replace actuator.
Defective wiring between position switch and indicator light.	Perform continuity check of indicator light circuit.	Repair as necessary.
THRUST-REVERSER POSITION LIGHT REMAINS ON WITH DOORS IN THE FORWARD-THRUST POSITION.		
Position switch on reverser actuator defective.	Perform continuity check of position switch.	Replace actuator.
Position switch plunger or plunger spring defective.	Check for plunger movement as actuator is cycled.	Replace actuator.
Defective wiring between position switch and indicator light.	Perform continuity check of indicator light circuit.	Repair as necessary.
THRUST-REVERSER CHATTERS IN FORWARD-THRUST POSITION.		
Defective guide lock (5003, 5042 and up).	Remove guide lock and inspect for broken or damaged parts.	Repair as necessary.
Broken guide spring. (5001, 5002, 5004 through 5041).	Visually check the guide spring.	Replace the spring as needed
Improperly adjusted guide spring. (5001, 5002, 5004, through 5041).	Visually check the guide spring position. (See figure 78-12.)	Adjust the spring as needed.
Actuator not rigged to proper linkage over-center position.	Check actuator rigging using actuator rigging tool.	Rig the actuator as needed.

78.4 REMOVAL AND REPLACEMENT.

Special Tools Required

Nomenclature	Description	Use and Application
Door Unlock Tool	Lockheed Part No. 11098-MEU-1.	To unlock thrust reverser doors.

78.4.1 REMOVAL. The procedure for removal is identical on all four engines. To aid the removal operation, the process is broken down into separate operations as follows:

Preparation (Paragraph 78.4.1.1)

Door and supporting link removal **(Paragraph** 78.4.1.2)

Overcenter links and sliding fittings removal **(Paragraph** 78.4.1.3.)

Actuator removal (Paragraph 78.4.1.4.)

Longeron assembly removal (Paragraph 78.4.1.5)

For removal of the complete thrust-reverser, it is recommended that these operations be performed in the sequence given.

78.4.1.1 PREPARATION.

WARNING

Use extreme caution when working on the thrust-reversers with electrical and hydraulic power on the airplane or when the thrust-reverser accumulator is charged. Disconnect all power and bleed off accumulator pressure by actuating reversers before beginning this procedure.

a. Be sure the reverser levers on the throttles are all in the forward-thrust position with the throttles retarded to "OFF." Disengage the thrust-reverser circuit breakers.

b. Remove the lower aft nacelle access door for access to the reverser.

c. Remove the thrust-reverser ground lock.

d. Remove the engine tailpipe, if installed. It is necessary to remove the engine only if the thrust-reverser actuator is to be removed.

e. Remove the ejector assembly, shown in figure 78-4, from its supporting brackets.

f. Using the door unlock tool as shown in figure 78-4, push the guide forward past the locked position.

g. Grasp the doors by the forward edges and pull them to the reverse thrust position.

78.4.1.2 DOOR AND SUPPORTING LINK REMOVAL. (See figure 78-5.)

Note

Paragraph 78.4.1.1 is to be accomplished prior to door removal. The procedure for removing one door is given. The procedure for both doors is identical.

a. Disconnect the follower links (1), from the longeron (2), and driver link hinge support fitting (3).

b. Disconnect the idler link (4) from the door.

c. Disconnect the driven link (5) from the door.

d. Remove the door.

e. Disconnect the follower links from the door.

78.4.1.3 OVERCENTER LINKS AND SLIDE FITTINGS REMOVAL. (See figure 78-6.)

Note

Paragraph 78.4.1.1 and paragraph 78.4.1.2 are to be accomplished prior to links and fittings removal.

a. Disconnect actuator rod end from guide (5).

b. Disconnect lower guide links (6) from guide.

c. Disconnect lower drive link (18) from lower driven link (9), and remove driven link.

d. Disconnect upper guide links (4) from guide (5).

e. Disconnect upper drive link (21) from driven link (8) and remove driven link from the drive link hinge support fitting (7).

78.4.1.3 (Cont).

f. Remove drive link hinge support fitting (7). It is necessary to gain access to the adjacent thrust-reverser to remove the bolts that go through the nacelle firewall.

Note

Use care in removing this assembly to retain filler shims (22) with particular hole locations.

Note

If necessary, bushings in drive link hinge support fitting may be removed with arbor press.

g. Remove positioning disc (23) and retaining discs (24).

h. Remove ground lock guide assembly (17) by removing attaching screws in lower nacelle skin. Remove guide (5), track (19), and ground lock stops (16 & 20) from firewall. It is necessary to gain access to the adjacent thrust-reverser to remove the bolts that go through the nacelle firewall.

i. Disconnect guide links (4 & 6) from drive links (21 & 1).

CAUTION

Failure to reinstall track and drive link hinge support fitting attachment bolts in adjacent thrust-reverser can result in damage to the linkage mechanism if the adjacent thrust-reverser is actuated.

j. Remove screws attaching retainer assembly (1) and plate (2) to guide (5).

k. Remove nut (13), lockwasher (12), plunger (14), outer spring (10), inner spring (11), and plunger guide (15).

78.4.1.4 ACTUATOR REMOVAL.
(See figure 78-7.)

Note

Paragraph 78.4.1.1 is to be accomplished prior to actuator removal.

CAUTION

The actuator over-center position must be re-rigged with rigging tool each time an actuator is replaced. (See figure 78-10 or 78-16.) Leakage at thrust-reverser actuators may not necessarily require actuator replacement. The piston rod seal may produce a maximum of one drop of leakage for each 10 cycles of operation. Cylinder end and switch plunger seals may show evidence of fluid seepage up to a maximum of one drop at each leak point. It is not considered necessary to inspect the actuator for leakage unless hydraulic fluid appears at the actuator tunnel drain outlet.

a. Disconnect the actuator electrical connector from its receptacle in the frame adjacent to the tunnel.

b. Remove the electrical fairlead (4) in the upper tunnel cover.

c. Remove the upper tunnel cover (5).

d. Disconnect the tunnel drain attached to the lower tunnel cover (2).

e. Remove the lower tunnel cover.

f. Remove the side tunnel cover (1).

g. Remove the lockwire from the actuator rod end locknut (8), loosen the locknut and remove the rod end (10).

Note

Use care in loosening the rod end jamnut so the lock (9) is not damaged.

h. Disconnect the two hydraulic metal flex lines.

i. Remove the actuator rod seal retainer (7) and the seals (6).

j. Disconnect the actuator from the actuator support fitting (3).

k. Move the actuator aft until the fixed end of the actuator clears the actuator support fitting and withdraw the actuator from the tunnel.

78.4.1.5 LONGERON ASSEMBLY REMOVAL.
(See figure 78-8.)

Note

Paragraphs 78.4.1.1 and 78.4.1.2 are to be accomplished prior to longeron removal.

78.4.1.5 (Cont).

a. Remove the eight bolts that attach the longeron to the adjacent fittings in the aft nacelle.

Note

Use care in removing the longeron assembly to keep the shims (2) with their original hole locations.

78.4.1.5 (Cont).

CONSUMABLE MATERIALS REQUIRED FOR THRUST-REVERSER REPLACEMENT.

MATERIAL	DESIGNATION	MANUFACTURER
Compound, Firewall Sealant	Pro-Seal 700	Essex Chemical Co. 19451 Susana Rd. Compton, CA 90221
Compound, High Temperature Anti-Seize	Fel-Pro C-5	FEL-PRO, Inc., 7480 No. McCormick Blvd. Skokie, IL 60076
Compound, High Temperature Lubricating	Led-Plate 250	Armite Laboratories 1845-49 Randolph St. Los Angeles, CA 90001
Dry Film Lubricant	Lubri-Bond "A"	Electrofilm, Inc., 7116-21 Laurel Canyon Blvd. North Hollywood, CA 91605

78.4.2 GENERAL, REPLACEMENT. The procedure for installation is identical on all four engines. It is necessary to use a torque wrench for the installation procedure and to have on hand a supply of the consumable materials listed above. The Fel-Pro C-5 compound is to be used on all threads. The Led-Plate 250 should be applied to all sliding surfaces, particularly to the ball and straight bearings. Pro-Seal 700 is used for sealing the seams and holes in the actuator cooling tunnel covers. To aid the replacement operation, the process is broken down as follows:

Preparation (Paragraph 78.4.2.1)

Longeron Installation (Paragraph 78.4.2.2)

Sliding Fitting Installation (Paragraph 78.4.2.3)

Actuator Installation (Paragraph 78.4.2.4)

Hinge Support Fitting Installation (Paragraph 78.4.2.5)

Overcenter Link Installation (Paragraph 78.4.2.6)

Door Installation (Paragraph 78.4.2.7)

Tunnel Cover and Ejector Installation
(Paragraph 78.4.2.8)

For installation of the complete thrust-reverser, it is recommended that these operations be performed in the sequence given. When the installation is complete or at the point indicated, rig and adjust the reversers in accordance with paragraph 78.5.

78.4.2.1 PREPARATION. The engine must be removed from the nacelle for installation of the actuator and tunnel covers. All other installations may be made with only the tail pipe removed from the engine. The nacelle stang fairing assembly must be installed before installation of the thrust-reverser.

78.4.2.2 LONGERON ASSEMBLY INSTALLATION. (See figure 78-8.)

Note

Paragraph 78.4.2.1 is to be accomplished prior to longeron installation. Use anti-seize compound on all threads.

a. Replace the previously removed shims in their proper position between the longeron assembly and the aft nacelle frame, and install the longeron assembly.

78.4.2.3 SLIDING FITTINGS INSTALLATION. (See figure 78-6.)

Note

Paragraph 78.4.2.1 is to be accomplished prior to fitting installation. Use anti-seize compound on all threads.

a. Install plate (3) and retainer assembly (1) on guide (5). Use AN960C10L washers (2) for shims as required to allow free installation of actuator rod end bolt through plate and guide when screws are tight.

b. Install guide (5) in track (19), clevis end forward.

c. Install track (19) and ground lock stops (16 & 20) on firewall. Position ground lock stops as shown in figure 78-6. It is necessary to gain access to the adjacent thrust-reverser to install the bolts that go through the nacelle firewall.

d. Install ground lock guide assembly (17).

78.4.2.4 ACTUATOR INSTALLATION. (See figure 78-7.)

Note

Paragraph 78.4.2.1 is to be accomplished prior to installation of the actuator.

78.4.2.4 (Cont).

CAUTION

The actuator overcenter position must be rerigged according to figure 78-10 or 78-16, each time an actuator is replaced.

Note

Use anti-seize compound on all threads.

a. Move the actuator rod to the fully retracted position and remove the rod end (10), lock (9), and lock nut (8), if installed.

b. With the ports positioned on top of the actuator, insert the rod through the seal hole in the aft end of the tunnel.

c. Connect the fixed end of the actuator to the forward fitting in the tunnel. Torque the bolt to 220 to 260 inch-pounds or the nut to 180 to 240 inch-pounds.

d. Install the seals (6) on the actuator shaft.

e. Install the seal retainer (7).

f. Install the locknut (8) and the lock (9) on the actuator rod end. Install the rod end (10). Do not tighten until proper position is established with tool Lockheed part number 11097-MEU or 11097-MEU-1.

CAUTION

The rod end lock may be damaged if the rod end is turned to rotate the shaft and the lock is not fully engaged in the slot of the actuator shaft.

Note

If only the actuator is being installed, the overcenter rigging should be accomplished at this point (see figure 78-10 or 78-16) and the replacement procedure continued with paragraph 78.4.2.8.

78.4.2.5 DRIVE LINK HINGE SUPPORT FITTING INSTALLATION.

(See Figure 78-6.)

Note

Paragraph 78.4.2.1 and paragraph 78.4.2.3 are to be accomplished prior to support fitting installation.

a. Install positioning discs (23) in upper leg of the drive link hinge support fitting.

b. Install positioning discs (23) and retaining discs (24) in lower leg of the drive link hinge support fitting.

Note

The slotted holes in these discs are not centered. Be sure adjacent discs have identical alignment.

c. Temporarily install the ground locking bar, noting its proximity to the track. The locking bar should be installed as close as possible to the track.

Note

The positioning and retaining discs may be rotated 180 degrees to change the relative position of the locking bar to the track.

d. Install the drive link hinge support fitting to the firewall. Reinstall the filler shims (22). It is necessary to gain access to the adjacent thrust-reverser to install the bolts that go through the nacelle firewall.

Note

The overcenter position rigging may be accomplished at this point of the installation. (See figure 78-10 or figure 78-16.)

78.4.2.6 OVERCENTER LINK INSTALLATION.

Note

Paragraph 78.4.2.1, paragraph 78.4.2.3, and paragraph 78.4.2.5 are to be accomplished prior to the overcenter link installation. Use anti-seize compound on all threads.

a. Install guide links loosely on driver links. Locate bolt head as shown in figure 78-9.

b. Insure that the driven link shafts are clean and free from dirt. Insert driven link (9), figure 78-6, shaft through bushings on drive hinge support fitting.

c. Position drive link (2) on hex end of driven link shafts (1), as shown in figure 78-9, so that the following conditions are met:

(1) The raised part number on the drive link is on the side of the drive link toward firewall.

(2) The concave surface of the link is positioned as shown in figure 78-9.

(3) The ball end of the drive link is aligned with the driven link as shown in figure 78-9.

78.4.2.6 (Cont).

d. Install washer and nut to retain drive links, figure 78-9, to driven links. Torque nut 180-240 inch-pounds.

e. Connect free ends of guide links to guide. (Bolt heads away from firewall). Torque all guide link bolts to 220-260 inch-pounds or nuts to 180-240 inch-pounds.

78.4.2.7 DOOR INSTALLATION. (See figure 78-5.)

Note

Paragraph 78.4.2.1, paragraph 78.4.2.2 and paragraph 78.4.2.6 are to be accomplished prior to door installation.

Use anti-seize compound on all threads.

a. The procedure for installation is identical for both doors. Install the upper door first.

b. Connect the straight bearing ends of the idler link (4) to the idler link support hinge fitting (2). Use one thick washer under the nut. The bolt head should be toward the engine centerline. Torque the bolts from 220 to 260 inch-pounds or the nuts from 180 to 240 inch-pounds.

c. Connect the ball bearing end of the follower links (1) to the door. Torque the bolts from 220 to 260 inch-pounds.

d. Connect the driven links to the doors with the bolt heads away from the engine centerline. Use two thick washers under the nut. Torque the bolts from 220 to 260 inch-pounds or the nuts from 180 to 240 inch-pounds.

e. Connect the idler link (4) to the door with bolt heads away from the engine centerline. Use two thick washers under the nut. Torque the bolts from 220 to 260 inch-pounds or the nuts from 180 to 240 inch-pounds.

f. The follower links (now attached to the doors) have their straight bearing ends installed as follows:

   (1) One link end is attached to the drive link support hinge fitting (3), one thick and one thin washer under nut.

   (2) One link end is attached to the idler link support hinge fitting (2), one thick and one thin washer under nut, bolt head away from engine centerline.

Torque the bolts from 220 to 260 inch-pounds or the nuts from 180 to 240 inch-pounds.

Note

Move the doors manually to check freedom of movement of the links and sliding fittings.

Note

The door preload adjustment may be accomplished at this point (see figure 78-11).

g. As shown in figure 78-6, install plunger guide (15), plunger (14), along with outer spring (10), and inner spring (11), lockwasher (12), and nut (13) on drive link hinge support fitting (7). Lockwire together, nut (13) and lockwasher (12). Check engagement as required in figure 78-11.

78.4.2.8 ACTUATOR TUNNEL COVER AND EJECTOR.

Note

Paragraph 78.4.2.4 is to be accomplished prior to this operation.

a. Install the actuator covers (1), (2), and (5) of figure 78-7. Seal with firewall sealant.

b. Install the electrical fairlead (4). Seal with firewall sealant.

c. Install the ejector assembly shown in figure 78-4.

78.5 RIGGING AND ADJUSTMENT.

Tools Required

Nomenclature	Description	Use and Application
Torque Wrench	10, 12 or 15 inch (effective length). Up to 1200 in. lb. torque capacity.	To measure the force necessary to unlock the thrust-reverser doors.
Hydraulic Handpump with Gage	0-500 PSI capacity (MIL-H-5606 hydraulic fluid)	To pressurize the actuator for positioning at the full-forward or full-reverse position.

78.5 (Cont).

Tools Required (Cont)

Nomenclature	Description	Use and Application
Door Unlock Tool	Lockheed Part No. 11098-MEU-1	To unlock the thrust-reverser doors during the checkout.
Thrust-Reverser Over-center Rigging Tool	Lockheed Part No. 11097-MEU (Optional, Part No. 11097-MEU-1)	To rig the overcenter position of the thrust-reversers.

78.5.1 GENERAL. The thrust-reversers are rigged in the forward thrust position to assure the proper dimensional relationship between the actuator limit switches and the over-center position, and to assure the proper over-center preload. On airplanes 5001, 5002, 5004 through 5041, adjustment of the guide spring is also required.

Note

The thrust-reversers should be rigged before engine installation, but rigging may be accomplished after the engine is installed if tailpipe is removed.

Rigging and adjustment are identical for all four thrust-reversers.

78.5.2 OVER-CENTER POSITION RIGGING. Over-center position rigging is accomplished as shown in figure 78-10, using tool part number 11097-MEU. Refer to figure 78-16 when optional tool part number 11097-MEU-1 is used.

78.5.3 DOOR PRELOAD ADJUSTMENT. The door preload adjustment is accomplished after the over-center position is rigged as shown in figure 78-11.

78.5.4 GUIDE SPRING ADJUSTMENT. On aircraft not incorporating Service Bulletin 329-90, the guide spring adjustment is accomplished after the door preload adjustment as shown in figure 78-12.

78.6 THRUST-REVERSER MINOR REPAIR.

CAUTION

Welding should not be used as a method of repair for the sheet metal components of the thrust reverser doors.

78.6.1 DOOR INNER SKIN BLIND RIVET REPAIR. See figure 78-13 for the number of loose blind rivets permitted on the door inner skin. Replace loose blind rivets as follows:

a. Punch out the rivet core.

b. Remove the locking collar.

Note

The locking collar may be loosened by inserting a slightly tapered drill rod tightly into the rivet sleeve and rocking the rod back and forth.

c. Drill out the rivet sleeve.

d. Drill the existing rivet hole oversize, using a No. 10 drill.

e. Install a CR2643-6-2 Cherry Lock blind rivet.

Note

Replace the rivets marked "A" in rows 3 and 4 with CR2643-6-3 Cherry Lock blind rivets.

78.6.2 DOOR END PLATE CRACK REPAIR. (See figure 78-15.) Door end plates should be repaired if cracks exceed two inches in length, as shown on sheet 1 of figure 78-15. Repair the door end plates as follows:

a. Fabricate a reinforcing clip from Type 321, Specification MIL-S-6721 corrosion-resistant steel, 0.125 inch thick, as shown in figure 78-15.

b. Remove the NAS1633C1 screws that attach the door inner skin, and attach the reinforcing clip using NAS1633C4 screws and NAS679C3W nuts. Use one AN960C10 washer and one AN960C10L washer under the head of each screw.

Note

Type A286 sheet material is used exclusively throughout the thrust-reverser doors. This material is best drilled using slow (500-1000 RPM) drill speeds and Cleveland Twist Drill Co. No. 973, or equivalent drills. These drills have no body clearance and a 118-degree point.

78.6.2 (Cont).

c. With the clip installed, locate and drill a hole through clip and door end plate with a No. 10 drill, as shown in figure 78-15.

d. Secure the clip to the end plate with an NAS1633C4 screw, AN960C10 washer, and NAS679C3W nut.

e. Determine the exact end of the crack by a penetrant check, and stop-drill with a No. 30 drill.

CAUTION

Stop-drill the crack only where the crack has progressed clear of the adjacent structure.

f. If the crack extends 1/4 inch above the door inner skin, as shown on sheet 2 of figure 78-15, install a reinforcing doubler on the outside surface of the end plate in addition to the clip.

g. Fabricate the doubler from Type 321, Specification MIL-S-6721 corrosion-resistant steel, 0.080 inch thick, as shown in figure 78-15.

h. Locate the doubler as shown in figure 78-15 and install eight NAS508M5-7 or MS20615-5MP7 rivets.

i. Drill a hole in the doubler, using the hole in clip previously installed as a guide, with No. 10 drill. Secure the clip, end plate, and doubler together with an NAS1633C4 screw, AN960C10 washer, and NAS697C3W nut.

78.6.3 DOOR AFT "Z" ANGLE WEAR REPAIR. (See figure 78-14.) A rub angle should be installed when wear area exceeds 1-1/2 inches, as shown in figure 78-14. Install the rub angle as follows:

a. Fabricate a rub angle from Type A286, AMS-5525 steel, 0.040 inch thick, as shown in figure 78-14.

Note

This angle may be made from part number 140-5287-19 frame. Approximately five rub angles can be made from one frame.

b. Drill out the three blind rivets that attach the inner door skin to the aft "Z" angle, as outlined in paragraph 78.6.1., steps (a), (b), and (c). (See figure 78-14.)

c. Locate the three No. 20 holes from existing holes in the inner skin, and drill three No. 20 holes in the rub angle.

d. Temporarily locate the rub angle, and drill the three No. 20 holes oversize with a No. 10 drill. Drill a single No. 10 hole through the rub angle and face of the "Z" angle.

e. Drill the single No. 10 hole in the rub angle oversize with a 0.250 drill.

f. Install four CR2643-6-2 Cherry Lock rivets, using an AN960C10L washer under the single rivet that attaches the rub angle to the face of the "Z" angle, as shown in figure 78-14.

78.6.4 DOOR OUTER SKIN CRACK REPAIR. (See figure 78-17.) Repair for a door outer skin crack depends on the row of rivets affected by the crack. The various rows are identified by letter in Detail 1. Reference is made to these rows identified in the repair information below.

a. In row A, E, or F cracks may run from rivet to rivet as shown in Detail 2, or from rivet toward the edge of skin as shown in Detail 3. Any of these types of cracks must have at least three non-defective rivets on each side in the same row. If this condition is met, no repair is required for a rivet to rivet crack. A crack extending toward the edge of the door skin should be stop-drilled with a No. 30 drill only when it has progressed clear of adjacent or supporting structure. Such a crack which has not progressed clear of adjacent structure is acceptable without repair. If it is desired from a quality standpoint, rivet-to-rivet cracks or stop-drilled rivet-to skin-edge cracks may be covered by a locally fabricated 0.04-inch-thick type 301 (1/4, 1/2 or full hard) stainless steel doubler, as shown in details 4 and 5. Such a doubler should be attached with BR125-6, -4 or -5 or LS10795M5-4 or -5 rivets or, if required, CR2538-5-2 or NAS1739M5-2 blind rivets.

b. In row B, C, or D cracks along the row may be repaired, provided the crack does not extend through five or more adjacent rivets and the location will permit the installation of a locally fabricated doubler as indicated below. Material and attachments for such a doubler are the same as the doubler described in step (a). For repair of a crack that involves only two rivets the doubler shall be long enough to pick up one non-defective rivet at each end, maintaining 0.31 inch minimum edge distance as shown in Detail 6. Where three rivets are involved, the doubler should pick up two non-defective rivets on each end, with the edge distance as shown in Detail 7. Where four rivets are involved, three non-defective rivets should be picked up at each end, as shown in Detail 8. All of these doublers should be wide enough to pick up an additional row of rivets on each side of the crack as shown in the details cited.

78.6.5 REPAIR OF DRY FILM LUBRICATED PARTS. If the dry film lubrication coating is worn, but the base material is not galled or worn excessively, the surface may be re-dry lubed as follows:

a. Scrub area to be dry film lubricated with trichloroethylene or equivalent. Dry thoroughly.

b. Apply Lubi-Bond "A" (Electrofilm, Inc., North Hollywood, CA) evenly in uniform cross coats. Allow each coat to flash dry before applying next coat.

c. Allow dry film lubricant to air cure at room temperature a minimum of 8 hours.

d. After cure period the coating shall be smooth and free from blisters and other surface imperfections.

78.6.5 (Cont).

e. Check adhesion by placing a strip of masking tape on the dry film lubricated surface; press firmly, and remove in one abrupt movement. The dry film lubricant shall not be removed to the bare metal. Slight discoloration of the tape is acceptable.

78.7 PERIODIC LUBRICATION. Lubrication of the link bearings is required every 600 hours as follows:

a. Remove the bearings indicated in figure 12-35 in accordance with paragraph 78.4.1.2.
b. Clean bearing surface with dry-cleaning solvent, Federal Specification P-S-661, or equivalent. Wipe dry with lint free cloth.

CAUTION

Wire brush should not be used for cleaning as this may damage dry film lubricant on some parts.

c. Apply Led-Plate 250 to the bearing surfaces in all links and reassemble in accordance with paragraph 78.4.2.7.

Note

No adjustment or rigging is required after accomplishment of the above provided only the doors and links are removed for lubrication.

78.8 INSPECTION AND OVERHAUL REQUIREMENTS.

78.8.1 600 HOUR INSPECTION.

78.8.1.1 DISASSEMBLY. Removal is accomplished in accordance with paragraph 78.4.1. It is not necessary to remove the actuator and associated structure (paragraph 78.4.1.4) for overhaul. If portable inspection methods are available, it may also be unnecessary to remove the longeron fitting assembly (paragraph 78.4.1.5).

78.8.1.2 CLEANING. Wash all removed metal parts with dry cleaning solvent, Federal Specification P-S-661, or equivalent. Use stiff bristle brush to remove stubborn accumulation of foreign matter. Drain and dry parts thoroughly with lint-free air.

CAUTION

For cleaning parts, use only a natural or nylon bristle brush. Wire brush may damage dry film lubricant on some parts.

78.8.1.3 INSPECTION.

a. Examine all sheet metal parts for cracks, burrs and wear.
b. Visually check door and longeron sheet metal parts for cracks.
c. Check security of all rivets.
d. Examine all threads for stripping.
e. Check condition of silver plating on all plated nuts.
f. Check the ID of both the cylindrical and the spherical bearing surfaces in all links (figure 78-5, Items 1, 4 and 5; figure 78-6, Items 8 and 21) for condition of the dry film lubricant.
g. Inspect track as shown in figure 78-19.

78.8.1.4 REASSEMBLY. Replacement and adjustment is accomplished in accordance with paragraphs 78.4.2 and 78.5.

78.8.2 OVERHAUL.

78.8.2.1 Overhaul will consist of an inspection in accordance with 78.8.1 plus the following:

a. The following parts are to be checked using the penetrant method (refer to Section 51.3): all links (figure 78-5, items 1, 4 and 5; figure 78-6, items 4, 6, 18 and 21), hinge support fitting (figure 78-6, item 7), ground lock stops (figure 78-6, items 16 and 20), plunger guide (figure 78-6, item 15), plunger (figure 78-6, item 14), guide (figure 78-6, item 5), plate (figure 78-6, item 2), track (figure 78-6, item 19), and the accessible portions of door fittings and longeron fittings.
b. Check the OD of the shaft of driven links and drive links (figure 78-6, items 8, 9, 18 and 21) together with associated bearings, races and bushings for penetration of the dry film lubricant.
c. Check condition of the internal portion of upper door, lower door and longeron fitting by X-ray inspection (refer to MIL-STD-00453B).
d. Check the plunger lock and plate (figure 78-6, items 14 and 3) for wear.
e. Examine and replace any part that fails to meet criteria given in paragraph 78.8.2.2.

78.8.2.2 PART REPLACEMENT CRITERIA.

a. Replace all parts damaged beyond simple repair.
b. Replace all bolts showing evidence of galling or scoring on shank.
c. Replace links (figure 78-5. items 1, 4 and 5, figure 78-6, items 18 and 21) and guide (figure 78-6, item 5), if the cylindrical and the spherical bearing surfaces show galling, scoring or excessive wear. Dry film lubricant may be replaced in accordance with paragraph 78.6.5.
d. Replace link (figure 78-5, item 5) if the bearing surfaces show galling, scoring or excessive wear. Dry film lubricant may be replaced in accordance with paragraph 78.6.5.
e. Replace spherical bearings if OD measures 0.7166 or less.
f. Replace cylindrical bearings if wear exceeds limits shown in figure 78-18.
g. Replace shoulder bushings in drive link support fittings (figure 78-6, item 7) if ID measures 1.2561 or more.
h. Replace guide (figure 78-6, item 5) if wear exceeds limits shown in figure 78-18.
i. Replace track (figure 78-6, item 19) if wear exceeds limits shown in figure 78-18.
j. Replace all nuts where silver plating is stripped off or penetrated.
k. Replace plate (figure 78-6, item 3) if wear penetrates hard surface. Hard surface is 0.003 thick.
l. Replace plunger (figure 78-6, item 14) if shaft OD measures .244 or less, or if surface shown in figure 78-18 is scored.

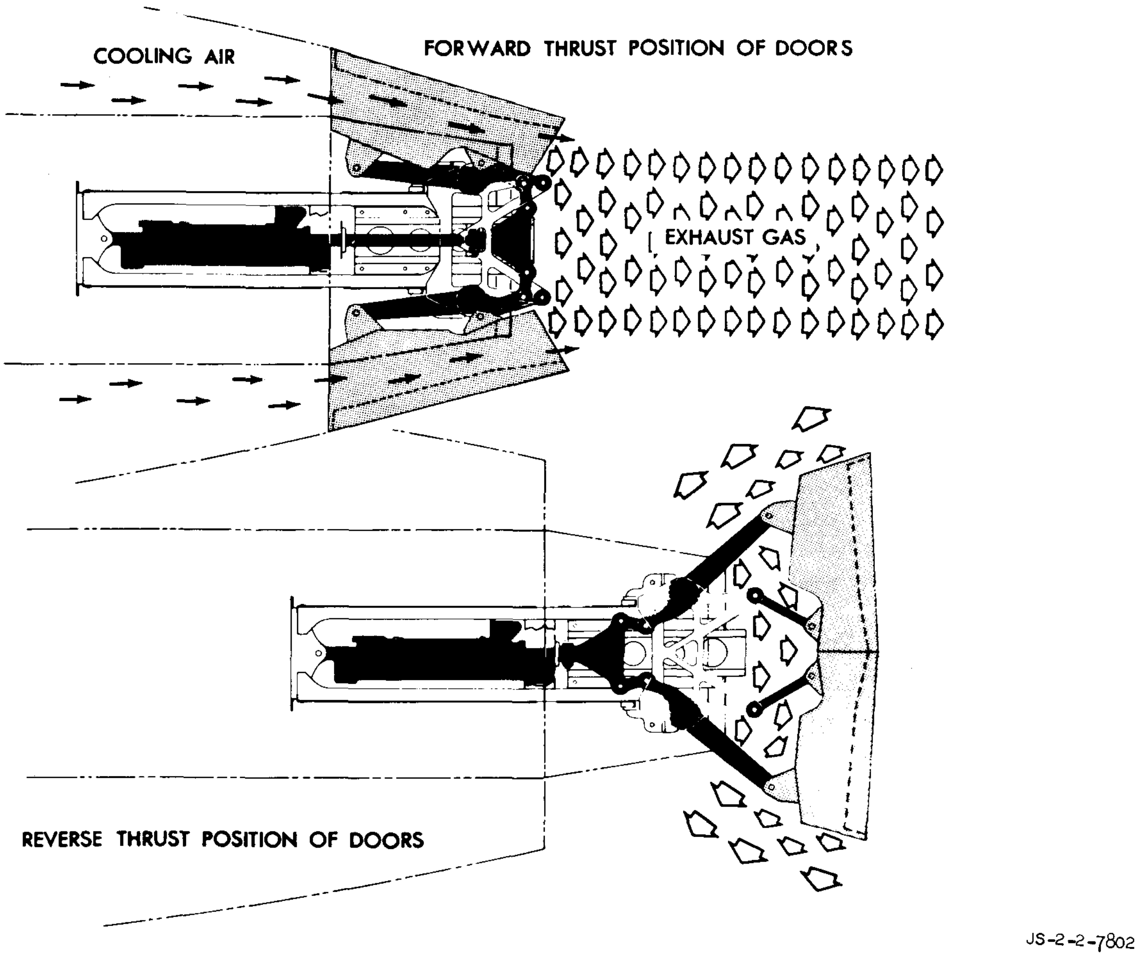
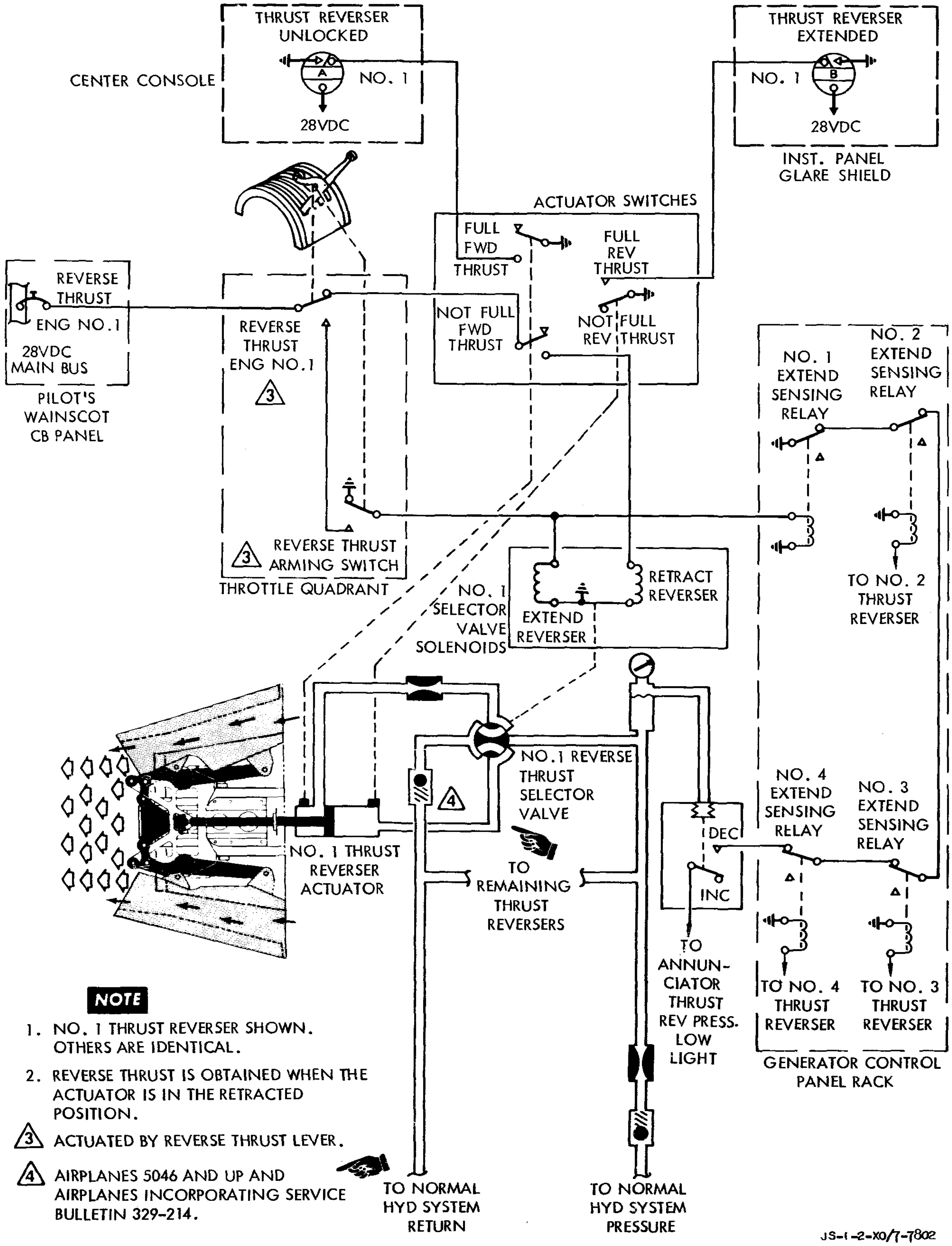

COOLING AIR
FORWARD THRUST POSITION OF DOORS
EXHAUST GAS
REVERSE THRUST POSITION OF DOORS
JS-2-2-7802

Figure 78-1. Thrust-Reverser Door Positions



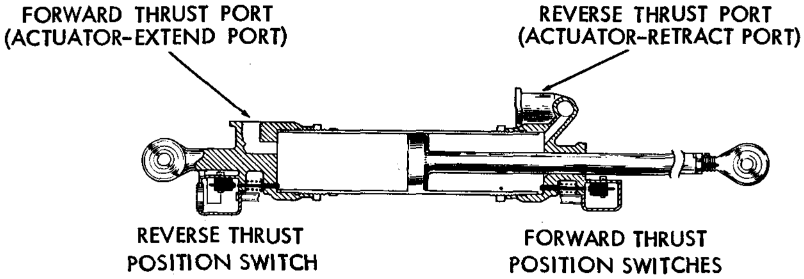
NOTE

1. NO. 1 THRUST REVERSER SHOWN. OTHERS ARE IDENTICAL.
2. REVERSE THRUST IS OBTAINED WHEN THE ACTUATOR IS IN THE RETRACTED POSITION.

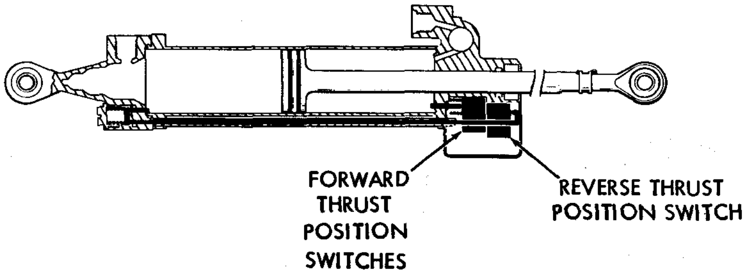
⚠3 ACTUATED BY REVERSE THRUST LEVER.

⚠4 AIRPLANES 5046 AND UP AND AIRPLANES INCORPORATING SERVICE BULLETIN 329-214.

Figure 78-2. Thrust-Reverser Schematic Diagram



ROVALCO (FORMERLY 'PNEU-TECH') ACTUATOR



BOBRICK ACTUATOR

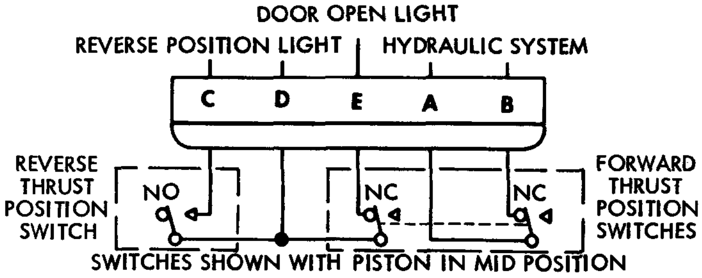


Figure 78-3. Actuator Switch Positions

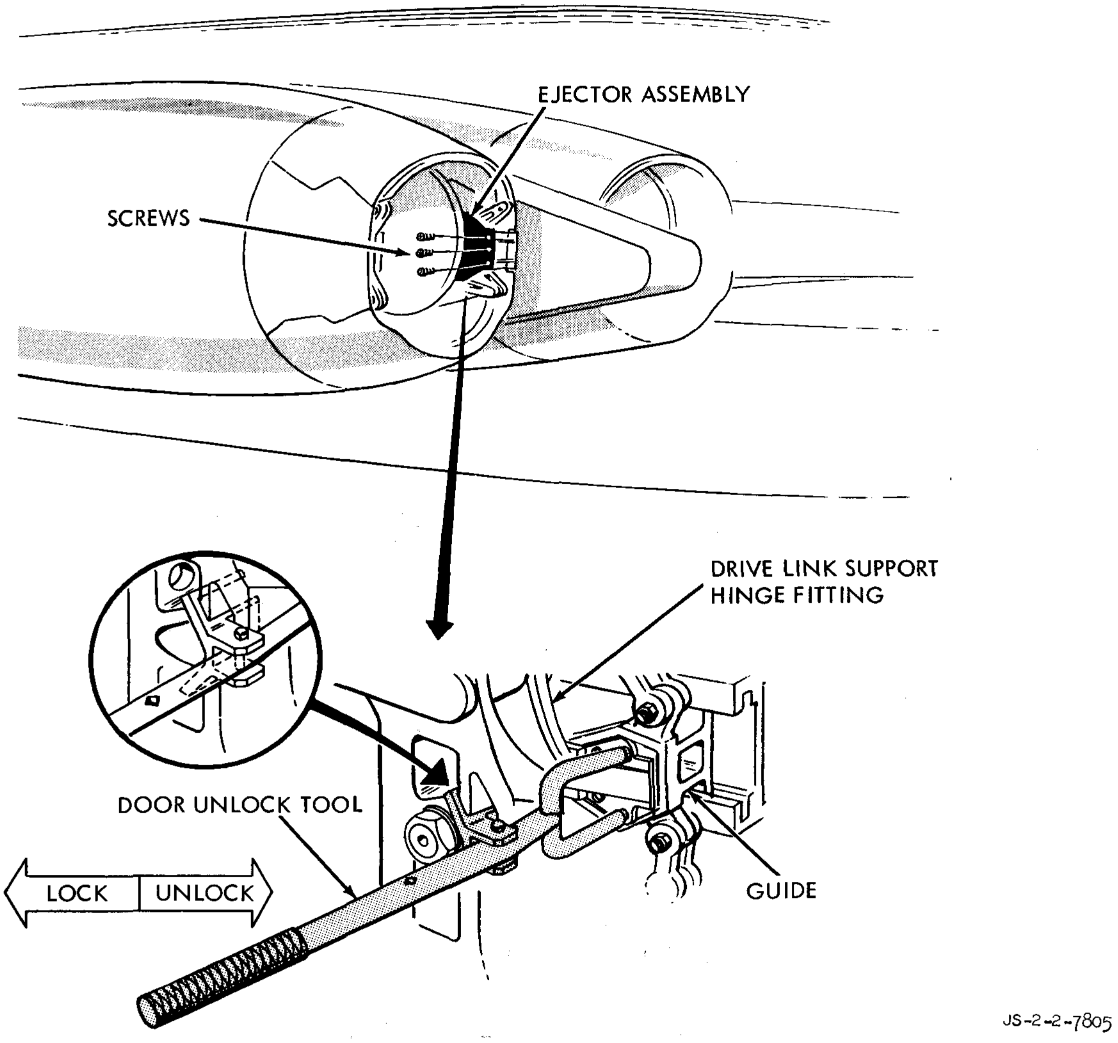


Figure 78-4. Unlocking Doors

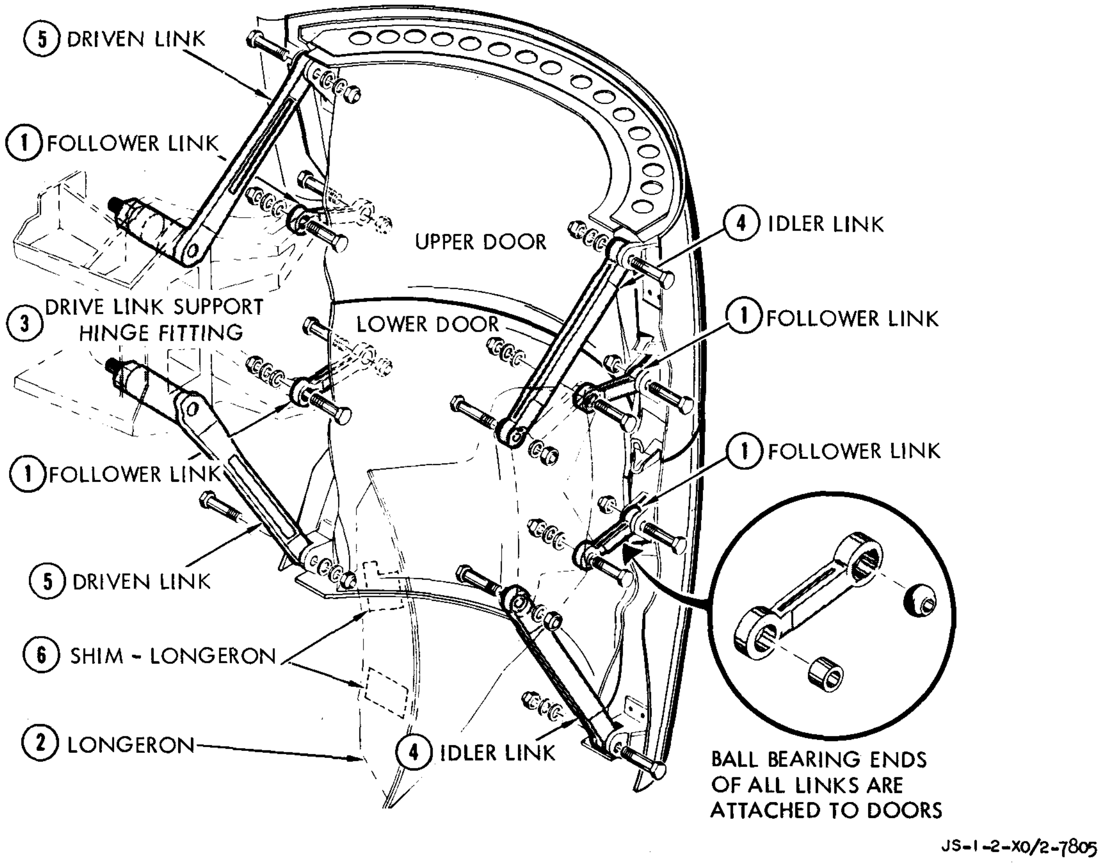


Figure 78-5. Doors and Supporting Links

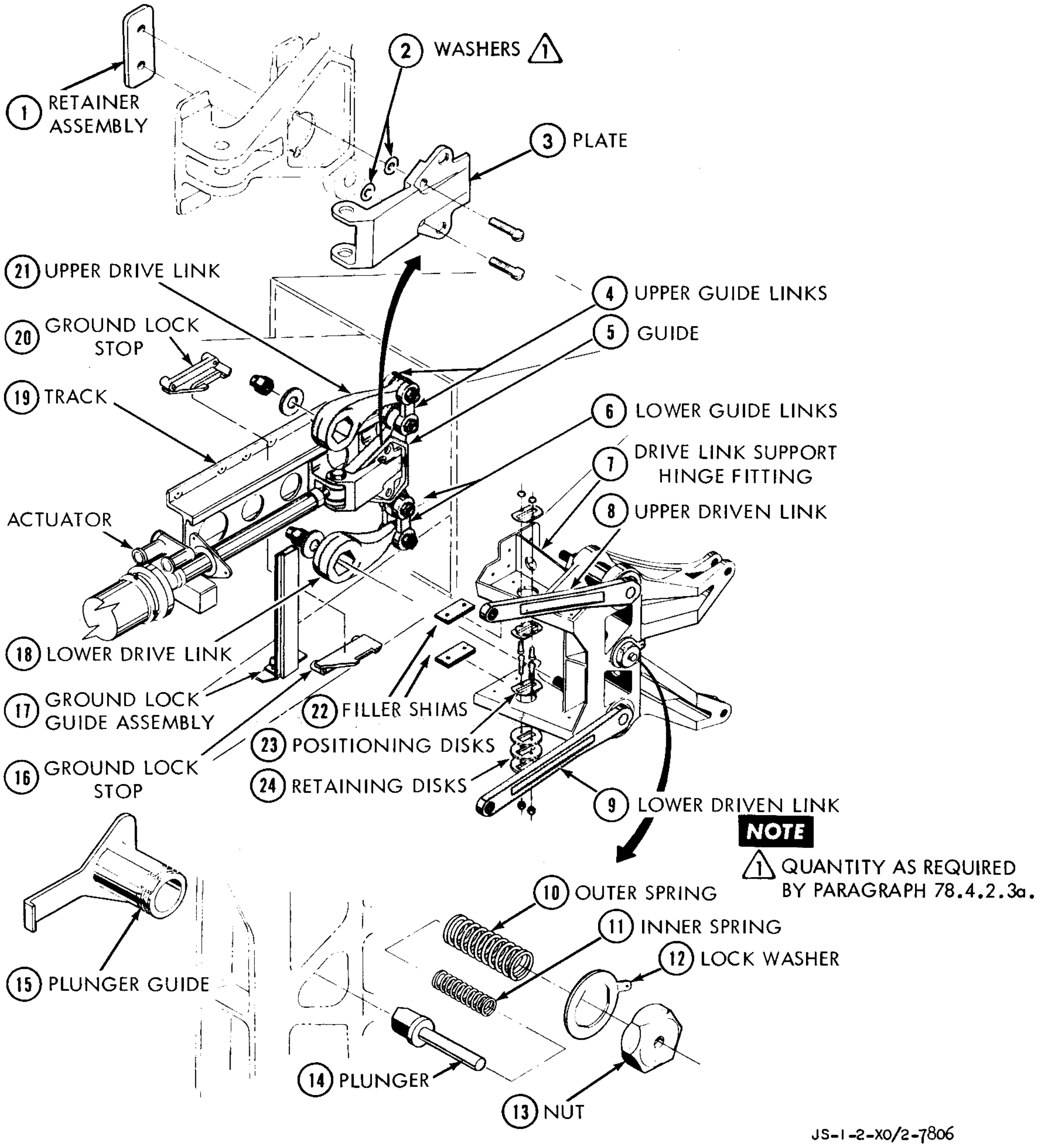


Figure 78-6. Overcenter Links and Sliding Fittings

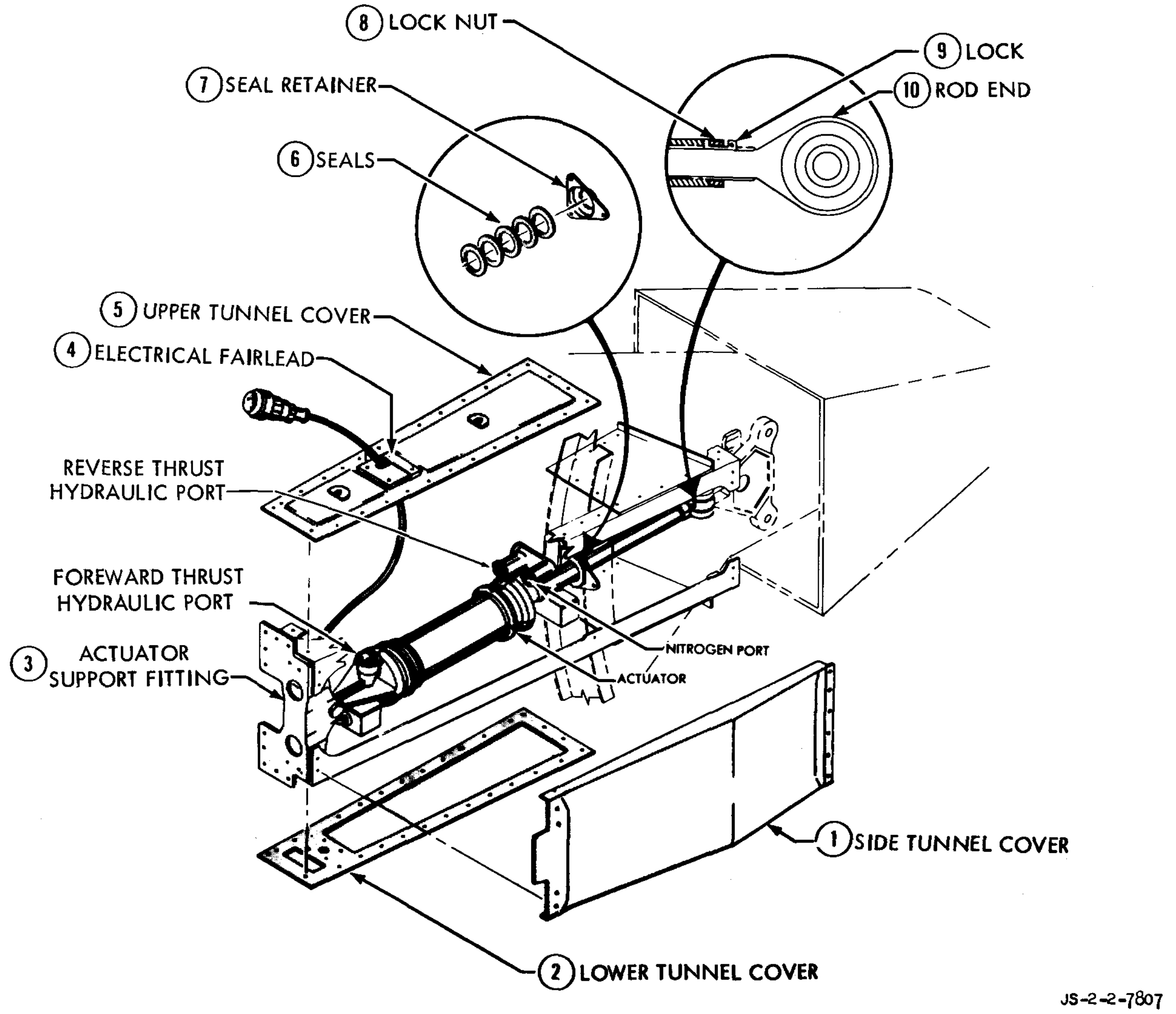


Figure 78-7. Actuator

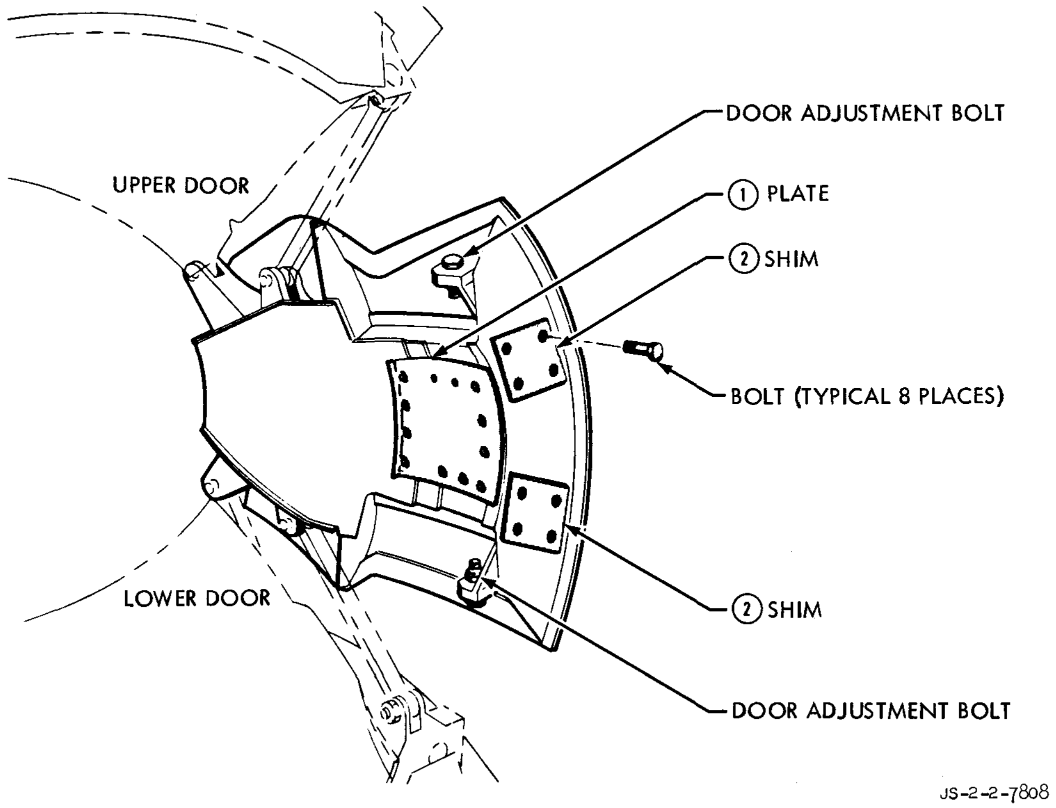


Figure 78-8. Longeron Assembly Removal

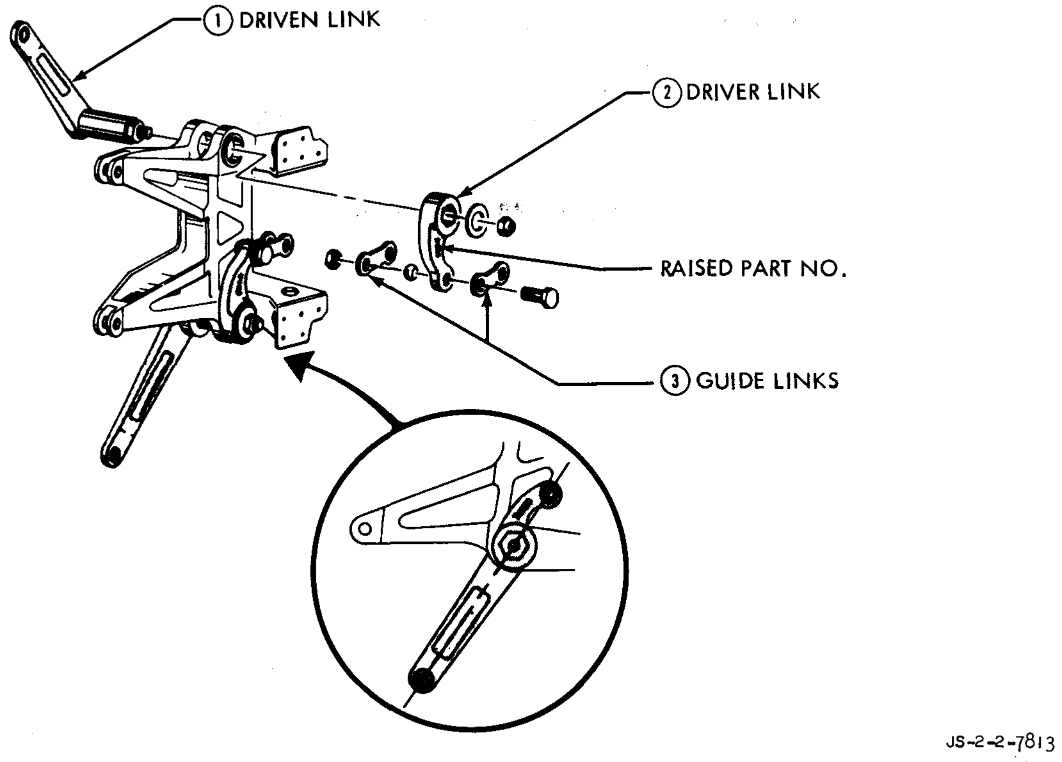


Figure 78-9. Driver Link and Driven Link Adjustment

OVERCENTER RIGGING (USING TOOL PART NO. 11097-MEU, SEE FIG 78-16 FOR USE OF OPTIONAL TOOL)

STEP 1

DISCONNECT GUIDE LINKS FROM GUIDE, DRIVE LINKS FROM DRIVEN LINKS AND DRIVEN LINKS FROM DOORS. REMOVE THE DRIVEN LINKS AND THE BOLT BETWEEN THE ACTUATOR AND THE GUIDE. (SEE FIGS 78-6 and 78-9).

STEP 2

REMOVE THE PLUNGER ASSEMBLY OR THE SCUFF PLATE FROM THE DRIVE LINK SUPPORT HINGE FITTING. REMOVE THE GUIDE SPRING FROM THE GUIDE IF INSTALLED. (SEE FIGS 78-6 AND 78-12.)

STEP 3

INSERT RIGGING TOOL INTO DRIVE LINK HINGE SUPPORT FITTING AS SHOWN.

STEP 4

CONNECT HAND PUMP TO ACTUATOR FORWARD THRUST HYDRAULIC PORT (SEE FIGURE 78-7).

STEP 5

WITH GUIDE POSITIONED AS ILLUSTRATED, ADJUST ACTUATOR ROD END SO THAT WITH 200 PSI APPLIED TO EXTEND ACTUATOR (TO INSURE PISTON BOTTOMED OUT) THE ROD END IS ALIGNED WITH THE RIGGING TOOL AND THE RIGGING PIN WILL PASS THROUGH THE TOOL, GUIDE AND ROD END.

CAUTION

ROD END LOCK MAY BE DAMAGED IF ROD ENDS IS USED TO ROTATE SHAFT AND LOCK IS NOT FULLY ENGAGED IN SLOT OF ACTUATOR SHAFT.

STEP 6

ENGAGE ROD END LOCK (9), FIGURE 78-7, TIGHTEN JAM NUT AND LOCK-WIRE.

STEP 7

REMOVE RIGGING PIN, RIGGING TOOL, AND HAND PUMP.

NOTE

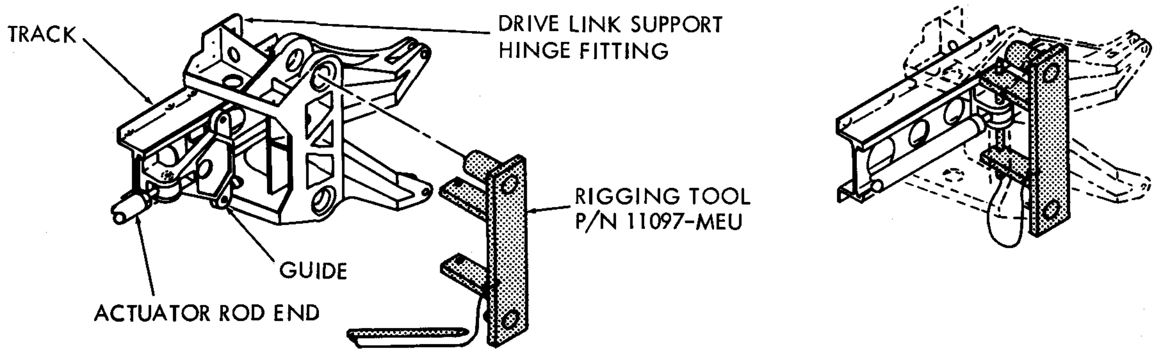
INSURE ROD END THREADS COVER WITNESS HOLE IN ACTUATOR SHAFT.

STEP 8

INSTALL GUIDE LINKS, DRIVE LINKS, DRIVEN LINKS. ATTACH DRIVE LINKS TO DOORS. RECONNECT HYDRAULIC HOSES.

STEP 9

AT THIS POINT, CONTINUE WITH DOOR PRELOAD ADJUSTMENT, STEP 3, FIG 78-12, IF REQUIRED. IF THIS ADJUSTMENT IS NOT REQUIRED, RESTORE REVERSER TO NORMAL IN ACCORDANCE WITH STEPS 7, 8 AND 9 OF FIG 78-11.



JS-2-2-7809

Figure 78-10. Overcenter Position Rigging

DOOR PRELOAD ADJUSTMENT PROCEDURE

STEP 1

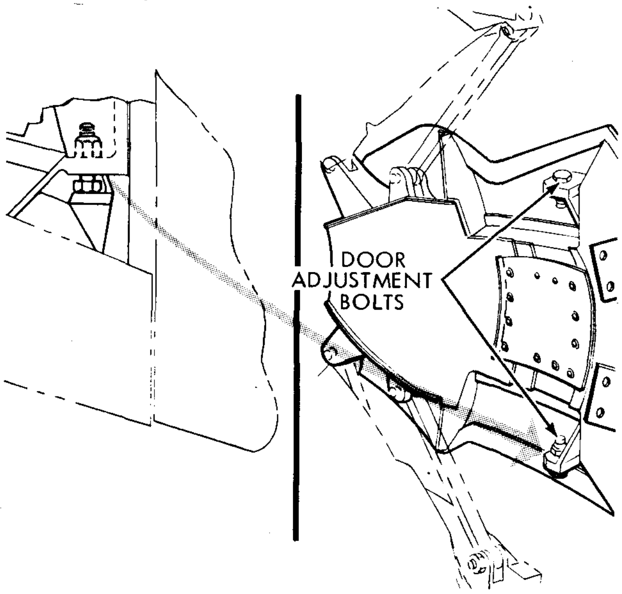
DISCONNECT GUIDE FROM ACTUATOR, IF CONNECTED. (SEE FIG. 78-6.)

STEP 2

ON 5003, 5042 AND UP REMOVE PLUNGER, PLUNGER GUIDE, INNER SPRING, AND OUTER SPRING IF INSTALLED. (SEE FIG 78-6.) ON 5001, 5002, 5004 THROUGH 5041 REMOVE GUIDE SPRING FROM GUIDE IF INSTALLED.

STEP 3

SET DOOR ADJUSTMENT BOLTS SO THAT THEY JUST ENGAGE DOORS.

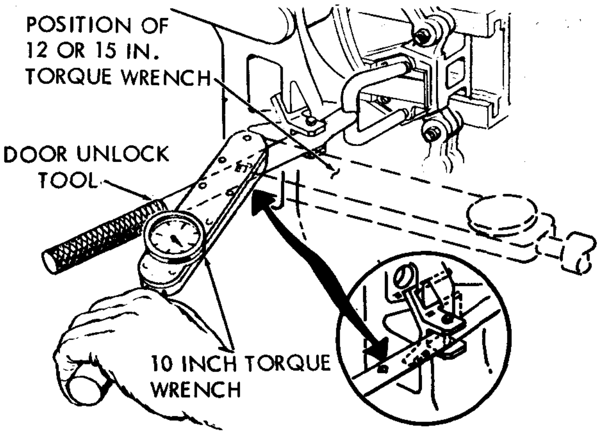


STEP 4

USING DOOR UNLOCK TOOL WITH TORQUE WRENCH AS SHOWN BELOW, CHECK TORQUE REQUIRED TO UNLOCK DOORS.

NOTE

A 1/2 INCH (FEMALE) TO 3/8 INCH (MALE) SOCKET DRIVE ADAPTER MAY BE REQUIRED WITH A 12 OR 15 INCH TORQUE WRENCH.



STEP 5

SET DOOR ADJUSTMENT BOLTS SO THAT THE TORQUE REQUIRED TO UNLOCK DOORS IS AS FOLLOWS:

10 INCH WRENCH	-	200 TO 800 INCH-POUNDS
12 INCH WRENCH	-	270 TO 1090 INCH-POUNDS
15 INCH WRENCH	-	275 TO 1100 INCH-POUNDS

STEP 6

HOLD DOOR ADJUSTMENT BOLTS WITH A WRENCH AND TIGHTEN JAM NUTS.

CAUTION

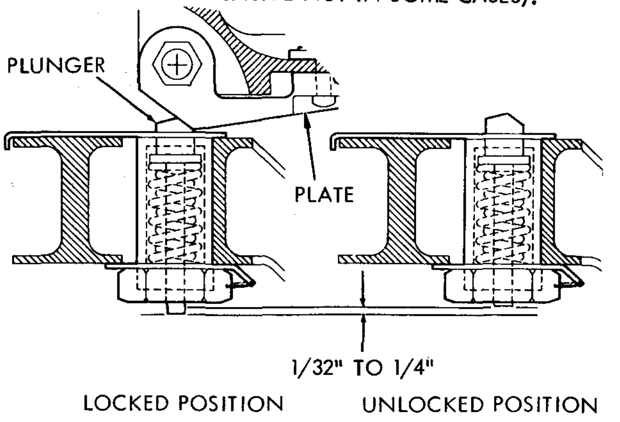
EXCESSIVE TORQUE IN STEP 7
CAN DAMAGE THE EARS OF THE PLATE.

STEP 7

CONNECT GUIDE TO ACTUATOR AND TORQUE BOLT 75-85 INCH POUNDS OR NUT 70-80 INCH POUNDS. PLATE AND RETAINER ASSEMBLY IS REQUIRED ON 5003, 5042 AND UP.

STEP 8

ON 5001, 5002, 5004 THROUGH 5041 PRIOR TO INCORPORATION OF SERVICE BULLETIN 329-90, REINSTALL GUIDE SPRING AND ADJUST IN ACCORDANCE WITH FIG. 58-12. ON 5003, 5042 AND UP OR AFTER INCORPORATION OF SERVICE BULLETIN 329-90, REINSTALL PLUNGER GUIDE AND PLUNGER ALONG WITH NUT, SPRINGS AND LOCKWASHER. POSITION LOCKWASHER TAB SO THAT IT WILL NOT INTERFERE WITH USE OF UNLOCKING TOOL AND LOCKWIRE TO NUT. CHECK ENGAGEMENT OF PLUNGER GUIDE AS SHOWN BELOW. WHEN IN LOCKED POSITION, PLUNGER SEATS AGAINST PLATE. WHEN UNLOCKED, PLUNGER SHOULD EXTEND 1/32 TO 1/4 INCH LESS THAN IN THE LOCKED POSITION. (MEASURE FROM EXTREME END OF PLUNGER ROD TO FACE OF NUT. PLUNGER MAY BE SLIGHTLY RECESSED INSIDE NUT IN SOME CASES).



STEP 9

REINSTALL EJECTOR ASSEMBLY AND RESTORE TO NORMAL.

JS-I-2-XO/2-7811

Figure 78-11. Door Preload Adjustment

GUIDE SPRING ADJUSTMENT (REQUIRED ONLY ON 5001, 5002, 5004 THROUGH 5041 PRIOR TO INCORPORATION OF SERVICE BULLETIN 329-90).

STEP 1

LOOSEN THE BOLT HOLDING THE BACK BAR (1) AND THE BASEPLATE (2) TO THE GUIDE.

STEP 2

INSTALL THE SPRING WITH THE SLOTTED END FORWARD, AND TEMPORARILY TIGHTEN THE BOLT.

STEP 3

WITH THE THRUST-REVERSER DOORS MOVED TO THE FORWARD-THRUST POSITION, CHECK FOR PROPER POSITIONING OF THE GUIDE SPRING (3) RELATIVE TO THE DRIVER LINK HINGE SUPPORT FITTING AS SHOWN. TORQUE THE BOLT TO BETWEEN 50 AND 70 INCH-POUNDS.

CAUTION

INSURE THAT THE THRUST-REVERSER OVER-CENTER LINKAGE IS IN THE LOCKED POSITION BEFORE ADJUSTING THE POSITION OF THE GUIDE SPRING.

NOTE

ADJUSTMENT OF ONE SERRATION MOVES GUIDE SPRING 0.036. THE BASEPLATE SHOULD BE SQUARELY ALIGNED TO THE GUIDE.

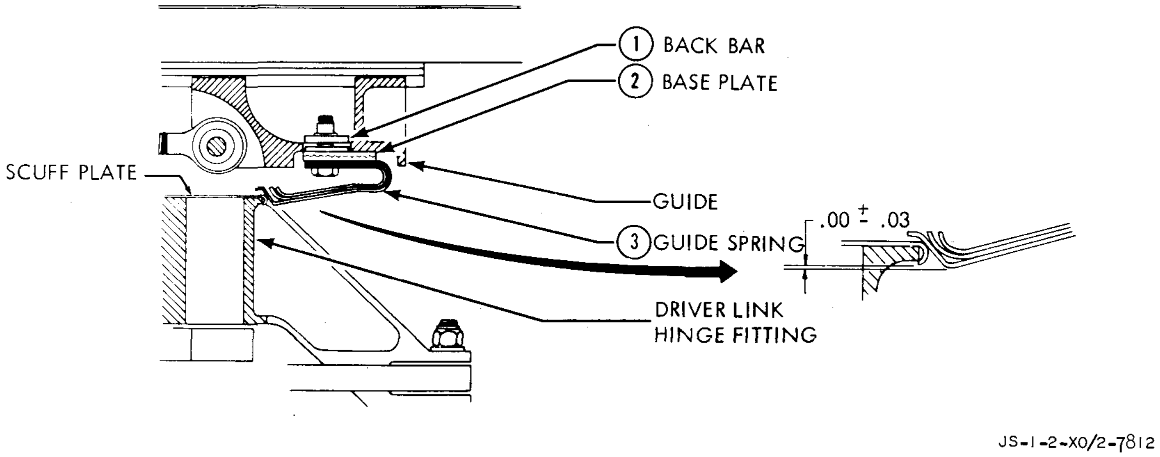
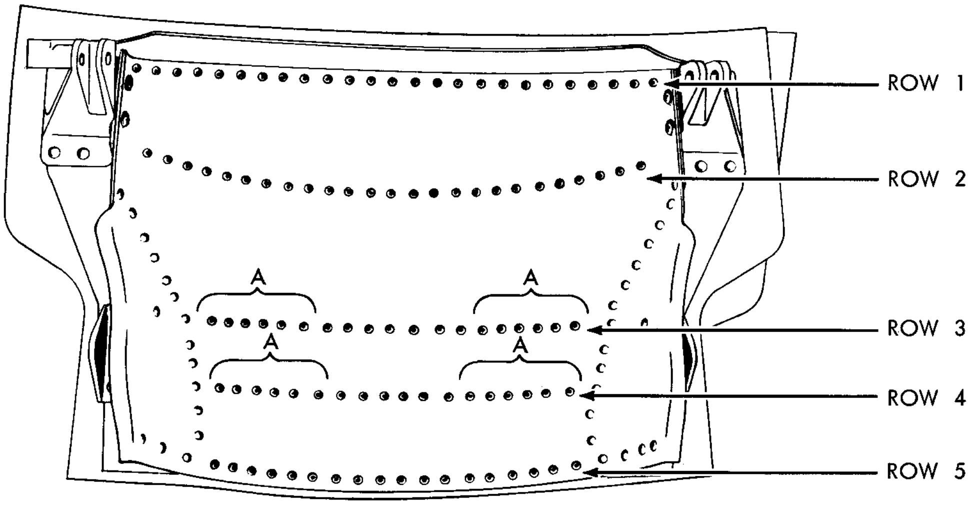


Figure 78-12. Guide Spring Adjustment



ROW: 1, 2, 3, 4	NO MORE THAN TWO LOOSE RIVETS AND NO TWO TOGETHER.
ROW: 5	NO MORE THAN FOUR LOOSE RIVETS AND NO MORE THAN TWO TOGETHER.

JS-2-2-7814

Figure 78-13. Door Inner Skin Blind Rivet Repair

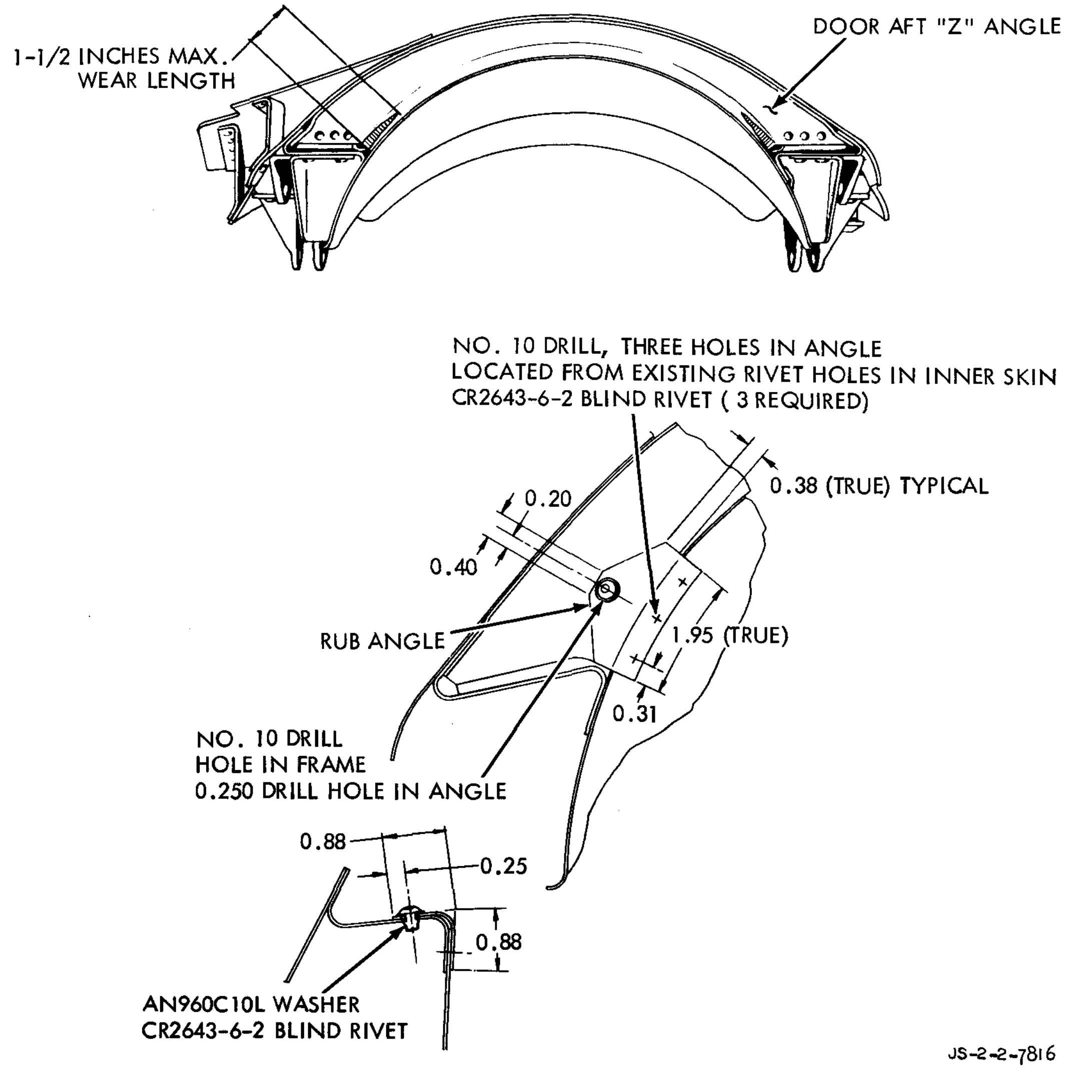

1-1/2 INCHES MAX. WEAR LENGTH
DOOR AFT "Z" ANGLE
NO. 10 DRILL, THREE HOLES IN ANGLE LOCATED FROM EXISTING RIVET HOLES IN INNER SKIN CR2643-6-2 BLIND RIVET (3 REQUIRED)
0.20
0.38 (TRUE) TYPICAL
0.40
RUB ANGLE
1.95 (TRUE)
0.31
NO. 10 DRILL HOLE IN FRAME 0.250 DRILL HOLE IN ANGLE
0.88
0.25
0.88
AN960C10L WASHER CR2643-6-2 BLIND RIVET
JS-2-2-7816

Figure 78-14. Door Aft "Z" Angle Repair

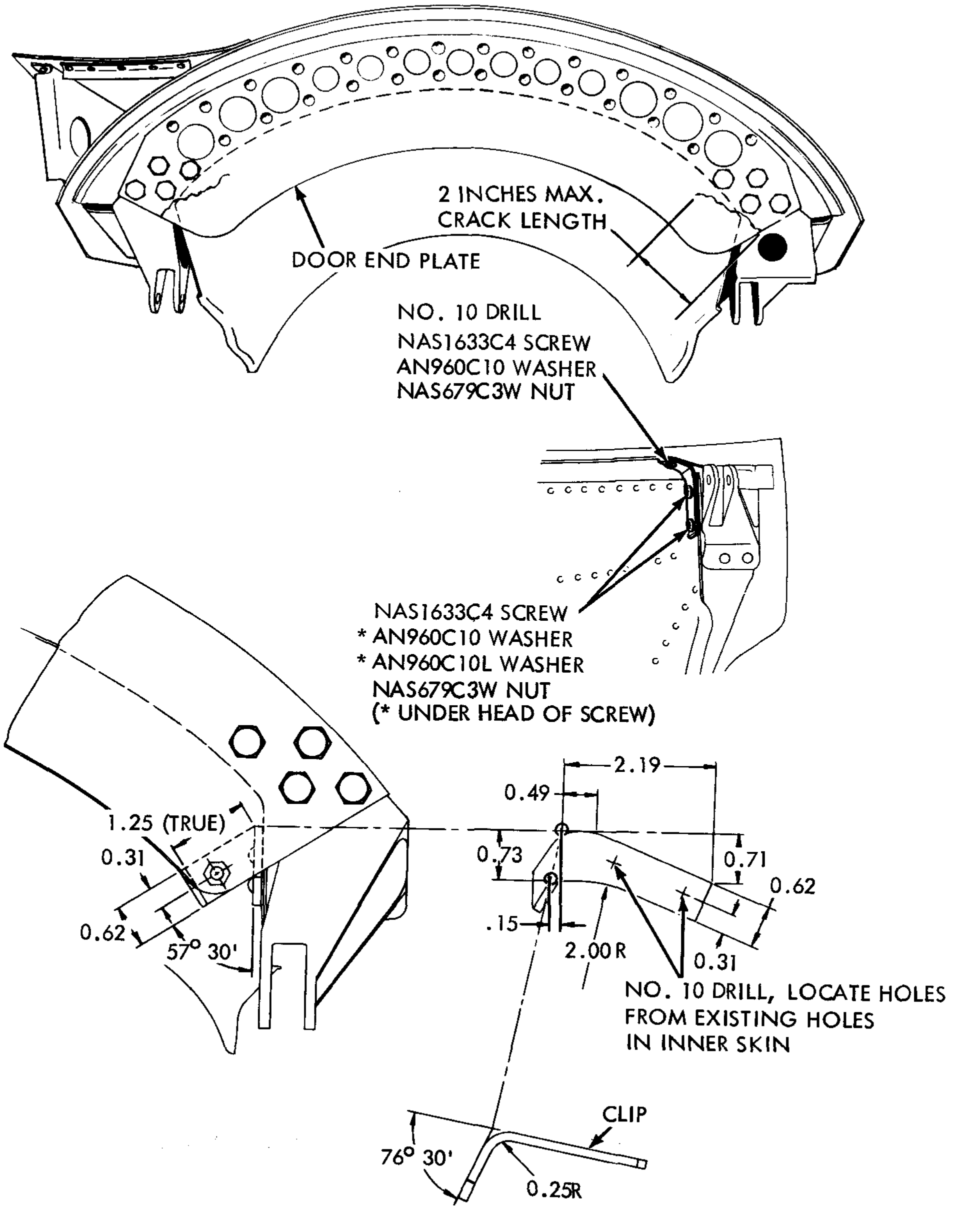


Figure 78-15. Door End Plate Crack Repair (Sheet 1 of 2)

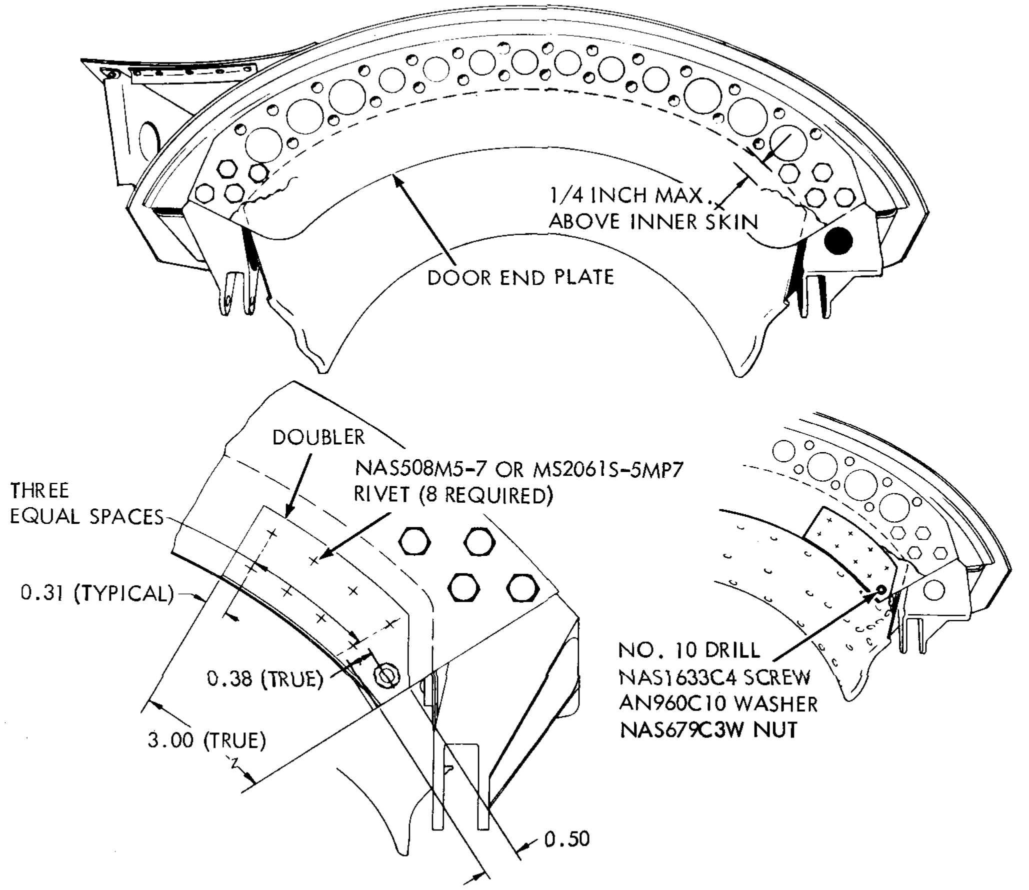


Figure 78-15. Door End Plate Crack Repair (Sheet 2 of 2)

OVERCENTER POSITION RIGGING (USING OPTIONAL TOOL PART NO. 11097-MEU-1)

STEP 1

DISCONNECT GUIDE LINKS FROM GUIDE AND DRIVEN LINKS FROM DOORS. REMOVE THE BOLT BETWEEN THE ACTUATOR AND THE GUIDE. (SEE FIG. 78-6).

STEP 2

REMOVE THE PLUNGER ASSEMBLY OR THE SCUFF PLATE FROM THE DRIVE LINK SUPPORT HINGE FITTING. REMOVE THE GUIDE SPRING FROM THE GUIDE IF INSTALLED. (SEE FIGS. 78-6 AND 78-12).

STEP 3

ATTACH GUIDE BAR TO LOWER GUIDE LINK HOLE IN GUIDE ASSEMBLY (WITH BEARING IN PLACE), AS SHOWN BELOW, USING WASHERS UNDER NUT AS REQUIRED. VISUALLY ALIGN GUIDE BAR SO THAT IT IS PARALLEL WITH TRACK.

STEP 4

ATTACH RIGGING TOOL BY INSERTING THE EXPANDABLE BUSHINGS INTO THE DRIVEN LINK PIVOT SHAFT LIGHTENING HOLES. TIGHTEN WING NUTS TO HOLD RIGGING TOOL IN POSITION. POSITION GUIDE ASSEMBLY SO THAT POSITIONING PIN IS ENGAGED IN SLOTTED HOLE IN GUIDE BAR.

STEP 5

CONNECT HAND PUMP TO ACTUATOR FORWARD THRUST HYDRAULIC PORT (SEE FIGURE 78-7).

STEP 6

WITH GUIDE POSITIONED AS REQUIRED BELOW, ADJUST ACTUATOR ROD END SO THAT WITH 200 PSI APPLIED TO EXTEND ACTUATOR (TO INSURE PISTON BOTTOMED OUT) THE ROD END IS ALIGNED WITH THE GUIDE ASSEMBLY AND THAT BOLT BETWEEN THE ACTUATOR AND THE GUIDE MAY BE FREELY INSERTED.

CAUTION

ROD END LOCK MAY BE DAMAGED IF ROD END IS USED TO ROTATE SHAFT AND LOCK IS NOT FULLY ENGAGED IN SLOT OF ACTUATOR SHAFT.

STEP 7

ENGAGE ROD END LOCK (9), FIGURE 78-7, TIGHTEN JAM NUT AND LOCK-WIRE.

STEP 8

REMOVE GUIDE BAR, RIGGING TOOL, AND HAND PUMP.

NOTE

INSURE ROD END THREADS COVER WITNESS HOLE IN ACTUATOR SHAFT.

STEP 9

ATTACH DRIVE LINKS TO DOORS, RECONNECT GUIDE LINKS TO GUIDE AND RECONNECT HYDRAULIC HOSES.

STEP 10

AT THIS POINT, CONTINUE WITH DOOR PRELOAD ADJUSTMENT, STEP 3, FIG 78-12, IF REQUIRED. IF THIS ADJUSTMENT IS NOT REQUIRED, RESTORE REVERSER TO NORMAL IN ACCORDANCE WITH STEPS 7, 8 AND 9 OF FIG 78-11.

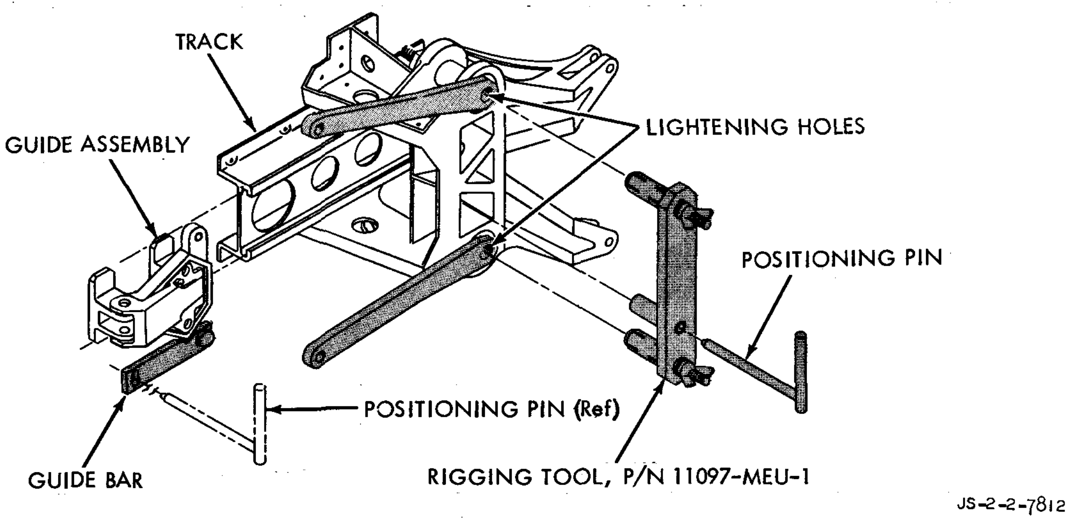
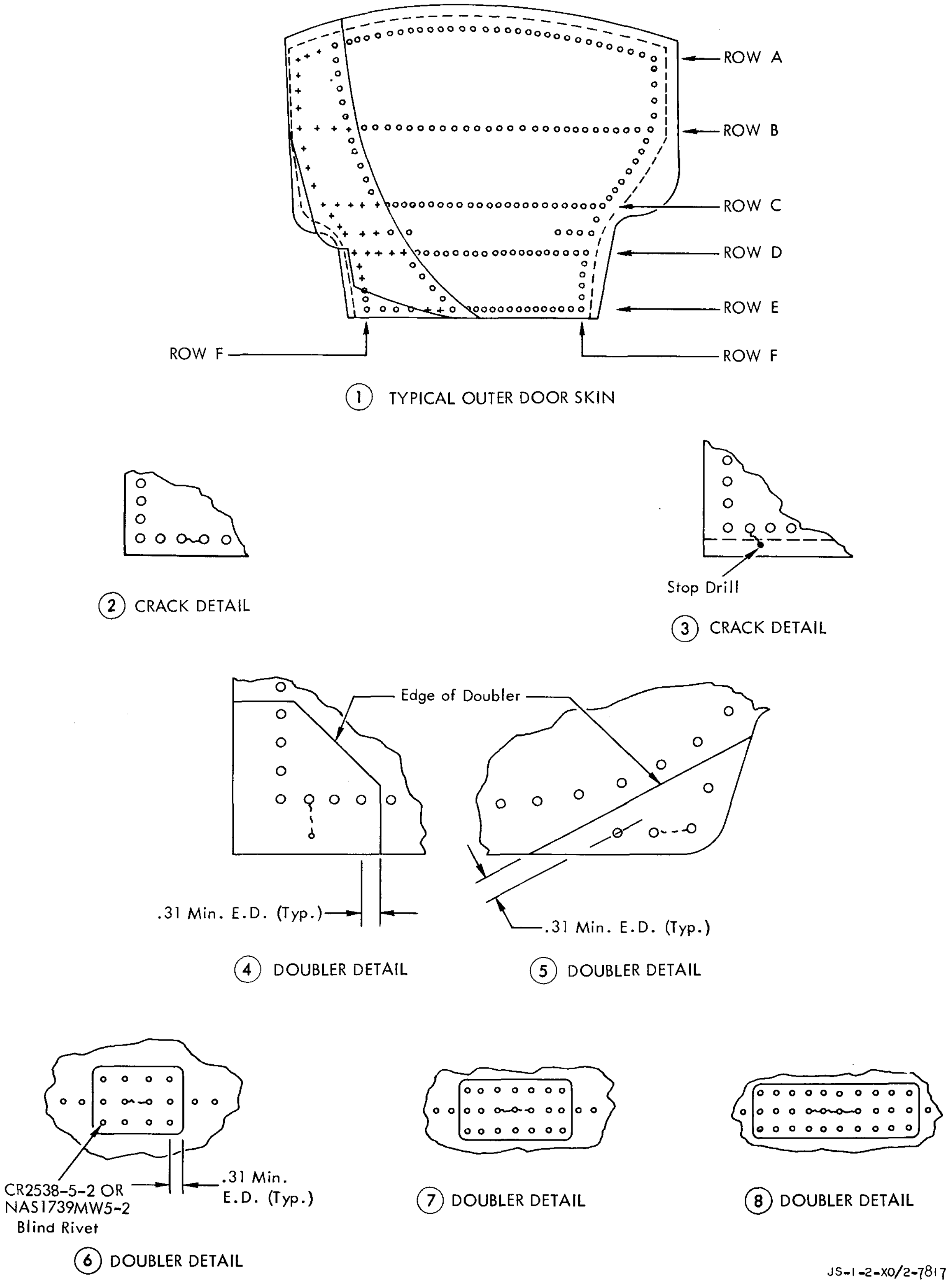


Figure 78-16. Overcenter Position Rigging—Optional Tool



① TYPICAL OUTER DOOR SKIN

② CRACK DETAIL

③ CRACK DETAIL

④ DOUBLER DETAIL

⑤ DOUBLER DETAIL

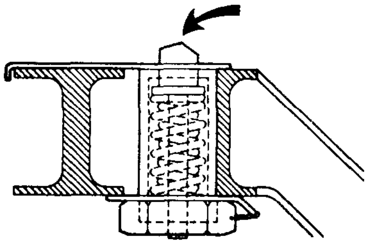
⑥ DOUBLER DETAIL

⑦ DOUBLER DETAIL

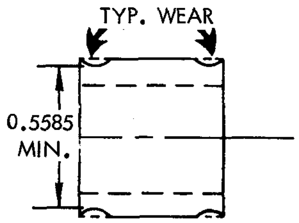
⑧ DOUBLER DETAIL

Figure 78-17. Door Outer Skin Crack Repair

REPLACE PLUNGER (FIG. 78-6, ITEM 14) IF WEARING SURFACE IS SCORE.



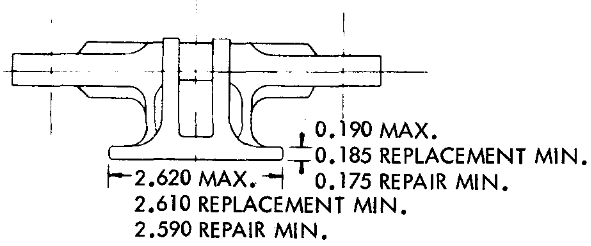
REPLACE CYLINDRICAL BEARINGS IF THE DIAMETER AT EITHER END MEASURES LESS THAN THE MINIMUM DIMENSIONS AS SHOWN.



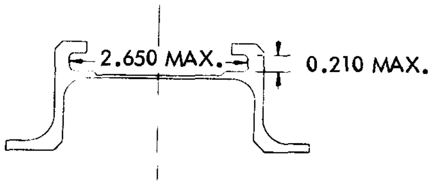
REPLACE OR REPAIR GUIDE (FIG. 78-6, ITEM 5) IF WEAR EXCEEDS "REPLACEMENT" MINIMUM DIMENSIONS AS SHOWN.

PARTS WORN TO DIMENSIONS BETWEEN "REPAIR MIN." AND "REPLACEMENT MIN" CAN BE REPAIRED BY BUILDING UP WEAR AREA WITH HARD CHROME PLATE AND GRINDING TO A DIMENSION BETWEEN "MAX." AND "REPLACEMENT MIN."

IF WEAR EXCEEDS "REPAIR MINIMUM" PART SHOULD BE SCRAPPED.



REPLACE TRACK (FIG. 78-6, ITEM 19) IF WEAR EXCEEDS DIMENSIONS AS SHOWN.



WEAR LIMITS

JS-1-2-X0/2-7818

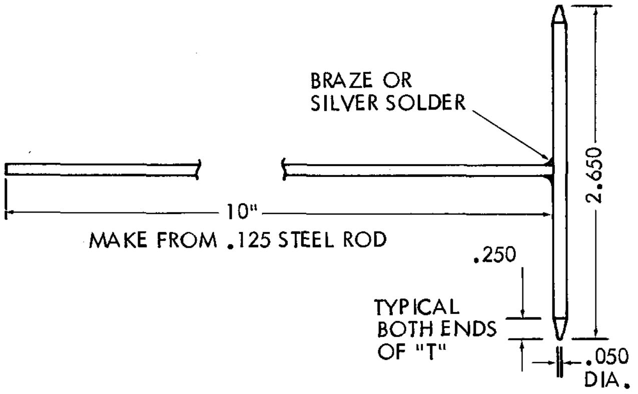
Figure 78-18. Thrust-Reverser Wear Limits

NOTE

INSPECTION OF TRACK WEAR IS TO BE ACCOMPLISHED AS FOLLOWS:

STEP 1

MAKE "T" GAGE AS SHOWN IN THE ILLUSTRATION BELOW.

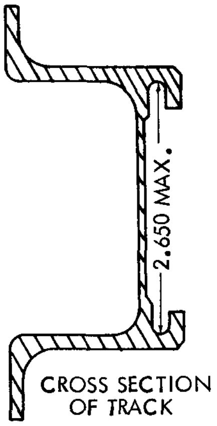


STEP 2

PLACE REVERSER DOORS IN REVERSE POSITION SO THAT THE GUIDE ASSEMBLY IS AT FORWARD PART OF TRACK.

STEP 3

INSERT THE GAGE AT AN ANGLE INTO THE TRACK JUST AFT OF THE GUIDE ASSEMBLY SO THAT THE TIPS OF THE "T" FIT INTO THE TRACK GROOVES WHEN THE GAGE HANDLE IS PARALLEL WITH THE TRACK. IF THE GAGE CAN BE ROTATED IN THE TRACK GROOVE SO THAT THE "T" SPANS ITS MAXIMUM WIDTH, WEAR IS EXCESSIVE AND THE TRACK SHOULD BE REPLACED.



STEP 4

CHECK THE ENTIRE LENGTH OF THE TRACK AFT OF THE GUIDE ASSEMBLY IN CASE THERE ARE HIGH OR LOW SPOTS IN THE GROOVES.

JS-1-2-X0/2-7819

Figure 78-19. Thrust-Reverser Track Wear Inspection

CHAPTER 80
STARTING

Contents Page

DESCRIPTION AND OPERATION 80-2
General 80-2
Starter-Generator 80-2
Engine Start Panel 80-2
Starter Relays 80-2
Start Lock-in Relay 80-2
Overvoltage Lockout Disconnect Relay 80-2A
Overvoltage Lockout Relays (Aircraft 5003, 5005, 5007 and up) 80-2A
Overvoltage Lockout Relay (Aircraft 5001, 5002, 5004 and 5006) 80-3
Voltage Regulator Lockout Relay 80-3
Starting Control Relays 80-3
Start-Lockout Relay 80-3
Battery Start 80-3
Battery Start Duty Cycle 80-3
Battery Lockout Switch 80-4
Battery Series Control Relay 80-4
Battery Series Relay 80-4
Ignition Bus 80-4
Battery-Start Slave Relay 80-4
Generator-On Sensing Relay 80-4
Ignition Bus Transfer Relay 80-4
Ignition Bus Overvoltage Relay 80-4
Ignition Bus Control Relay 80-4A
Ignition-Engaged Rectifiers 80-4A
External Power Start 80-5
Ground Starting Ignition 80-5
Stall Warning Test Relay 80-5
Ignition Disconnect Relays 80-5
Ignition Exciters 80-5
Igniter Plugs 80-5

OPERATIONAL CHECKOUT 80-5

TROUBLESHOOTING 80-6
Troubleshooting Start System 80-6
Parallel Start System 80-8A

SERVICING START SYSTEM 80-8B

MAINTENANCE START SYSTEM 80-8B

FIGURES
80-1. Start System Schematic Diagram (Aircraft 5001, 5002, 5004 and 5006) 80-9
80-2. Start System Schematic Diagram (Aircraft 5003, 5005, 5007 Thru 5055) 80-10
80-3. Engine Starter Generator 80-11
80-4. Ignition Circuit Schematic Diagram (Aircraft 5001, 5002, 5004 and 5006) 80-12
80-5. Ignition Circuit Schematic Diagram (Aircraft 5003, 5005, 5007 and up) 80-13
80-6. Ignition Transfer Circuit Schematic Diagram 80-14
80-7. Start System Schematic Diagram (Aircraft 5056 and up) 80-15
80-8. Wiring Schematic of Parallel Start Relays 80-16
80-9. Start System Schematic Diagram (Aircraft 5066 and up) 80-17
80-10. Simplified Schematic — Parallel Start Relays 80-19
80-11. Start System Schematic Diagram (Aircraft 5001 thru 5055 Modified by Service Bulletin 329-294) 80-21
80-12. Start System Schematic Diagram (Aircraft 5056 thru 5065 Modified by Service Bulletin 329-294) 80-23
80-13. Start System Schematic Diagram (Aircraft 5066 and Up Modified by Service Bulletin 329-294) 80-25

80.1 DESCRIPTION AND OPERATION.

80.1.1 GENERAL. (See figures 80-1 and 80-2.) The JetStar starter system is completely electrical, with each engine equipped with a combination starter-generator. Electrical power is supplied to a starter bus by either an external, 28-volt, controlled DC voltage source or by the two airplane batteries connected in series. Starting power, which can be applied to only one starter-generator at a time, is connected to terminal B of the selected starter-generator through a corresponding starter relay. Engine starting is controlled from an engine start panel. The starting control circuits are supplied essential DC bus power through a START circuit breaker on the pilot's side circuit breaker panel.

Note

> Aircraft 5001, 5002, 5004 and 5006 are equipped with an automatic start system which is initiated by momentarily depressing the start switch to the "START" position. This system is designed to automatically stop the start cycle when the start is completed. On aircraft 5003, 5005 and 5007 and up, it is necessary to hold the start switch in the "START" position during the start cycle.

80.1.2 STARTER-GENERATORS. Each starter-generator functions as a direct cranking starter motor when supplied electrical power from the starter bus. Driven by a starter-generator, the corresponding engine accelerates to 23 to 30 percent tachometer reading, at which point the air flow and fuel flow are sufficient for self-sustaining engine operation.

CAUTION

> The starter-generator duty cycle as a starter is 30 seconds on, 2 minutes off, 30 seconds on, and then 15 minutes off. DO NOT EXCEED THE DUTY CYCLE.

80.1.3 ENGINE START PANEL. (See figure 80-3.) The engine start panel, at the right front of the overhead panel console, contains the following switches for the operation of the engines: an ENG SEL switch, a POWER SEL switch, an AIR START switch, a START switch, and a FUEL AIR BLEED LINE switch. The ENG SEL switch has five positions ("1", "2", "3", "4", "OFF") and is positioned for the engine to be started. The POWER SEL switch, with three positions ("GPU START," "BAT START," "OFF"), selects the electrical circuit applicable to the power source being used. The three-position ("NORMAL", "AIR START", "OFF") AIR START switch, when placed in the "NORMAL" position, arms the ground starting circuit, including the throttle ignition switch circuit. When held in the "AIR START" position, the AIR START switch enables the igniters to be energized independently of the throttle ignition switch.

A two-position ("START-OFF") start switch is used on JetStar airplanes 5003, 5005, 5007 and up. It is necessary to hold the start switch in the "START" position for the duration of the start cycle. When the start switch is released, during the start, it returns to the "OFF" position, interrupting the start cycle. On aircraft 5001, 5002, 5004, and 5006 with automatic start system, it is necessary to actuate the start switch to the "STOP" position to interrupt the start cycle before the start cycle is completed. On all airplanes, when the start switch is depressed to "START" position, it initiates a circuit to the starter-generator of the engine selected by the ENG SEL switch, causing the engine to rotate. On 5066 and up, the start cycle will not be initiated if the ESS-MN BUS TIE switch is in the closed position. (Refer to paragraph 24.0.1.15, figure 24-9B and figure 80-9.)

80.1.4 STARTER RELAYS. When one of the four starter relays is energized, its corresponding starter-generator is connected to the starter bus. This occurs when a given engine is selected with the ENG SEL switch and the START switch is depressed to the "START" position. Selecting a given engine routes power to energize the starting control circuit in the corresponding generator control panel, and completes a circuit to the starter relay coil. Whenever any one of the starter relays is energized, a ground is applied to the circuit which causes a STARTER ENGAGED module on the annunciator panel to illuminate (see Chapter 33). On JetStar aircraft 5001, 5002, 5004, and 5006 the starter relays incorporate a holding circuit which is actuated when a current of 490 amperes or more flows through the main contacts between the M and B terminals of the starter relay. This is sufficient to close the auxiliary contracts between terminals C and S of the starter relay. This allows a continuous power circuit to the START LOCK-IN relay which will hold in until the current flow decreases to 250 (±15) amperes, at which time the engine should have accelerated to 23 to 30 percent tachometer reading and be self sustaining.

80.1.5 START LOCK-IN RELAY. When the START switch is depressed to the "START" position, a start lock-in relay is energized, thereby initiating the start control circuits.

Initially, the start lock-in relay is energized with a circuit through the cutoff position of an appropriate throttle switch in the throttle quadrant. Once energized, however, the start lock-in relay provides its own holding circuit through the "NORMAL" position of the AIR START switch. This holding circuit is necessary because the initial energizing circuit is subsequently opened when the throttle is advanced out of the cutoff position. The start lock-in relay remains energized for the duration of the start cycle.

80.1.6 OVERVOLTAGE LOCKOUT DISCONNECT RELAY. (Aircraft 5003, 5005, 5007 and up). When the start switch is depressed, the overvoltage lockout disconnect relay is energized. This removes one of the two grounds from the overvoltage relay coil in each of the generator control panels. Further, a ground is provided for each of the four overvoltage lockout relays, even though only one is energized at a time (see chapter 24).

80.1.7 OVERVOLTAGE LOCKOUT RELAYS. (Aircraft 5003, 5005, 5007 and up). There are four overvoltage lockout relays, one for each starter-generator. During ground starting, the overvoltage lockout relay for the selected starter-generator is energized through the starting control circuits of the corresponding generator control panel (see chapter 24). When an overvoltage lockout relay is energized, the second ground is removed from the overvoltage coil in the related generator control panel, thereby suspending overvoltage protection for the selected starter-generator during the start cycle. A bypass of the voltage

80.1.7 (Cont).

regulator for the selected starter-generator is also furnished; this removes undesirable control and resistance from the starting shunt field. This, in turn, allows maximum shunt field current during start, which increases the torque output of the starter.

80.1.8 OVERVOLTAGE LOCKOUT RELAYS (Aircraft 5001, 5002, 5004, and 5006). Four overvoltage lockout relays are on the aft compartment electrical equipment rack. When the engine START switch is momentarily depressed, the overvoltage lockout relay corresponding to the selected engine is energized. This removes the ground from the overvoltage relay coil in the related generator control panel, thereby suspending overvoltage protection during engine starting. Otherwise, the increased starting voltage would trip the field trip relay.

80.1.8.1 On aircraft modified by Service Bulletin 329-294, the overvoltage lockout relay in conjunction with the voltage regulator lockout relay allows maximum shunt field current during engine start and bypasses the generator control panel thereby eliminating the need for overvoltage protection. (See figures 80-11, 80-12, and 80-13.)

80.1.9 VOLTAGE REGULATOR LOCKOUT RELAY. JetStar aircraft 5001, 5002, 5004, and 5006 only. Four voltage regulator lockout relays are on the aft compartment electrical equipment rack. When the start circuit is energized, the voltage regulator lockout relay for the selected engine is energized and its voltage regulator is bypassed, removing voltage control during the start cycle. This allows maximum shunt field current during start, which increases the torque output of the starter.

To insure continual satisfactory battery starts it is necessary that the batteries be maintained in accordance with the instructions in Chapter 24.

80.1.10 STARTING CONTROL RELAYS. As each engine is started, the corresponding starting control relay, located inside the generator control panel, is energized. This disconnects the generator circuit from both the main DC bus an the equalizer bus. If the starting control relay is energized with the field relay closed, both the corresponding starter relay and overvoltage lockout relay are energized. Then starting current can pass through the related starter relay to the appropriate starter-generator.

80.1.11 START-LOCKOUT RELAY. During emergency generator operation of one of the starter-generators, its corresponding starter relay is used to route the generator output to the starter bus, through the emergency reverse current cutout relay, and finally to the essential DC bus (see chapter 24). With the output of this generator on the starter bus, it is then necessary to ensure that a ground start is not attempted. Thus, when any generator control switch is in the "EMER" position, the start-lockout relay is energized. This disconnects power supplied through the START circuit breaker from the ground engine starting control circuits, thereby preventing any of the other starter relays from energizing and preventing the batteries in series from being connected to the starter bus. Also, when the start-lockout relay is energized, the ground circuit for the STARTER ENGAGED module on the annunciator panel is opened. This prevents a STARTER ENGAGED indication when a starter relay is energized for emergency generator operation.

80.1.12 BATTERY START. To effect a battery start, the POWER SEL switch must be in the "BAT START" position, and the external power receptacle compartment door must be closed. Once the start-lock-in relay is energized, a battery series control relay is energized in turn. This applies voltage to the coil of the battery series relay, which actuates and connects the two batteries in series, providing 48 VDC to the starter bus. Battery power is then applied to a starter-generator when one of the starter relays is energized.

On airplanes 5001 - 5055 all engines are started with the batteries in series combination. Aircraft 5056 and up (see figure 80-7) are equipped with automatic series-parallel start system which deactivates the battery series circuit after the first two engines are started. The batteries in parallel combination with the generators supply starting current to the starters for the remaining two engines to be started.

80.1.13 BATTERY START DUTY CYCLE. To prolong battery life and to prevent hazardous overheat conditions, limit the battery starting duty cycle as follows:

PHASE I Limit the starting cycle to six attempts, successful or unsuccessful. If additional starts are necessary, requiring phase II, charge the battery at 28 volts for 20 to 30 minutes.

PHASE II Limit the starting cycle to four attempts, successful or unsuccessful. If additional starts are necessary, requiring Phase III, charge the battery at 28 volts for 20 to 30 minutes.

PHASE III Limit the starting cycle to two attempts, successful or unsuccessful followed by a 4-hour cooling period, and then by charging at 28 volts for 20 to 30 minutes. Repeat Phases I, II and III.

Note

If the battery starting duty cycle and the starter duty cycle conflict, the more stringent requirement applies. If, at any time during the starting duty cycle, the battery fails to turn

80.1.13 (Cont).

the engine to starting speed, it may be due to a reduced state of charge or to low battery temperature. Apply 28 volts for 30 minutes to recharge the battery. Apply external heat to warm the battery, and then recharge for 30 minutes.

80.1.14 BATTERY LOCKOUT SWITCH. The battery lockout switch is positioned by the door of the external power receptacle compartment. When the door is closed, the ground starting control circuits function for battery starting only; when open, only external power ground starts can be made.

This prevents applying 48 volts (battery start) and 28 volts (ground power start) to the start bus simultaneously.

80.1.15 BATTERY SERIES CONTROL RELAY. When the battery series control relay is energized, the battery series relay is, in turn, energized with essential DC bus power delivered through a BAT SERIES PWR circuit breaker. However, before the battery series control relay is energized, it provides a path for power to the start lock-in relay coil. This path is opened when the battery series control relay is energized to prevent connecting the start bus to the essential bus during the first battery start. Once the start lock-in relay is energized, it routes power through the POWER SEL and battery lockout switches to allow the battery series control relay to remain energized. On aircraft 5046 and up, the battery series control relay, when energized, provides a power circuit from the start bus to the ignition bus control relay and the ignition transfer relay. On airplanes 5056 and up the battery series circuitry is de-activated by the series-parallel relays after the first two engines are started.

80.1.16 BATTERY SERIES RELAY. When energized, the battery series relay connects the two batteries in series providing 48 VDC, thus allowing for the large voltage drop which occurs in delivering the high current needed for starting. This battery power is simultaneously connected to the starter bus when the battery series relay is energized. Also, during a battery start, power to energize the starting control relay in the appropriate generator control panel passes through the auxiliary contacts of the battery series relay.

80.1.16.1 Series-Parallel Relays (Airplanes 5056 and up) (See figures 80-7 and 80-10.) The series-parallel relays function as sensors to by-pass the battery series circuitry after the first two engine starts which allows the batteries to assume their normal parallel configuration for the remaining two engine starts. This is accomplished by taking a signal from the "IND" terminal of the generator contactor which assures the generator is on the line. After two generators are on the line the series-parallel relays function so that the power circuit to the battery series control relay coil is deactivated and a signal is applied to the "APP" terminal of the emergency reverse current relay for the third and fourth engine starts. At this time the batteries will assume their normal parallel configuration and the "Emergency Reverse Current Relay" connects the essential and start bus together. This allows the batteries in parallel combination with the generators to supply the starting current to the starter-generators.

80.1.17 IGNITION BUS (see figures 80-4 and 80-5.) During battery starting, the voltage to the essential DC bus drops so low as to be ineffective. Consequently, certain important DC loads are supplied by an ignition bus. Its requirements are furnished during battery starting from the starter bus, until the starter bus voltage increases to 29 (±1) volts. At all other times, the essential DC bus supplies the ignition bus loads. A network of relays effects the bus switching.

Ignition transfer circuit operation can be detected by observing the dome light located on the overhead interphone panel. If the transfer circuit is functioning correctly the dome light will dim twice during the first engine battery start.

80.1.18 BATTERY-START SLAVE RELAY. On aircraft 5001 through 5045 only, when the battery series control relay is energized, a battery - start slave relay is energized in turn. This provides a power circuit from the starter bus to the ignition bus control relay and the ignition transfer relay.

80.1.19 GENERATOR-ON SENSING RELAY. If a starter-generator is connected to the DC distribution buses, a generator-on sensing relay is grounded through the affected generator-out relay (see Chapter 24). If the battery starting control circuits are energized at the same time, the generator-on sensing relay is actuated. This prevents the starter bus from being connected to the ignition bus, for with a starter-generator on the line during battery starting, the voltage on both the starter and the essential DC buses is considerably higher. Thus, it is neither necessary nor desirable to switch the ignition bus requirements to the starter bus.

80.1.20 IGNITION BUS TRANSFER RELAY. (See figure 80-6). An ignition bus transfer relay, when de-energized, connects the ignition and the essential DC buses. During battery starting, without a generator on the bus, the ignition bus transfer relay is energized by starter bus power, if the ignition bus control circuit is complete. One of the starter relays must be energized to furnish a ground to the relay coil. When the ignition bus transfer relay is energized, the ignition and starter buses are connected.

80.1.21 IGNITION BUS OVERVOLTAGE RELAY. The ignition bus overvoltage relay is designed to operate at 29 volts. The purpose of the ignition bus overvolt-

80.1.21 (Cont).

age relay is to open the circuit to the ignition transfer relay when the starter bus voltage reaches a value of 29 (±1) volts. When the circuit to the ignition bus transfer relay is de-energized, the ignition bus loads are assumed by the essential DC bus. This ensures that excessive voltage is not applied to the ignition bus loads.

80.1.22 IGNITION BUS CONTROL RELAY. The circuit used to energize the start lock-in relay initially passes through a de-energized ignition bus control relay. Subsequently, during a battery start without a generator on the bus, the ignition bus circuit is energized with starter bus voltage. The ignition bus control relay is then energized, thereby applying starter bus voltage to the coil of the ignition bus overvoltage relay. Also, this connects starter bus voltage to the ignition control circuit. This sequences the application of voltage to the ignition bus overvoltage relay to prevent prematurely de-energizing the ignition bus transfer relay.

80.1.23 IGNITION-ENGAGED RECTIFIERS. Two ignition-engaged rectifiers in the ignition bus circuit operate during battery start to prevent undesirable

80.1.23 (Cont).

reverse current flow. This might otherwise occur, since the STARTER ENGAGED module and the ignition bus overvoltage and ignition bus control relays share a common ground through one of the starter relays. Thus the No. 1 ignition-engaged rectifier prevents voltage on the starter bus from causing a current flow into the annunciator panel. Conversely, the No. 2 ignition-engaged rectifier prevents current flow from the STARTER ENGAGED module into the ignition bus control circuit.

80.1.24 EXTERNAL POWER START. To effect an external power start, the POWER SEL switch must be in the "GPU START" position, and the external power receptacle compartment door must be open. When the START switch is actuated to the "START" position, the start lock-in relay energizes, routing control current through the POWER SEL, battery lockout, and ENG SEL switches to the selected generator control panel. The corresponding starting control relay is then energized, permitting the appropriate starter relay and overvoltage lockout relay to be energized. Thus, starting power is applied to terminal B of the selected starter-generator.

CAUTION

For external power cart starting, use only constant-potential sources with a maximum of 1,000 amperes or constant-current sources with current limiting set not higher than 1,200 amperes. Higher settings will damage the starter-generator. Carts which require an output signal from the aircraft are not acceptable, since the aircraft is not designed to supply this signal. Cart voltage should be set as close to 28 volts as possible. Prolonged cart voltages higher than 29 volts can cause excessive battery gassing and damage. It is permissible to use an oval plug from the external power cart in the rectangular receptacle, provided the ground power cart has a rating of 1,000 amperes.

80.1.25 GROUND STARTING IGNITION. The ignition control circuit is energized during ground starting when the appropriate engine throttle switch is advanced to the ignition position. On aircraft 5003, 5005, 5007 and up, this routes control current through two "B" stall warning system test switches, the AIR START switch, and the ENG SEL switch to one of four AIR IGNITION CONT circuit breakers. On aircraft 5001, 5002, 5004, and 5006, this routes control current through the ignition blocking rectifiers. This control current then passes through a de-energized stall warning test relay to the coil of an ignition relay for the selected starter-generator. In the case of the No. 2 and 3 ignition control circuits, a path through the de-energized contacts of one of the two ignition-disconnect relays is included. With the appropriate ignition relay energized, power is supplied to two igniters corresponding to the selected starter-generator through one of four ENG IGNITER PWR circuit breakers. The igniters remain energized until the START switch is released.

80.1.26 STALL WARNING TEST RELAY. When either of the "B" stall warnings system test switches is held in the "TEST" position, the stall warning test relay is energized. This prevents power from being applied to any of the igniters, for either a ground or an air start. Also, the ground start ignition control circuit through the test switches is opened when either is in the "TEST" position. (See Chapter 27.)

80.1.27 IGNITION DISCONNECT RELAYS. If either the No. 2 or No. 3 fire emergency handle is pulled, the corresponding ignition-disconnect relay is energized. This prevents power from being applied to the igniters of the affected engine.

80.1.28 IGNITION EXCITERS. Two identical four-joule ignition exciters are installed on each engine, one for each igniter plug.

CAUTION

Engine ignition system duty cycle is two minutes on, three minutes off, 2 minutes on, 23 minutes off, which is pertinent to both exciters. DO NOT EXCEED THE DUTY CYCLE.

80.1.29 IGNITER PLUGS. Igniter plugs are installed in the No. 3 and No. 6 combustion chambers in each engine.

CAUTION

Turn off power to the ignition circuits before working around the ignition exciters. The exciters generate 30,000 volts when actuated by the air start switch.

80.2 OPERATIONAL CHECKOUT.

80.2.1 ENGINE STARTER SYSTEM. If an engine starts satisfactorily, it can be assumed that the engine starter system is operating properly.

80.3 TROUBLESHOOTING.

80.3.1 TROUBLESHOOTING START SYSTEM.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE WILL NOT ACCELERATE TO STARTING RPM ON EXTERNAL POWER.		
Power output of auxiliary power unit insufficient.	Make battery start in order to check for high voltage on APU and defective overvoltage lockout relay.	Rectify or substitute APU and/or replace overvoltage lockout relay.
Generator winding defective.	Attempt to start the other three engines.	If all other engines start normally, replace the starter-generator.
No power to starter shunt field.	Check circuit.	Repair circuitry as necessary.
Mechanical failures.		See Chapters 72 and 73.
Overvoltage lockout relay.	To check the overvoltage lockout relay and insure it is operating correctly, a voltmeter can be placed across terminals "A" and "B" of the starter - generator to monitor voltage during start. When the start switch is depressed, the voltmeter should read approximately 1 to 3 volts. (Voltage drop across shorting relay contacts.) When the start switch is released the voltage should rapidly increase to approximately 12 to 18 volts (voltage drop across carbon pile voltage regulator).	Replace defective overvoltage lockout relay.
ENGINE WILL NOT ACCELERATE TO STARTING RPM — BATTERY POWER		
Batteries not sufficiently charged.	Make GPU start.	See battery maintenance Chapter 24.
Overvoltage lockout relay.	To check the overvoltage lockout relay and insure it is operating correctly, a voltmeter can be placed across terminals "A" and "B" of the starter generator to monitor voltage during start. When the start switch is depressed, the voltmeter should read approximately 1 to 3 volts. (Voltage drop across shorting relay contacts.) When the start switch is released the voltage should rapidly increase to approximately 12 to 18 volts (voltage drop across carbon pile voltage regulator).	Replace defective overvoltage lockout relay.

80.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE WILL NOT ACCELERATE TO STARTING RPM - BATTERY POWER (Cont).		
Starter-generator windings defective.	Attempt to start the other three engines.	If the other three engines start normally, replace the starter-generator.
No power to starter shunt-field.	Check circuit.	Repair circuitry as necessary.
Mechanical failures.	See Chapters 72 and 73.	See Chapters 72 and 73.
ENGINE WILL NOT CRANK-EXTERNAL POWER OR BATTERY - OTHER ENGINES NORMAL		
Starter drive shaft failed.	Starter can be heard running when start switch is depressed. Remove starter and check for either sheared drive shaft in starter or sheared drive shaft in accessory gearbox.	Replace starter or engine, depending on which drive shaft is found to be sheared.
No power on starter relay coil.	Throttle not in cutoff position. Check start circuitry from start switch to start relay.	Rectify circuitry as necessary.
Starter-generator winding open.	Check for continuity between starter-generator terminals "D" and "E" (Remove wires from terminal "D" for check.)	Replace defective starter-generator.
Mechanical failure.	Rotate compressor by hand.	See Chapter 72.
Generator control panel field relay tripped.	Check to insure fire emergency handle in.	Reset.
No power on essential bus	If first start: (a) Check APU output (b) Check battery switch should be "ON."	(a) Rectify APU generator system. (b) Place battery switch "ON."
Start lockout relay energized.	Check emergency generator selector switch position "NORMAL."	Place switch in "NORMAL" position.
Start SEL. SW. Defective PWR SEL SW Defective Start Lock-in Relay Defective. BAT. SER. Relay Defective GEN CONT PNC Defective Start Relay Defective	Make continuity check of control circuitry.	Replace any defective items.
No start bus voltage during battery start.	(a) Check for defective BAT lockout SW	Replace any defective items.

80.3.1 (Cont).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE WILL NOT CRANK - EXTERNAL POWER OR BATTERY - OTHER ENGINES NORMAL (Cont).		
No start bus voltage during battery start. (Cont)	(b) Check for defective BAT series cont relay	Replace any defective items.
	(c) Check for defective BATTERY series relay.	
ENGINE FAILS TO LIGHT-OFF - ALL OTHER ENGINES NORMAL.		
No ignition (EXT or BATT power)	Purge engine and depress air start-normal switch to AIR START. If ignition is heard, check circuit between air start switch and throttle quadrant switch. If no ignition is heard check circuit between air-start switch and ignition relay.	Repair circuitry or replace defective items as necessary.
	Check voltage at ignition relay terminal when air start switch is depressed.	If 28 volts is present, replace exciter and/or sparkigniters.
Weak ignition on BATTERY starts, external power normal.	Check 3-amp fuse and 20-amp circuit breaker in the ignition transfer circuit.	Rectify cause of open circuit, and replace fuse or reset circuit breaker.
Ignition transfer circuit inoperative.	Dome light should dim at least twice.	Rectify transfer circuit as necessary.
ENGINE STOPS ROTATING WHEN THROTTLE IS ADVANCED.		
Defective control circuitry.	Check start lock-in relay.	Replace start lock-in relay.
Mechanical defects.	See Chapters 72 and 73.	See Chapters 72 and 73.
ENGINE START LIGHT FAILS TO EXTINGUISH.		
Defective start relay.	Check circuit.	Rectify circuit as necessary.
NO ENGINE STARTS - EXTERNAL OR BATTERY POWER.		
ESS-MN BUS TIE switch closed.	Check switch position.	Place to open position.
ENGINE ROTATES BUT NO IGNITION ON BATTERY START — GPU START NORMAL.		
Ignition bus transfer fuse.	Check fuse.	Replace fuse.

80.3.2 PARALLEL START SYSTEM. The purpose of the parallel start system is to prevent excessive wear on the batteries when making battery starts. The starter draws an initial surge current of approximately 1200 amps. This current is supplied by the batteries in series for the first two starts. When the generators are turned on, the parallel start system provides a means of allowing the operating generators to supply starting current for the third and fourth starts, assisted by the batteries in parallel or an APU generator.

Essentially, the parallel start system may be thought of as an electric counter and switch assembly, which counts the number of generators that are on, and switches from series start configuration to parallel start configuration as soon *as the second generator is turned on.* (*An APU generator* will assist the engine generators in making parallel starts, but cannot be used to activate the parallel start system.)

For the first two starts, the parallel start switch circuit is positioned as shown in figure 80-6, circuit 1. When the start switch is depressed, power is applied through the circuit to energize the battery series control relay, which then energizes the battery series relay to connect the batteries in series to the start bus. When any two generators are on, the parallel start switch circuit operates to open the series circuit and close the parallel circuit, as shown in figure 80-6, circuit 2. During remaining starts, power is supplied through the circuit to energize the emergency reverse current relay, which connects the essential DC bus to the start bus, so that starting power is supplied by the operating engine generators and the batteries in parallel (or, engine generators and APU generator).

The parallel start switch circuit shown in figure 80-6, circuits 1 and 2, actually consists of four relays whose contacts are wired so as to function as a single switch. Each relay is energized when the corresponding generator is turned on (No. 1 generator ON energizes the No. 1 parallel start relay, No. 2 generator ON energizes the No. 2 parallel start relay, etc.). Power to energize the relay comes from the IND terminal of the generator contactor, through the generator out relay fuse. Figure 80-10, circuit 3, *shows a simplified schematic of the system for engines* 3 and 4. In the circuit shown, No. 3 engine has been started and the generator turned on. The No. 3 generator contactor is energized, connecting the generator output to the main DC bus, and through the normal reverse current relay to the essential DC bus. Power from the IND terminal has the No. 3 parallel start relay energized. The "A" contacts have partially opened the series circuit, and the "B" contacts have partially completed the parallel circuit. When the No. 4 engine is running and the generator is turned on, the No. 4 generator contactor closes to connect the generator output to the bus. At this time, power from the IND terminal energizes the No. 4 parallel start relay. This action completes the switching from series to parallel configuration, so that the remaining starts are parallel.

The four parallel start relays are of the plug-in type, mounted on the copilot's side shelf at fuselage station 226. The parallel start system has proven to be a reliable one; however, as with any system, faults can occur. In this system, there are some faults that can occur which are readily apparent, some which may go undetected for days or even weeks, and others which may never be known. As an example of the latter type of failure, suppose you always start the engines in the "normal" sequence (3, 4, 2, 1). In this case, several faults can exist in the circuitry for the No. 1 and No. 2 parallel start relays, or the relays themselves can be failed, and you would never know it. Remember, it is the first two (in this case, 3 and 4), that establish the parallel start configuration for the last two. A fault in the No. 1 parallel start relay circuit would never be detected unless the No. 1 engine was one of the first two to be started.

Failure of any of the parallel start relays to energize will not prevent engine starting. If a relay fails to energize when the generator is turned on, no switching will occur at this time. Since the switching from series to parallel requires that two relays be energized, the system will remain in series start configuration until there are two relays energized. For example, suppose that you start engines 3 and 4, and the No. 4 parallel start relay fails to energize when you turn on the No. 4 generator. Since only one of the relays is energized (No. 3), the system remains in series start configuration, and the third start will be a battery series start. If the generator for that engine is turned on, and if there are no other faults, the fourth start will be parallel. This type of fault may go undetected for some time. If your batteries are maintained in good condition, the fact that you are making three series starts and only one parallel start is not readily apparent unless you are specifically watching for it. One way of detecting this condition is to monitor starting voltage. Prior to starting an engine, set the voltmeter selector switch to the generator position for that engine. When the start switch is depressed for a series start, the voltmeter needle will initially jump to 19 or 20 volts, climb steadily as RPM increases, and peak out at 30 volts or more by the time the start switch is released. On a parallel start, the initial indication will be 22 volts or more, depending on the number of generators that are on, and peak at 28 volts.

If you experience a starting problem on the last two starts, *the first step in locating the problem area is to determine* whether or not you have a parallel start problem, and if so, where within the parallel start system the fault has occurred. Suppose, for example, you have started engines 3 and 4, turned on the generators, and No. 2 does not rotate, nor does the STARTER ENGAGED light illuminate. In order to help isolate the trouble, try to start No. 1 engine. If either engine rotates and/or the STARTER ENGAGED light illuminates, the trouble is not in the parallel start system, but in the circuitry for the engine that does not rotate. On the other hand, if neither engine rotates, and the STARTER ENGAGED light does not illuminate for either one, you should suspect a parallel start problem (provided that two separate faults have not developed at the same time).

If the problem is in the parallel start circuit you can get the engines running by turning off either one of the engine generators. This will cause the parallel start relay assembly to revert to series start configuration, and the No. 2

80.3.2 (Cont).

engine can now be started with battery series power. (Remember to turn the battery switch off if you had previously turned it on.) When the engine is running, turn on the generator, and also turn on the generator that was previously turned off. If the No. 1 engine now starts, the fault is in the parallel start relay circuitry; either there is a loose or corroded pin connector for one of the relay contacts of the No. 3 or the No. 4 parallel start relay, or a broken wire. In either case, the problem must be with No. 3 or No. 4 since those are the relays which should have established the parallel start for the last two engines. In order to circumvent this problem until the fault can be found and fixed, the engines can be started in a different sequence, such as 2, 3, 4, 1 or 2, 4, 3, 1 or even 2, 1, 3, 4.

If No. 1 engine still does not start in parallel configuration with three engine generators on, the problem is probably a failed emergency reverse current relay or a blown parallel start relay fuse. Both of these components are located on the power shield. A failed emergency reverse current relay can be eliminated or verified by trying emergency generator operation. If emergency operation can be established with any one generator, the emergency reverse current relay cannot be failed. If this procedure is used, be sure to restore normal operation before attempting subsequent engine starts, since no starts can be made if any generator switch is in the emergency position.

80.4 SERVICING START SYSTEM.

a. Starter - Generator.
   (Refer to Chapter 24.)

b. Engine Ignition System.
   (Refer to Pratt and Whitney
   Maintenance Manual JT12A.)

80.5 MAINTENANCE START SYSTEM.

(Refer to Chapter 24.)

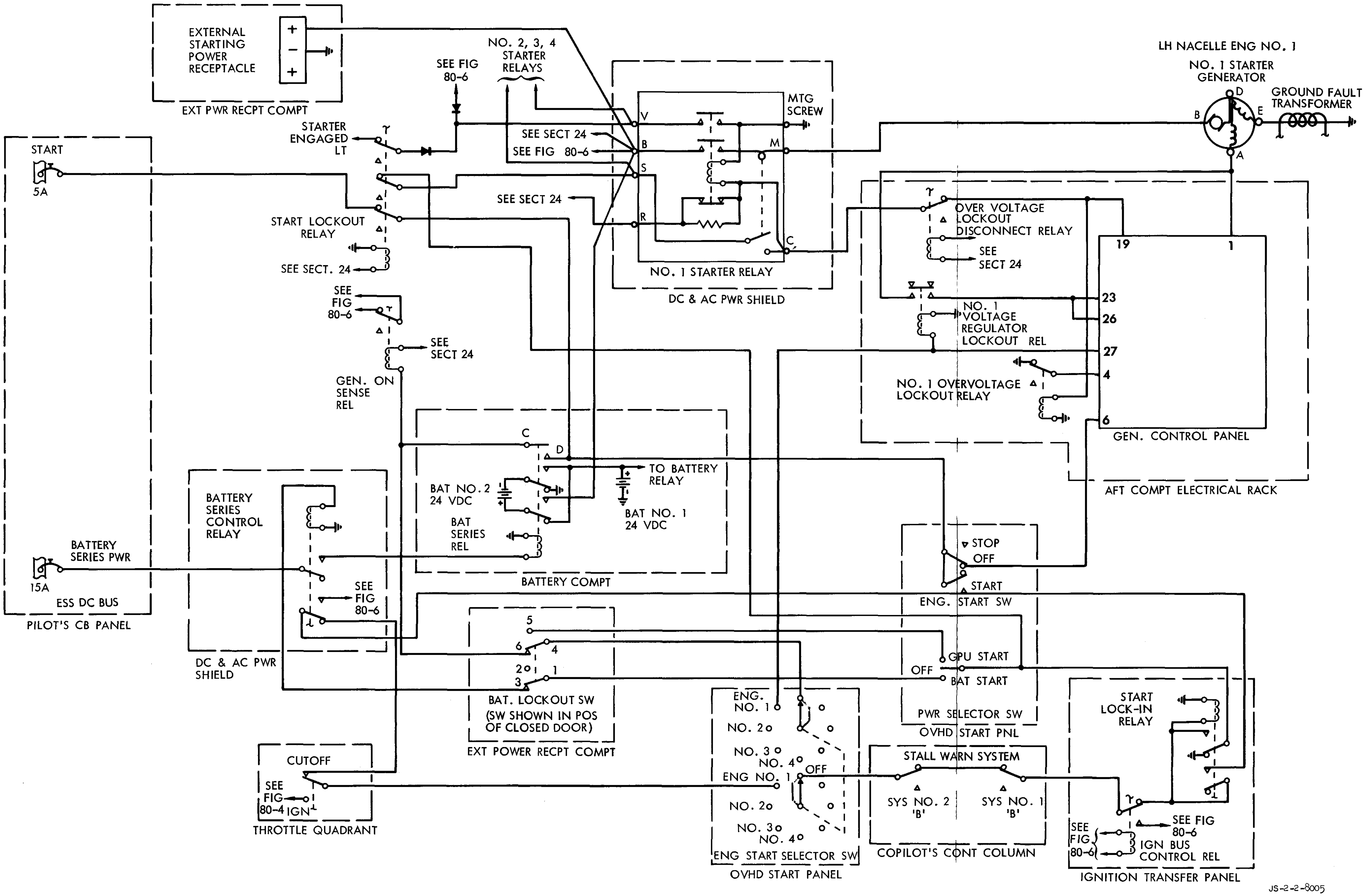
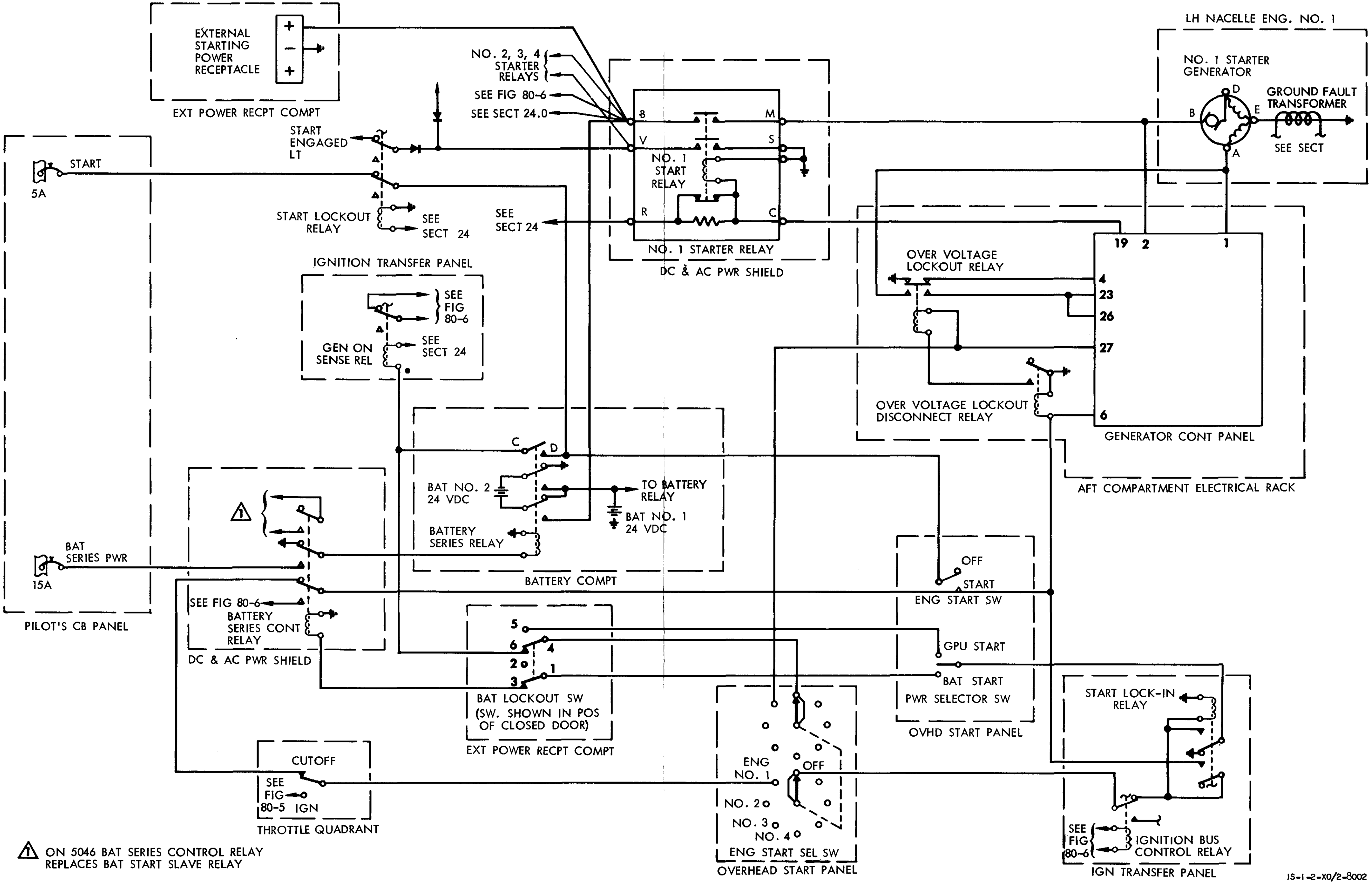
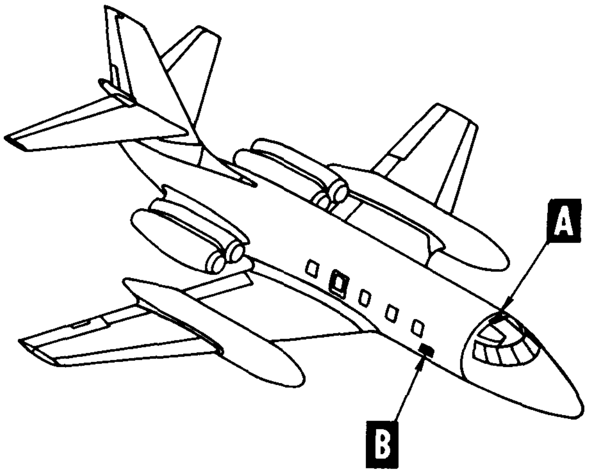


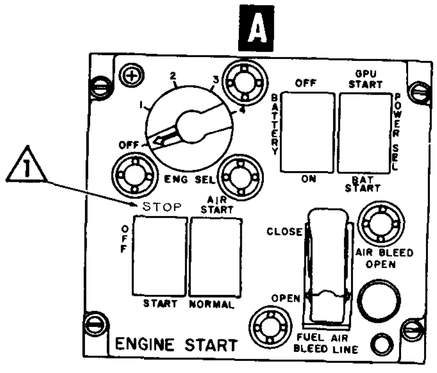
Figure 80-1. Start System Schematic Diagram (Aircraft 5001, 5002, 5004, and 5006)



⚠1 ON 5046 BAT SERIES CONTROL RELAY REPLACES BAT START SLAVE RELAY

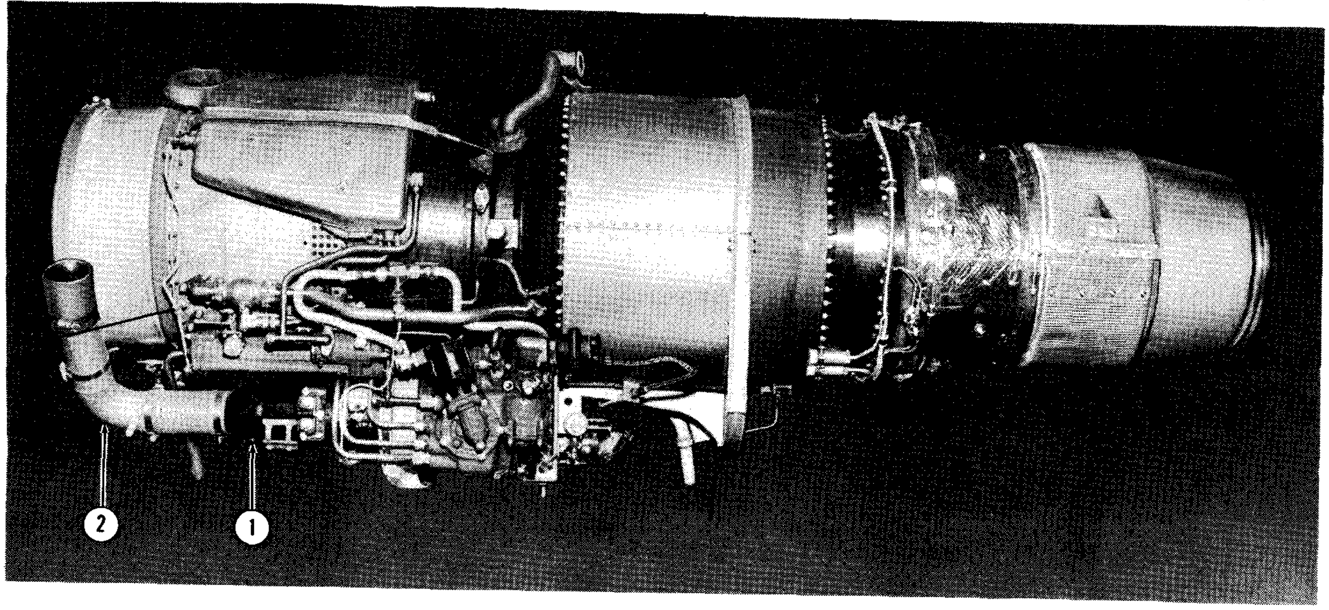
Figure 80-2. Start System Schematic Diagram (Aircraft 5003, 5005, 5007 thru 5055)

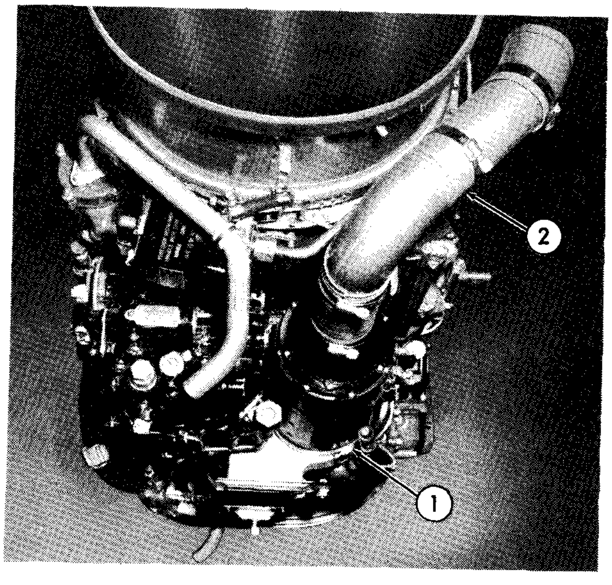


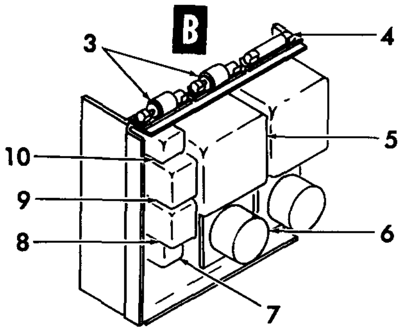


ENGINE START PANEL

⚠1 AIRPLANES 5001, 5002, 5004 & 5006 ONLY.



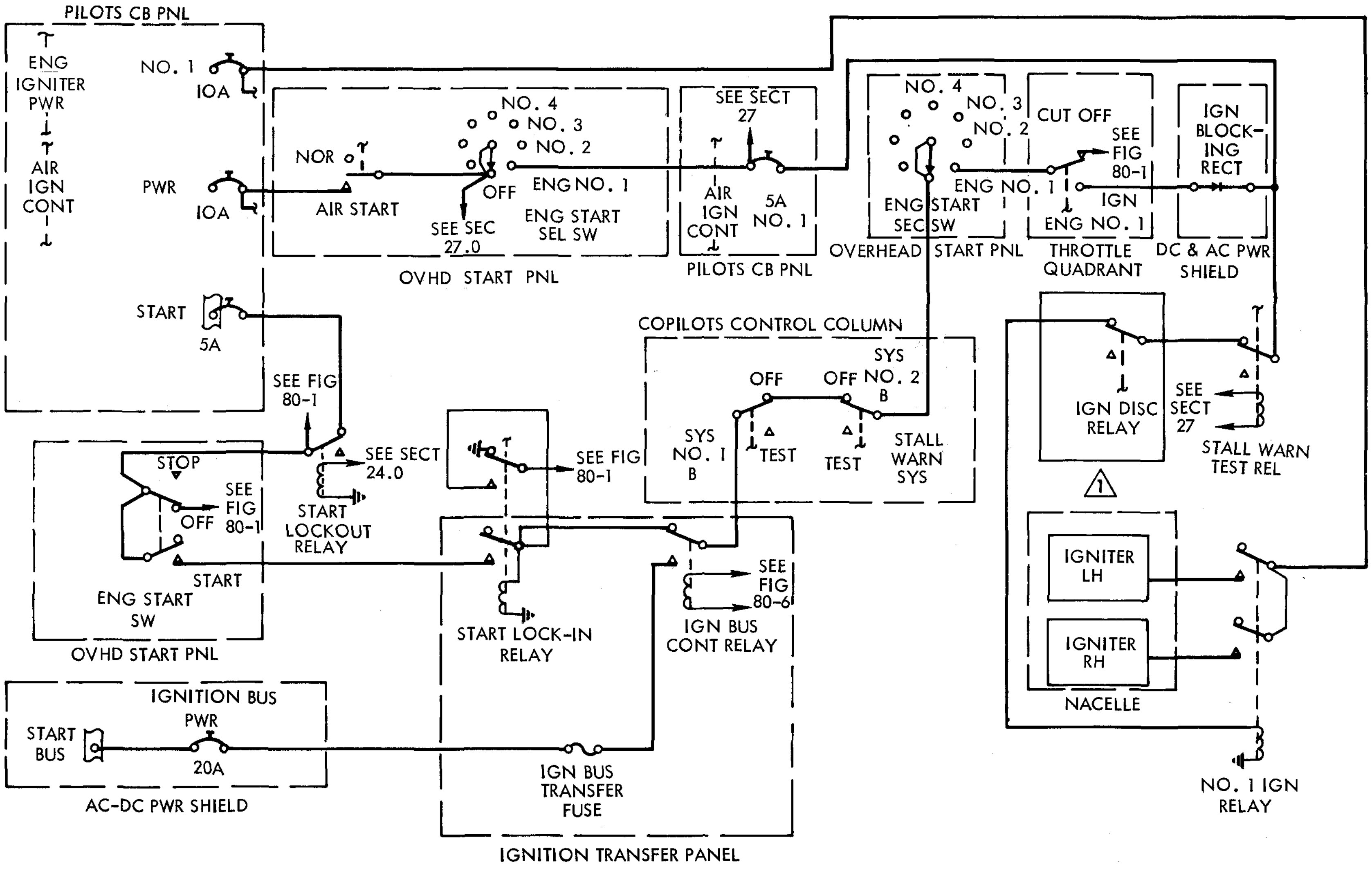




IGNITION TRANSFER CONTROL PANEL

1. STARTER GENERATOR
2. GENERATOR AIRBLAST TUBE
3. NO. 1 & NO. 2 START ENGAGE RECTIFIERS
4. IGNITION BUS TRANSFER FUSE 3A
5. START LOCKOUT RELAY
6. IGNITION BUS OVERVOLTAGE RELAY
7. BATTERY START SLAVE RELAY
8. START LOCK-IN RELAY
9. IGNITION BUS CONTROL RELAY
10. GENERATOR ON SENSE RELAY

Figure 80-3. Engine Start Generator



NOTE

⚠1 ONLY IN CIRCUIT FOR NO. 2 & 3 ENG

JS-1-2-X0/2-8004

Figure 80-4. Ignition Circuit Schematic Diagram (Aircraft 5001, 5002, 5004 and 5006)

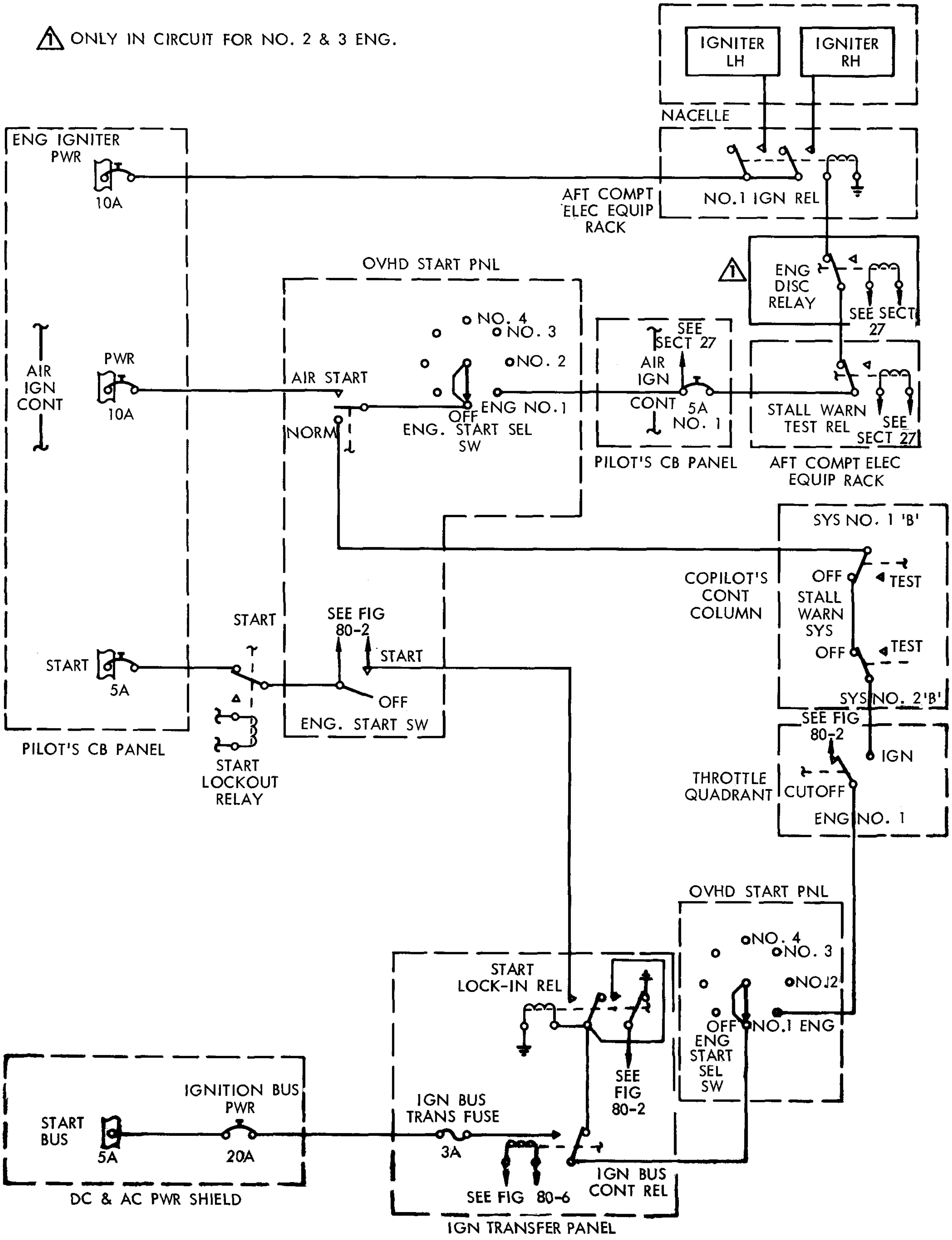
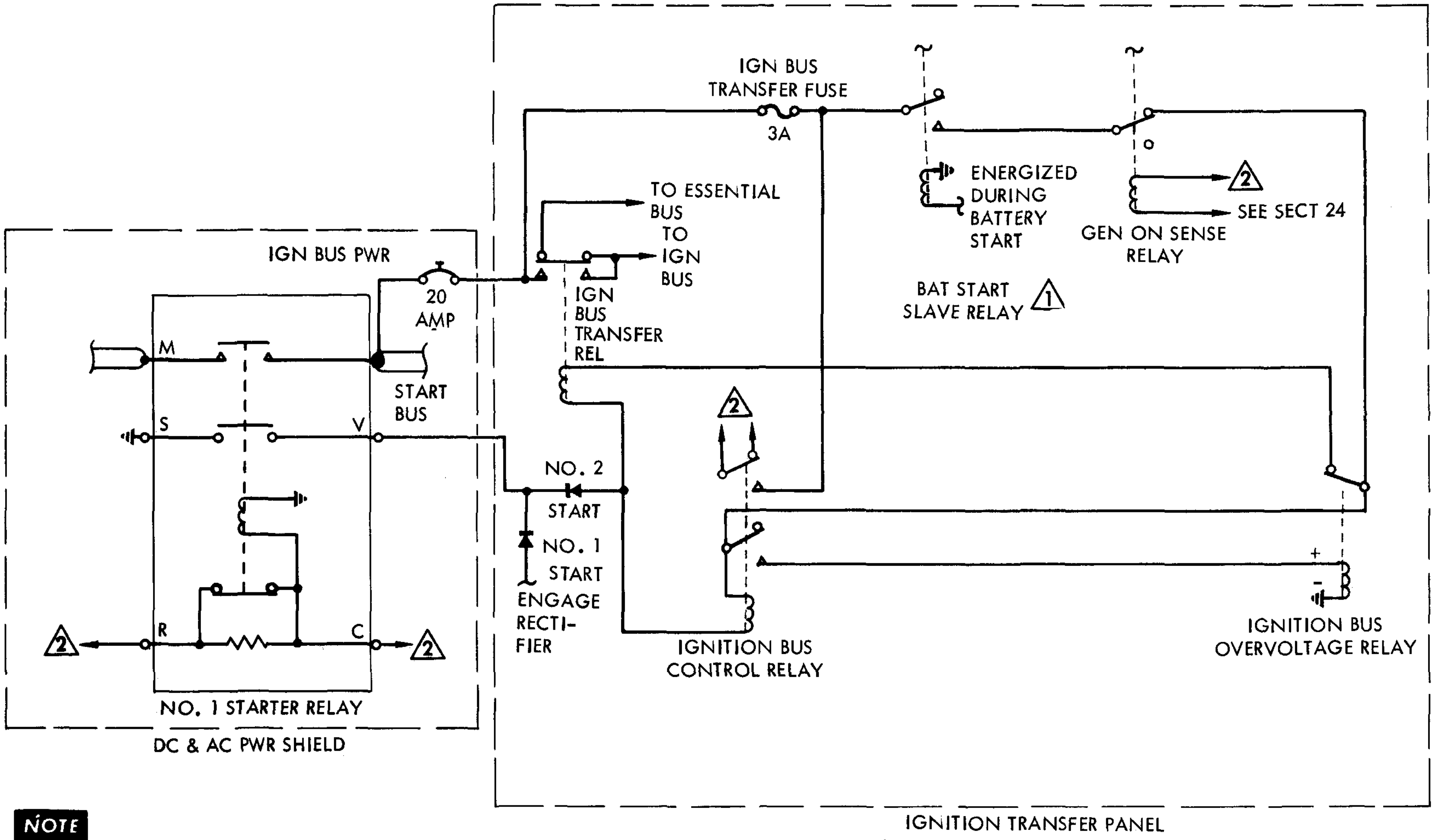

⚠1 ONLY IN CIRCUIT FOR NO. 2 & 3 ENG.

Figure 80-5. Ignition Circuit Schematic Diagram (Aircraft 5003, 5005, 5007 and Up)



NOTE

1 ON AIRCRAFT 5046 & UP BATTERY SLAVE RELAY REPLACED BY BATTERY SERIES CONTROL RELAY

2 SEE FIG 80-1 FOR AIRCRAFT 5001, 5002, 5004 AND 5006.
SEE FIG 80-2 FOR AIRCRAFT 5003, 5005, 5007 AND UP.

JS-2-2-8003

Figure 80-6. Ignition Transfer Circuit Schematic Diagram

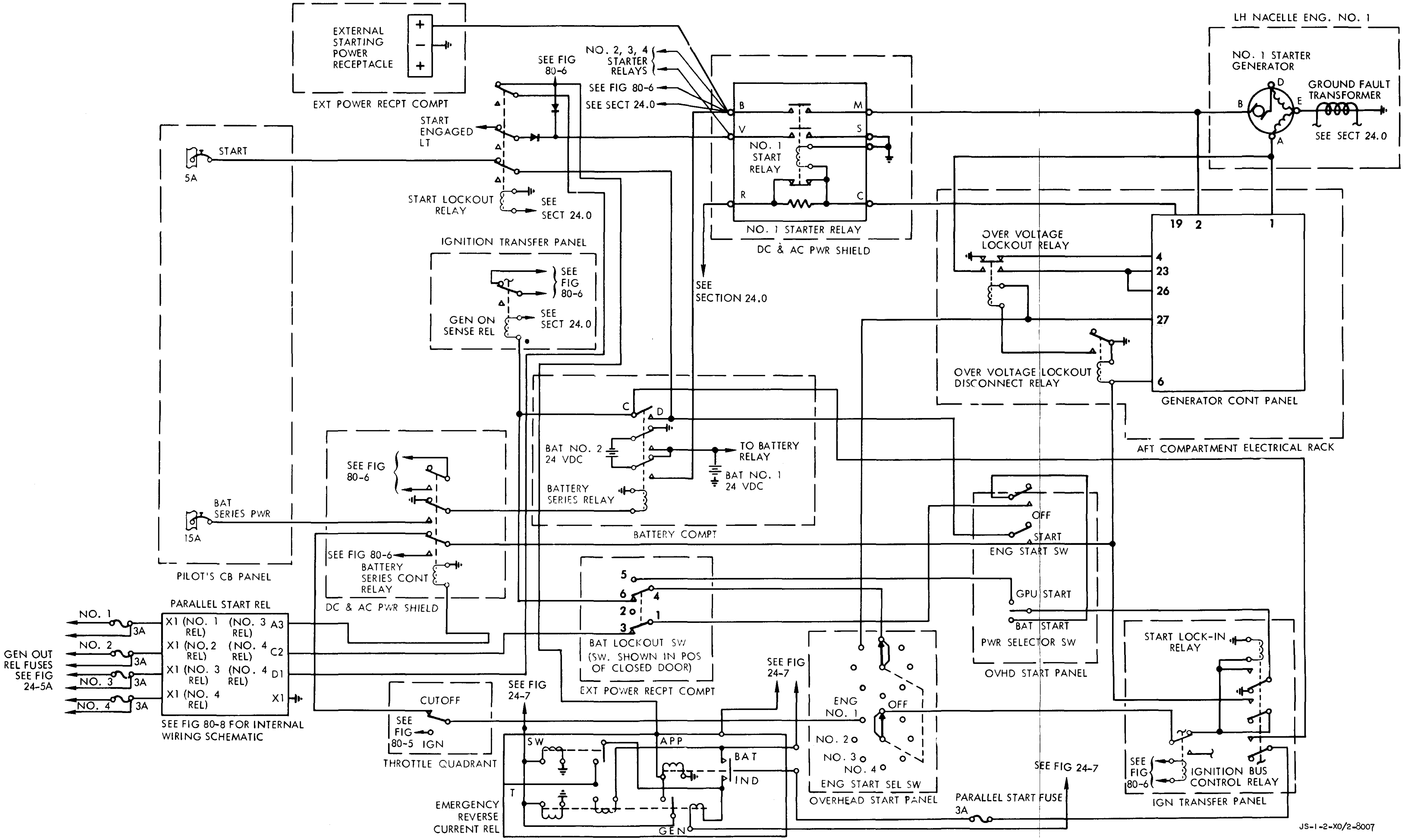


Figure 80-7 Start System Schematic Diagram (Aircraft 5056 Thru 5065)

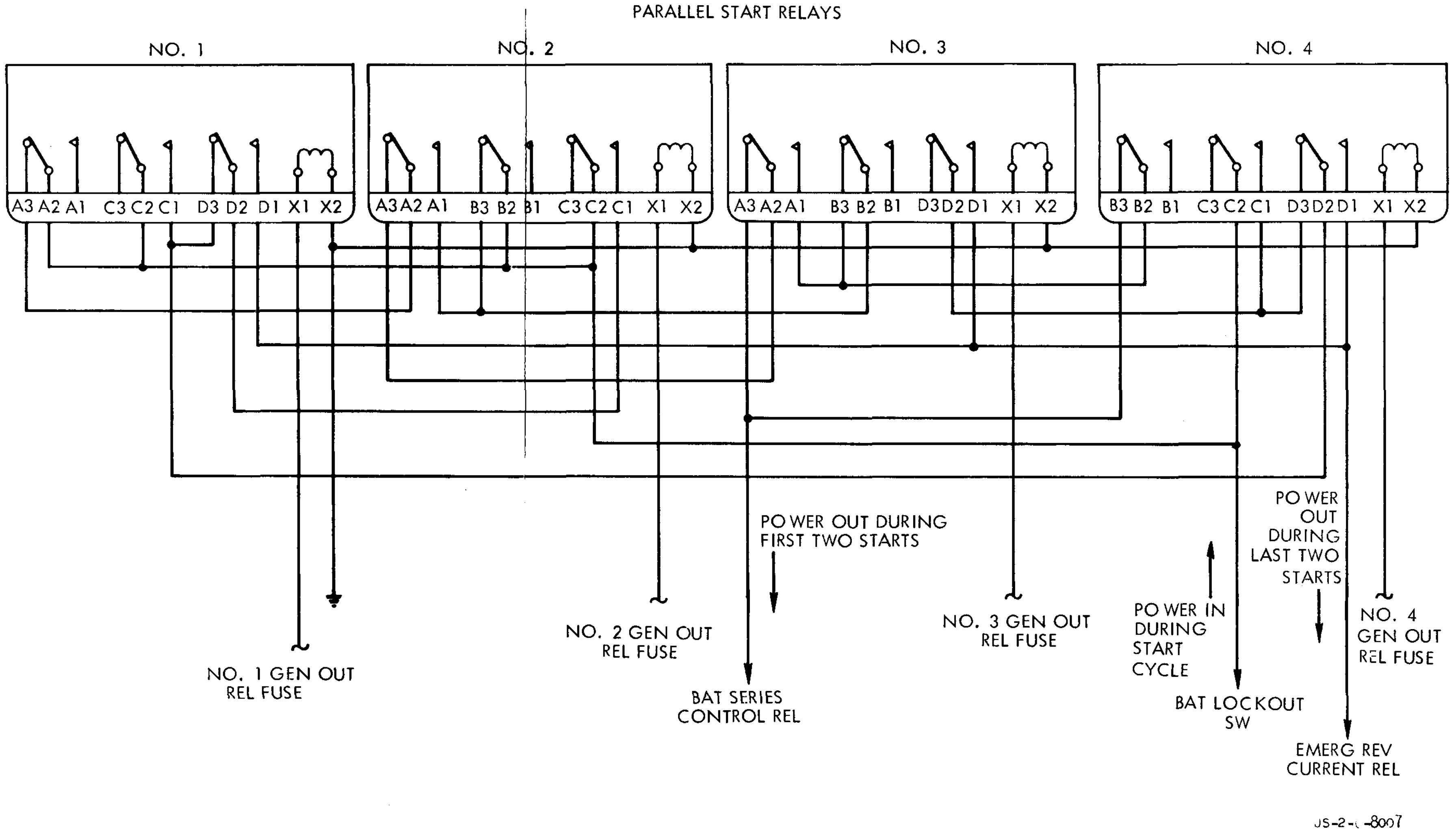

PARALLEL START RELAYS
NO. 1
NO. 2
NO. 3
NO. 4
A3 A2 A1 C3 C2 C1 D3 D2 D1 X1 X2
A3 A2 A1 B3 B2 B1 C3 C2 C1 X1 X2
A3 A2 A1 B3 B2 B1 D3 D2 D1 X1 X2
B3 B2 B1 C3 C2 C1 D3 D2 D1 X1 X2
POWER OUT DURING FIRST TWO STARTS
POWER OUT DURING LAST TWO STARTS
POWER IN DURING START CYCLE
NO. 1 GEN OUT REL FUSE
NO. 2 GEN OUT REL FUSE
NO. 3 GEN OUT REL FUSE
NO. 4 GEN OUT REL FUSE
BAT SERIES CONTROL REL
BAT LOCKOUT SW
EMERG REV CURRENT REL
JS-2- -8007

Figure 80-8. Wiring Schematic of Parallel Start Relays

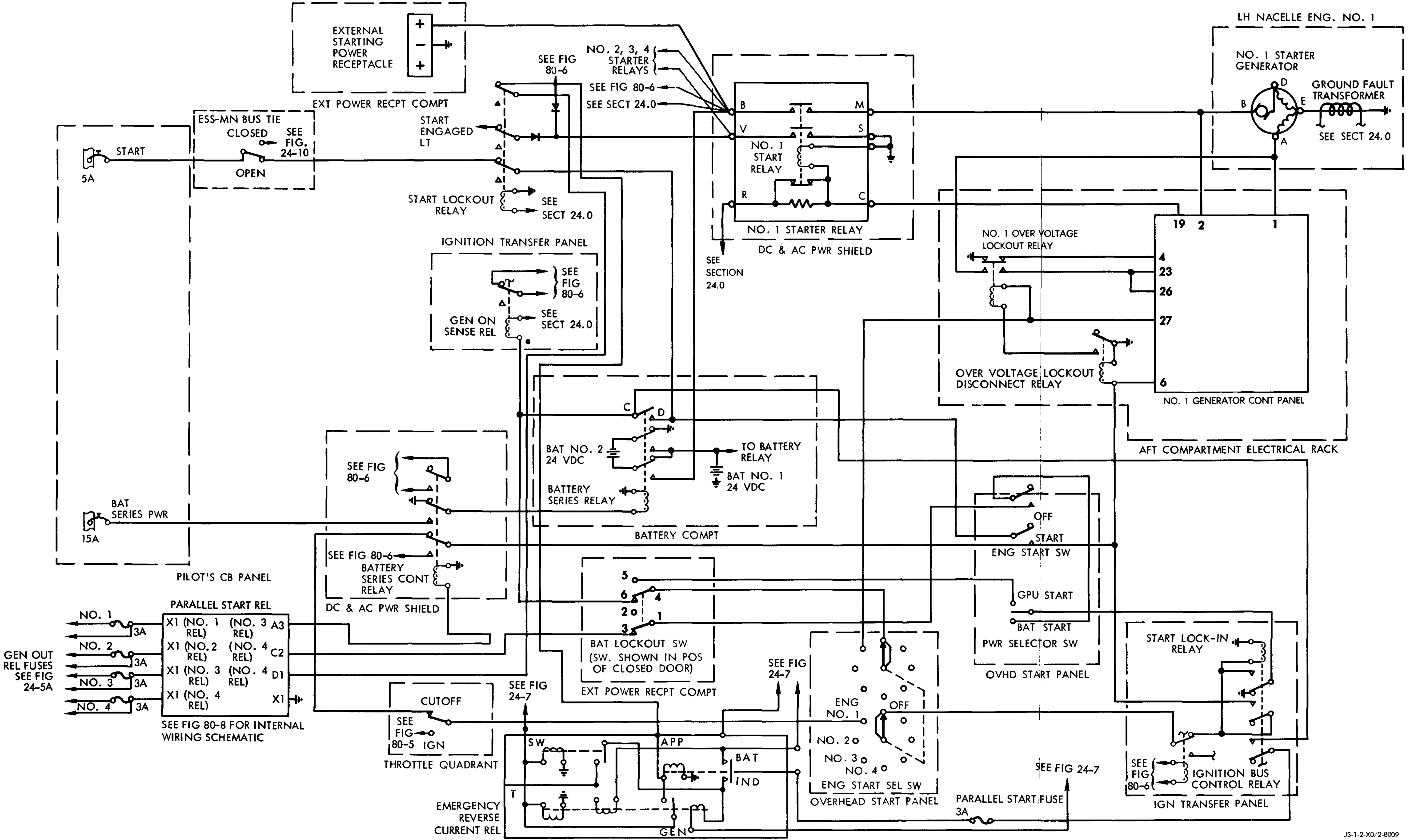
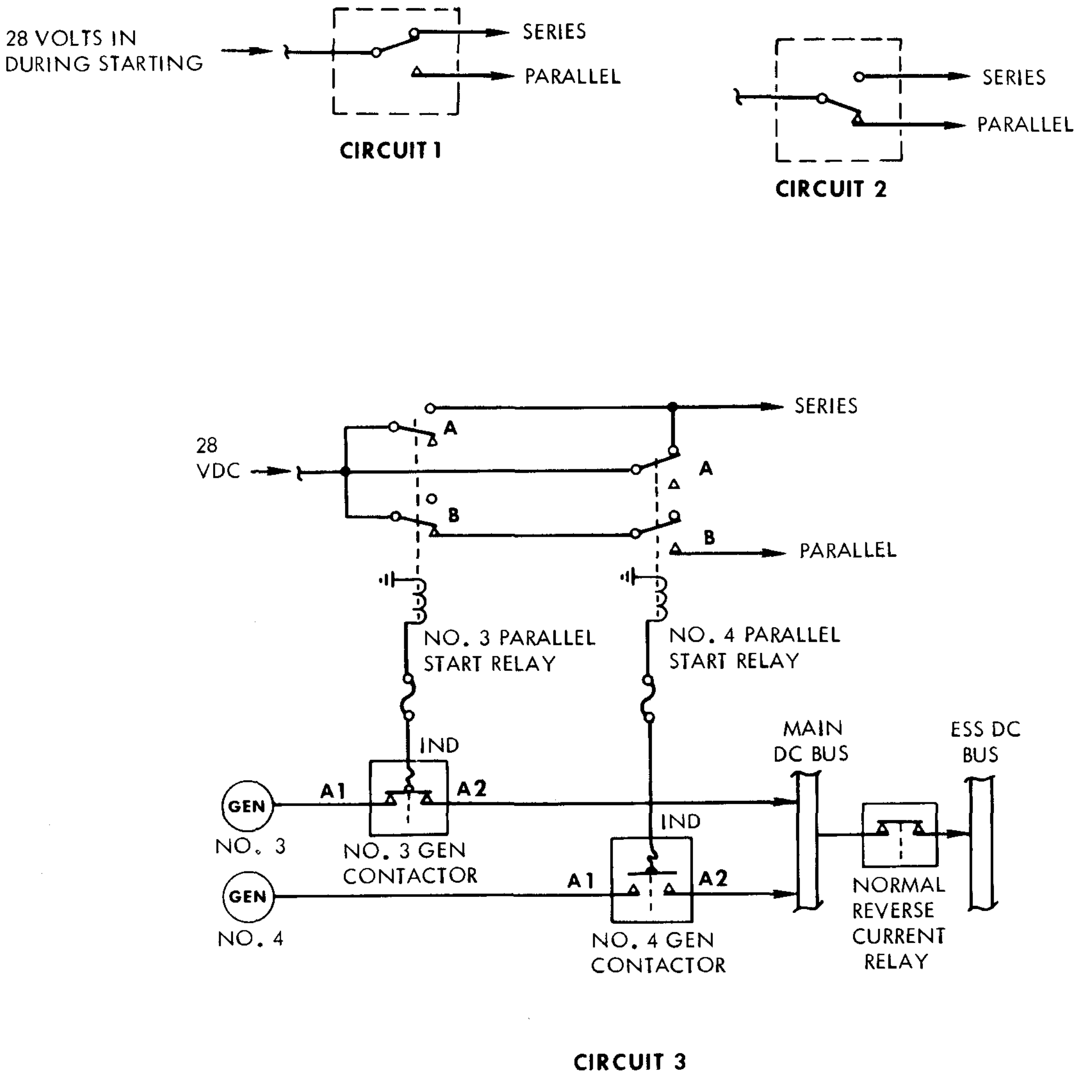


Figure 80-9. Start System Schematic Diagram (Aircraft 5066 and Up)



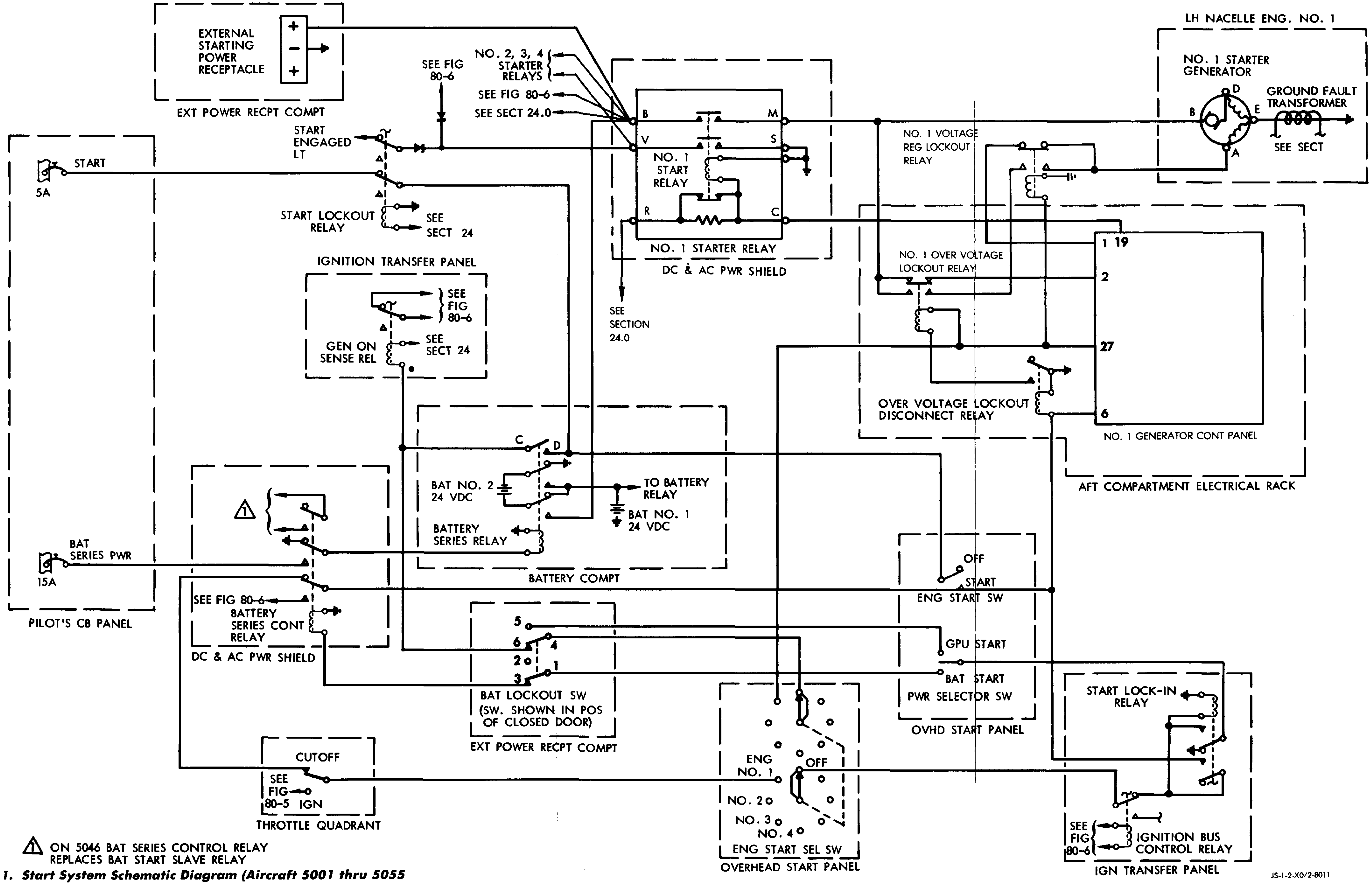
CIRCUIT 1

CIRCUIT 2

CIRCUIT 3

JS-1-2-X0/2-8010

Figure 80-10. Simplified Schematic-Parallel Start Relays



1 ON 5046 BAT SERIES CONTROL RELAY REPLACES BAT START SLAVE RELAY

Figure 80-11. Start System Schematic Diagram (Aircraft 5001 thru 5055 Modified by Service Bulletin 329-294)

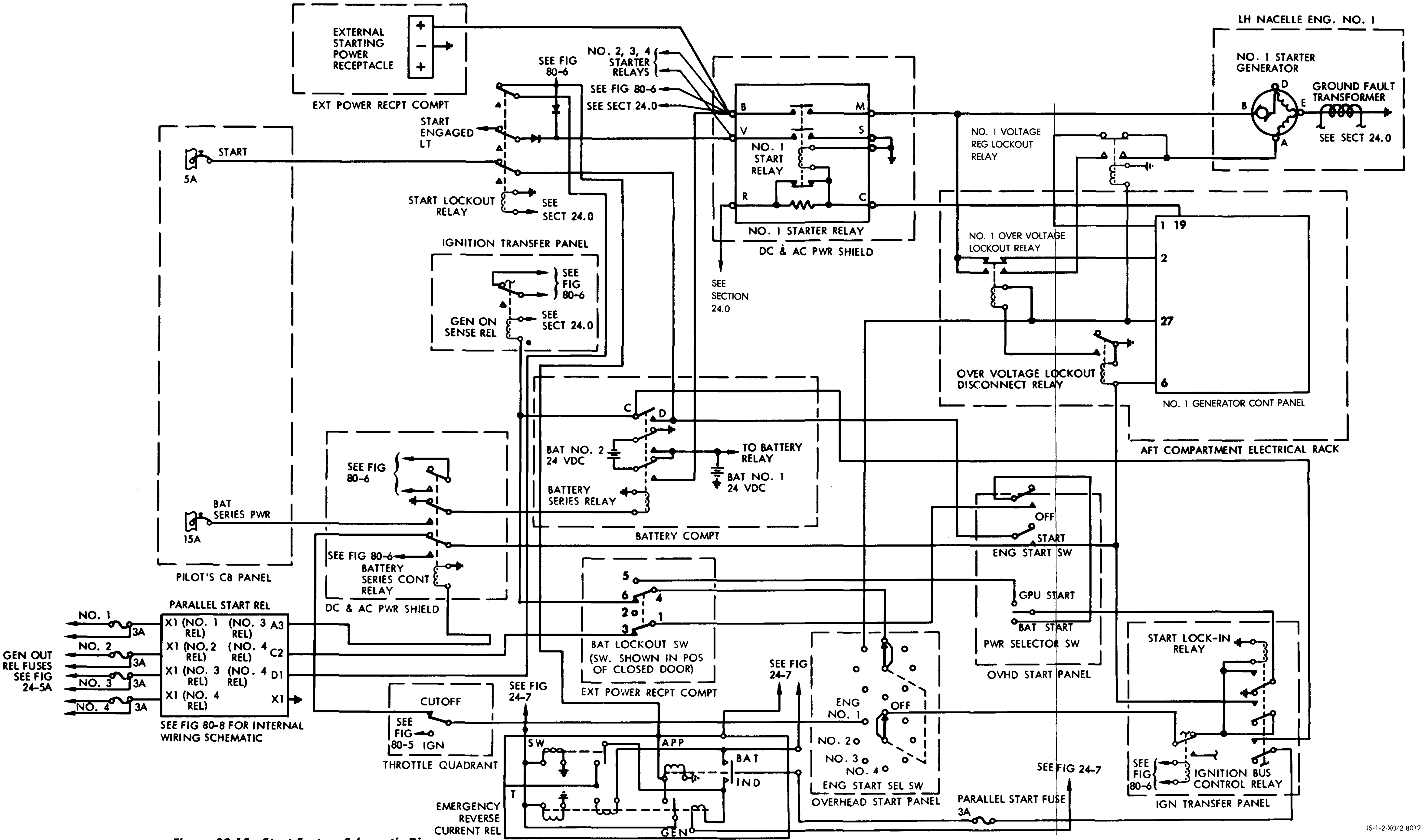


Figure 80-12. Start System Schematic Diagram
(Aircraft 5056 thru 5065 Modified by Service Bulletin 329-294)

Changed 31 August 1983

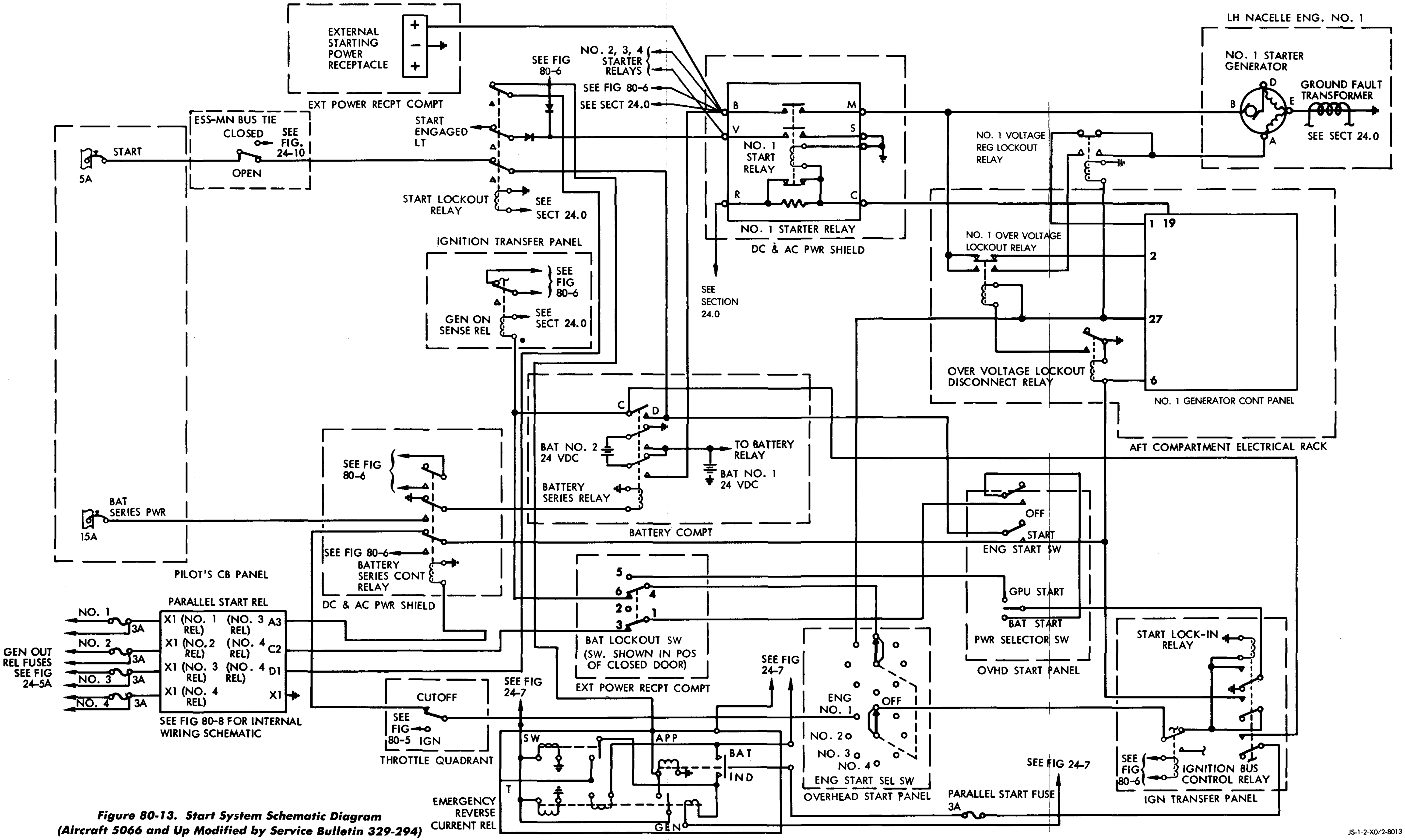


Figure 80-13. Start System Schematic Diagram (Aircraft 5066 and Up Modified by Service Bulletin 329-294)

JS-1-2-X0/2-8013